"The best safety device in any aircraft is a well-trained crew"™

MTM V1 XL/XLS/XLS+ Maintenance Training Manual • Vol. 1 — Second Edition - Rev. 0.3

MTM V2 XL/XLS/XLS+ Maintenance Training Manual • Vol. 2 — Second Edition - Rev. 0.3

MSM XL/XLS/XLS+ Maintenance Schematic Manual — Second Edition - Rev. 0.2

IPP Instrument Panel Poster — CITATION EXCEL IPP — 05.01.03

IPP Instrument Panel Poster — CITATION XLS IPP — 07.11.07

IPP Instrument Panel Poster — CITATION XLS+ IPP — 07.16.08

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CITATION XL/XLS/XLS+ SERIES

Maintenance Training Materials

UNCONTROLLED DOCUMENTS

COLLECTION DATE
27Jan16

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CITATION XL/XLS/XLS+ MAINTENANCE TRAINING MANUAL

VOLUME 1

REVISION 0.3

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INSERT LATEST REVISED PAGES, DESTROY SUPERSEDED PAGES

LIST OF EFFECTIVE PAGES

Dates of issue for original and changed pages are:

Original.......... 0 ....April 2010
Revision .... 0.1 ....August 2010
Revision .... 0.2 ....October 2013
Revision .... 0.3 ....January 2016

NOTE:
For printing purposes, revision numbers in footers occur at the bottom of every page that has changed in any way (grammatical or typographical revisions, reflow of pages, and other changes that do not necessarily affect the meaning of the manual).

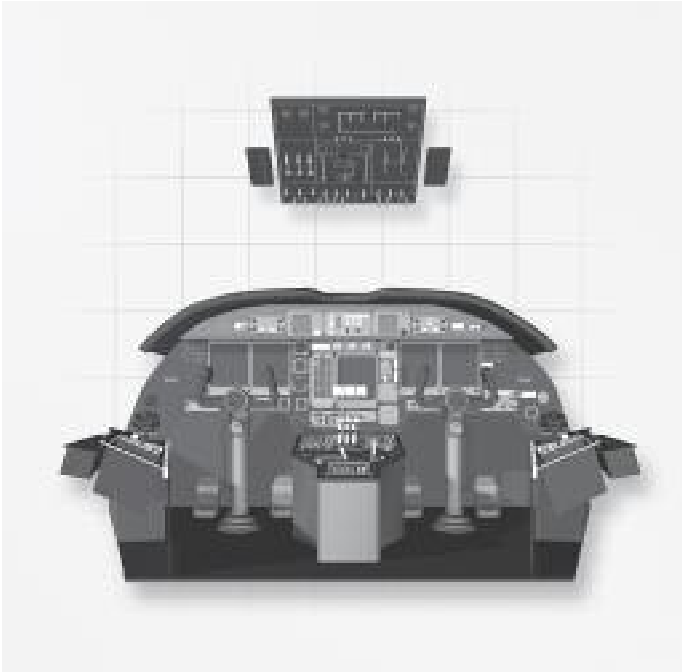
THIS PUBLICATION CONSISTS OF THE FOLLOWING:

Page No.	*Revision No.
Cover—Copyright	0.3
i—iii	0.3
iv—vi	0.2
1-1—1-2	0.2
2-i—2-iv	0.2
2-1—2-20	0.2
5-i—5-viii	0.2
5-1—5-26	0.2
6-1—6-18	0.2
7-1—7-8	0.2
8-1—8-4	0.2
9-1—9-12	0.2
10-1—10-4	0.2
11-1—11-4	0.2
12-1—12-62	0.2
21-i—21-vi	0.2
21-1—21-70	0.2
22-i—22-iv	0.2
22-1—22-18	0.2
23-i—23-iv	0.2
23-1—23-38	0.2
24-i—24-vi	0.2
24-1—24-68	0.2
25-i—25-iv	0.2
25-1—25-26	0.2
26-i—26-iv	0.2
26-1—26-28	0.2
27-i—27-viii	0.2
27-1—27-88	0.2
28-i—28-vi	0.2
28-1—28-52	0.2
28-53	0.3
28-54—28-56	0.2
29-i—29-iv	0.2
29-1—29-5	0.2
29-6	0.3
29-7—29-22	0.2
30-i—30-iv	0.2
30-1—30-34	0.2
31-i—31-vi	0.2
31-i—31-52	0.2

*Zero in this column indicates an original page.

CONTENTS
VOLUME 1

CHAPTER 1
INTRODUCTION



INTRODUCTION

This training manual provides a description of the major airframe and engine systems as installed in the Cessna Citation 560 Excel aircraft. This information is intended as an instructional aid only; it does not supersede, nor is it meant to substitute for, any of the manufacturer's maintenance or operating manuals. This material has been prepared from the basic design data, and all subsequent changes in aircraft appearance or system operation will be covered during academic training and subsequent revisions to this manual.

GENERAL

The first chapter of this manual, "ATA 100,"is an introduction to the Air Transport Association (ATA) format for aircraft maintenance manuals. It is intended to describe simply the basic format for all ATA 100 *Maintenance Manual* chapters and also to explain where variations may exist from one manufacturer to another.

Each chapter following "ATA 100" of this book has listed on the divider tab the ATA chapter(s) included, such as "24 Electrical." In some cases it is appropriate, for training purposes, to include more than one ATA chapter in one chapter of this book, such as Chapters 5 through 12 in "Aircraft General." The tab marked "Aircraft General 5–12" indicates that applicable ATA 100 *Maintenance Manual* Chapters 5 through 12 are covered in that chapter. Any chapter not included in the manufacturer's *Maintenance Manual* for that particular aircraft is not included in that chapter of this training manual.

Appendix A in this manual contains a pictorial Walkaround on a Cessna Citation 560 Excel aircraft. Appendix B displays all light indications and can be folded out for reference while reading this manual.

The goal of this course is to provide the very best training possible for the clients in our maintenance initial program. So there is no uncertainty about what is expected of the client, the following basic objectives are presented for this course.

Given the *Maintenance Manual*, class notes, and this training manual (as specified by the FlightSafety instructor), the client will be able to pass a written examination upon completion of this course to the grading level prescribed by the FlightSafety Director of Training. The maintenance technician will be able to:

- Outline the ATA 100 system of maintenance documentation, including the major chapter headings and symbology.
- Describe the meaning and application of each piece of manufacturer's maintenance documentation and use the documentation in practical applications.
- Outline the recommended maintenance schedule and applicable options.
- Locate major components without reference to documentation and other components with the aid of documentation.
- Describe the operation of all major systems in the normal and various abnormal operating modes.
- Perform maintenance preflight and postflight inspections.
- Perform selected normal and emergency cockpit procedures as required for engine start/run-up, APU start, battery check, aircraft taxiing, etc. (requires use of a simulator).

The FlightSafety instructor will modify the stated overall objective conditions and criteria to satisfy selected performance requirements, when appropriate. The performance levels specified will not vary from those directed by the FlightSafety Director of Training.

NOTES

ATA 100

CONTENTS

ILLUSTRATION

TABLES

ATA 100

INTRODUCTION

The purpose of this chapter is to describe the arrangement, numbering system, and special features of the Air Transport Association (ATA) format for aircraft maintenance manuals. To take advantage of all the material presented in an ATA 100 manual, the maintenance technician must become thoroughly familiar with the outline and contents presented for any given airplane.

GENERAL

The Cessna Citation 560XL/XLS/XLS+ *Maintenance Manual, Illustrated Parts Catalog*, and *Wiring Diagram Manual* are prepared in accordance with the Air Transport Association Specification No. 100 for manufacturer's technical data.

These manuals have been prepared to assist maintenance personnel in servicing and maintaining Citation airplanes. They provide the necessary information required to enable the mechanic to service, inspect, troubleshoot, remove and replace components, or repair systems.

Information beyond the scope of these manuals may be found in the Cessna Citation *Overhaul Manual, Structural Repair Manual*, Tool and Equipment List, or *Component Maintenance Manuals*.

Model 560XL/XLS aircraft are delivered with a complete set of avionics wiring diagrams specifically prepared for that serial number (SN). These diagrams, which are to be carried aboard the airplane, must be used in conjunction with the *Aircraft Maintenance Manual (AMM)* when performing maintenance on the airplane.

Technical publications available from the manufacturer of the various components and systems which are not covered in the *AMM* must be utilized as required for maintenance of those components and systems.

These manuals have been designed for aerofiche presentation. To facilitate the use of the manual for aerofiche, fiche/frame numbers have been added to the various tables of contents, and alphabetical and numerical indexes as applicable. Refer to the header of the applicable fiche for location of various indexing information.

MAINTENANCE MANUAL

TEMPORARY REVISION

Additional information that becomes available may be provided by a temporary revision. This service provides, without delay, new information to assist in maintaining safe flight/ground operations. Temporary revisions are numbered consecutively within the ATA chapter assignment and page numbering, utilizing the three-element number, which matches the manual. Temporary revisions are normally incorporated into the manual at the next regularly scheduled revision.

REGULAR REVISION

Pages to be removed or inserted in the manual are controlled by the effectivity page. Pages are listed in sequence by the three-element number (chapter/section/ subject) and then by page number. When two pages display the same three-element number and page number, the page displaying the most recent “date of page issue” shall be inserted in the manual. The date column on the corresponding chapter effectivity page verifies the active page.

Revision Bars

Additions, deletions, or revisions to text in an existing section are identified by a revision bar in the left margin of the page adjacent to the change.

When technical changes cause unchanged text to appear on a different page(s), a revision bar is placed in the margin opposite the page number of all affected pages, provided no other revision bar appears on the page. These pages updated to the current regular revision date.

When extensive technical changes are made to text in an existing section that requires complete retype of the copy, revision bars appear full length of the text.

When art in an existing illustration is revised, a pointing hand appears in the illustration pointing to the area of the art revision.

New art added to an existing section is identified by a single pointing hand adjacent to the diagram title.

If using manuals on DVD, revisions are highlighted in light blue.

List of Effective Pages

A list of effective pages is provided with each manual chapter. All pages in the chapter are listed in sequence with the most recent revision date for each page. A revised list of effective pages is provided for each chapter with every regular manual revision.

DIVISION OF SUBJECT MATTER

The Model 560XL/XLS *AMM* is divided into four major sections. The major sections are in turn separated into chapters with each chapter having its own effectivity page and table of contents. The manual divisions are as follows:

Major Section 1—Airplane General

Chapter	Title
4	Airworthiness Limitations
5	Time Limits/Maintenance Checks
6	Dimensions and Areas
7	Lifting and Shoring
8	Leveling and Weighing
9	Towing and Taxiing
10	Parking, Mooring, Storage and Return to Service

Major Section 2—Airframe Systems

Chapter	Title
11	Placards and Markings
12	Servicing
20	Standard Practices—Airframe
21	Air Conditioning
22	Auto Flight
23	Communications
24	Electrical Power
25	Equipment/Furnishings
26	Fire Protection
27	Flight Controls
28	Fuel
29	Hydraulic Power
30	Ice and Rain Protection
31	Indicating/Recording Systems
32	Landing Gear
33	Lights
34	Navigation
35	Oxygen
36	Pneumatic
37	Vacuum
38	Water/Waste

Major Section 3—Structures

Chapter	Title
51	Standard Practices and Structures—General
52	Doors
53	Fuselage
54	Nacelles/Pylons
55	Stabilizers
56	Windows
57	Wings

Major Section 4—Power Plant

Chapter	Title
71	Power Plant
73	Engine Fuel and Control
74	Ignition
76	Engine Controls
77	Engine Indicating
78	Exhaust
79	Oil
80	Starting

PAGE NUMBERING SYSTEM

The page numbering system used in the *AMM* consists of three element numbers separated by dashes, under which the page number and date are printed.

28-21-01

Chapter/System (Fuel) — Section/Subsystem — Subject/Unit (Fuel Boost Pump)

When the chapter/system element number is followed by zeros in the section/subsystem and subject/unit element number (28-00-00), the information is applicable to the entire system.

When the section subsystem element number is followed by zeros in the subject/unit element number (28-21-00), the information is applicable to subsystems within the system.

The subject/unit element number is used to identify information applicable to units within the subsystems. The subject/unit element number progresses sequentially from the number "01" in accordance with the number of subsystem units requiring maintenance information.

All system/subsystem/unit (chapter/section/subject) maintenance data is separated into specific types of information: description and operation, troubleshooting, and maintenance practices. Blocks of sequential page numbers are used to identify the type of information:

- Pages 1 through 99—Description and Operation
- Pages 101 through 199—Troubleshooting
- Pages 201 through 299—Maintenance Practices

Relatively simple units may not require description and operation and/or troubleshooting information; in such cases, unused page number blocks are omitted. In addition, for those items requiring many types of maintenance practices, page block 201 through 300 is omitted, and page numbering maintenance practices are broken out as follows:

- Pages 301 through 399—Servicing
- Pages 401 through 499—Removal/Installation
- Pages 501 through 599—Adjustment/Test
- Pages 601 through 699—Inspection/Check
- Pages 701 through 799—Cleaning/Painting
- Pages 801 through 899—Approved Repairs

A typical page number:

Distribution Subsystem

Fuel System — Fuel Boost Pump Unit

28-21-01
Page 202
Jan 1/78

Second Page of Fuel Boost Pump Unit Maintenance Practices — Date of Page Issued

Illustrations use the same figure numbering as the page block in which they appear. For example, Figure 202 would be the second figure in a "Maintenance Practices" section.

WARNINGS, CAUTIONS, AND NOTES

Throughout the text in the manuals, there are warnings, cautions, and notes pertaining to the procedures being accomplished. These additions to the text highlight or emphasize important points when necessary:

- WARNING—Calls attention to use of materials, processes, methods, procedures, or limits that must be followed precisely to avoid injury or death to persons.
- CAUTION—Calls attention to methods and procedures that must be followed to avoid damage to equipment.
- NOTE—Calls attention to methods that make the job easier.

ILLUSTRATED PARTS CATALOG

PAGE NUMBERING SYSTEM

The page numbering system used in the *Illustrated Parts Catalog* consists of three-element numbers separated by a dash, under which the page number and date are printed.

Section/Subsystem
(Flap System)

Chapter/System
(Flight Controls)

Unit
(Flap Actuator)

27-50-01

Page 2—Second Page of Flap (Flap Actuator Assembly)

Date of Page Issue—Jan 1/91

The pages of this manual are numbered so that the illustration page faces the text page, with corresponding index numbers:

- The first page of text and illustration reflects index numbers 1 through 74.
- The second illustration and text page reflects index 75 through 149.
- The third illustration and text page reflects index 150 through 224.
- The fourth illustration and text page reflects index 225 through 299.

GENERAL SYSTEM OF ASSEMBLY ORDER

The indention system used in the detailed parts list of this catalog shows the relationship of one part to another. For a given item, the indentation code shows a system, installation, or general heading starting in the extreme left position continuing on down into succeeding columns until the end detail is reached, as follows:

1 2 3 4 5

Installation

  Detail Parts for Installation

  Assembly

  Attaching Parts for Assembly

    Detail Parts of Assembly

    Subassembly

    Attaching Parts for Subassembly

      Detail Parts for Subassembly

      Sub-Subassembly

      Attaching Parts for Sub-Subassembly

        Detail Parts for Sub-Subassembly

NUMERICAL INDEX (PAPER ONLY)

The numerical index is a complete listing of all parts included in the detailed parts list and shows in reverse, as well as forward, all information relative to superseded parts. When a part is superseded for full effectivity at a specific location, both the superseding and superseded parts are listed. The superseding part number is listed with the note "Supersedes (superseded P/N)." All part numbers are cross-referenced to the applica-

ble chapter, section, figure, and item number within the detailed parts list.

Abbreviations:

- ALT—Alternate
- AR—As required
- ASSY—Assembly
- BKI—Bulk item
- FS—Fuselage station
- FSO—For spares order
- LH—Left
- NP—Not procurable
- RF—Reference
- RH—Right
- WEU—When exhausted use
- WS—Wing station

CESSNA PART NUMBERING SYSTEM

The basic number identifies the Cessna drawing only. Each installation, assembly, or detail part is assigned a part number that consists of the drawing number and an appropriate dash number.

Example:

Part Number
6515300-1

Basic Number (Drawing Identification Only)
Dash Number

How to find a part:

- When the part number is unknown:
  1. Turn to Alpha Index.
  2. Refer to main group in which part should be listed.
  3. Find the chapter, section, unit, and figure number in which the part should be shown.
  4. Turn to the illustration and find the part.
  5. Refer to corresponding item number in the parts list.
- When the part number is known:
  1. Find the part number in numerical index. Note chapter, section, unit, figure, and item number.
  2. Turn to chapter, section, unit, and figure.
  3. Locate part on illustration and in parts list by item number.

WIRING DIAGRAM MANUAL

WIRING DIAGRAM NUMBERING

Wiring diagram numbering is in accordance with ATA Specification 100. On Citation wiring diagrams, this number is shown as three sets of two numbers (e.g. 28-20-01).

Chapter (Fuel)
Subsystem (Distribution)
Specific Diagram (Left Fuel Distribution)

28-20-01

WIRING DIAGRAM PAGE NUMBERING

The page numbers 1 to 100 are used to number wiring diagrams. Sheet numbers are used in the title block for diagrams that require more than one page for illustration.

NOTE

Whenever a diagram is referenced to another, the diagram number only is used. Therefore, where there is more than one page of the same diagram, it is necessary to refer to the effectivity block to make certain the diagram applies to the aircraft of interest.

The page numbers 101 and subsequent are used for schematics. Sheet numbers are used for schematics that require more than one page for illustration.

WIRE IDENTIFICATION

The wiring diagrams in the 560XL manual *do not* show the wire number for each wire; however, the wires in the airplane have wire numbers as shown in the following example:

/ * BPT481-PJT482(24)(SP 2-B)

- B — Color of Wire
- 2 — Numerical Sequence**
- SP — TP—Twisted Pair; TT—Twisted Triple; SS—Single Conductor Shielded; SP—Shielded Pair; ST—Shielded Triple; CX—Coax
- (24) — Wire Size
- JT482 — Connector (or Other Component)
- P — Connector Pin
- PT481 — Connector (or Other Component)
- B — Connector Pin
- * — Connector Pin Following is Lower Case
- / — Indicates Wire is Spliced with Other Wires within 3 Inches of Component

**A numerical sequence for each type of wire (SP1, SP2, SP3; ST1, ST2, ST3) on a diagram page. May be used to identify wires within a shield or twisted group when they are not drawn adjacent and enclosed by a twisted or shielded symbol. Example: SP2-B and SP2-W are the two wires within a shield.

Thermocouple leads are banded for identification and are color coded (Table 2-1).

Table 2-1. THERMOCOUPLE LEAD CODES

COLOR CODING	WIRE MATERIAL
Green Tracer	Alumel
White	Chromel
Yellow	Constant
Red	Copper
Black	Iron

ATA 100

Table 2-2. EQUIPMENT DESIGNATORS

DESIGNATOR LETTER	CATEGORY ASSIGNED
A	Ammeters
AD	Warning Horns
BATT	Batteries
BOTTLE	Extinguisher Bottles
C	Capacitor
CB	Circuit Breaker
CS	Current Sensor
CT	Current Transformer
CU	Control Units
D	Diodes
E	Component Mounting Board
F	Fuse
FL	Fuse Limiters
H	Heaters
IND	Indicators
INV	Inverters
J	Receptacles
K	Relays
L	Lights
M	Motors
P	Plugs
PCB	Printed Circuit Board
PJ	Headset/Microphone Jack Audio System
Q	Transistor
R	Resistor
SG	Starter-Generator
SH	Shunt
SL	Solenoid
SLV	Solenoid Valve/Motor-Operated Valve
SP	Splice
S	Switch
TB	Terminal Board
U	Integrated Circuit
V	Voltmeter
VR	Voltage Regulator
XMTR	Transmitter

EQUIPMENT LIST

All electrical and electronic equipment in the wiring diagrams are identified by an alphanumeric reference designator. This reference designator is a cross-reference symbol to the equipment list where the part number, part description, zone, and five-digit Federal Supply Code are given. Manufacturers' names and addresses that correspond with the Federal Supply Code may be found in the Introduction.

The model 560XL/XLS/XLS+ has three methods for assigning a reference designator to a component. Method one is one to four letters followed by one to three numbers. A reference designator assigned by this method has no relationship to where a component is located in the airplane.

The list in Table 2-2 shows the categories assigned to the basic equipment designator letters:

Method two is two letters followed by three numbers. A reference designator assigned by this method provides information about where a component is located in the airplane. The following list shows the categories assigned to basic equipment designator letters by method two:

COLUMN A EQUIPMENT ITEM

Splice	A%###
Controller	B%###
Capacitor	C%###
Diode	D%###
Instrument	E%###
Light	F%###
Ground	G%###
CB/Fuse/Bus Bar	H%###
Receptacle	J%###
Relay	K%###
Inductor	L%###
Servo/Motor	M%###
Printed CKT BD	N%###
Plug	P%###
Transistor	Q%###
Resistor	R%###
Switch	S%###
Thermal	T%###
Element Unit/Module	U%###
Valve	V%###
Solenoid	W%###
JCT/Terminal BD	X%###
Cessna Assembly	Z%###

COLUMN B LOCATION IN AIRPLANE

Aft Bulkhead	*A###
FWD Fuselage Bulkhead	*B###
Fuselage (Cabin)—LH	*C###
LH Engine (Nacelle)	*D###
RH Engine (Nacelle)	*E###
Fuselage (Cabin)—RH	*F###
Landing Gear	*G###
Horizontal Stabilizer	*H###
Instrument Panel	*I###
LH Wing	*L###
AFT Baggage	*K###
LH Wing Feedthrough	*M###
Forward Nose	*N###
RH Wing	*R###
RH Wing Feedthrough	*S###
Tail cone	*T###
Vertical Stabilizer	*V###
**Insertion Cable	*X###
Fuselage Fairings	*Y###
Inside Cessna Assembly	*Z###

Code:	%—Letter from Column B *—Letter from Column A #—0, 1, 2, 3, 4, 5, 6, 7, 8, or 9 (001 to 299 to be used for electrical diagrams; 300 to 999 to be used for avionics diagrams)
Examples:	PB105 Forward Bulkhead Plug EI302 Instrument in Instrument Panel
**Definition:	Insertion Cable—A cable that originates from optional equipment and plugs into a break in a standard cable, and will have an "XX" reference designator as indicated above.

The reference designator codes for a bulkhead feedthrough receptacle and its attaching plugs shall contain the same numeric value. For example:

- JS140—Receptacle, right wing feedthrough
- PR140—Plug in right wing
- PF140—Plug in right cabin

Method three is three letters followed by two numbers and is used for wire ground blocks only.

The reference designators assigned to the ground blocks reference the location, type of ground, and No. 1 or No. 2 system:

C CS 1 0

- 0 — The second number numerically identifies the ground
- 1 — The first number indicates either No. system
- CS — The second and third let indicate the type of grou
  - DC—Direct current
  - AC—Alternating current
  - SG—Signal
  - CS—Chassis-shield
  - LG—Logic
  - GS—Ground stud
- C — The first letter refers to location:
  - A—Nose
  - C—Cabin
  - T—Tail

Additional part numbers are provided for some components by adding extra letters to the basic reference designator.

Examples:

- L63LB—Light bulb used in light L63
- S24LB—Light bulb used in switch S24
- TB2A—Special terminal for thermocouple wire used on terminal board TB2
- P33-B—Backshell used on connector P33

When using the wiring diagram:

- All operable electrical components, such as switches, relays, etc., are shown with the airplane on the ground, all circuits off or deenergized, and no electrical power on the circuits.
- The equipment list consists of two test lines. However, some equipment does not use both lines. Make certain that both lines are observed as part descriptions are not always complete on the first line.
- Some wire diagrams have too many paths to list on one page of text (Examples: L 24-30-01 Figure 1, LH DC Power Distribution and Start; 33-10-04 Figure 2, Panel Light Inverters and Control; 39-20-02 Figure 1, RH Circuit-Breaker Panel). In this case, identical wire diagrams are used with a different parts list for each diagram.

Figure 1-1 illustrates the symbology used on must wire diagrams.

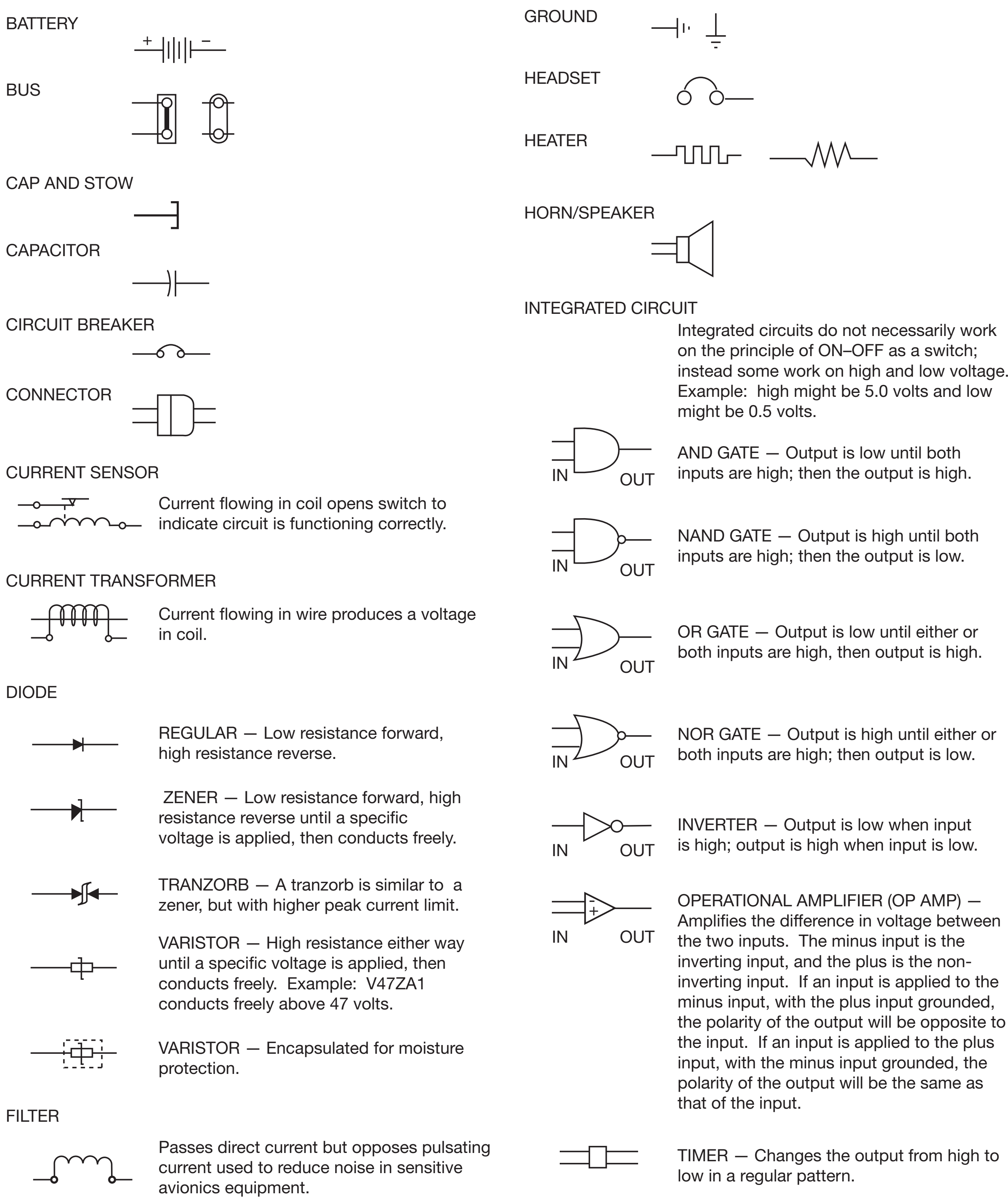
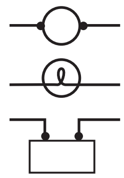
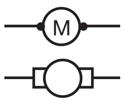


Figure 1-1. Symbology List and Description (Sheet 1 of 4)

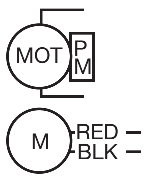
LAMP



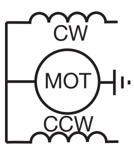
MOTOR



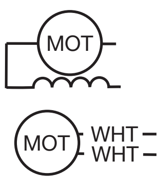
Basic symbol for motor.



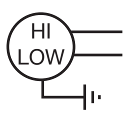
REVERSIBLE MOTOR — Direction of rotation is controlled by reversing power and ground on input wires.



REVERSIBLE MOTOR — Direction of rotation is controlled by applying power to either field winding input wire.



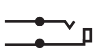
NONREVERSIBLE MOTOR — Direction of rotation is controlled by design; input wires may be connected either way.



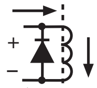
NONREVERSIBLE MOTOR — Two speed controlled by applying power to either input wire.

FUSE/LIMITER

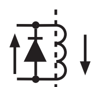
PHONE JACK



The symbol for the solenoid may be a box or a coil; the operation is identical.

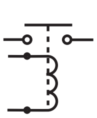


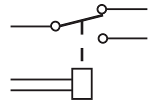
CURRENT FLOW WITH POWER APPLIED

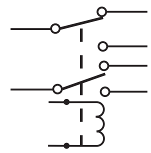


CURRENT FLOW THE INSTANT POWER IS REMOVED

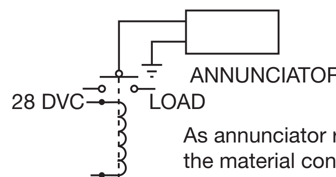
RELAY





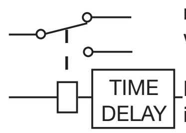


A diode is connected across the input wires of relays and solenoid operated devices such as valves to protect voltage sensitive navigation and electronics equipment. The diode is reverse-biased for normal power and no current flows through the diode. Current flowing through the coil of wire produces a magnetic field to operate the relay or valve. The instant power is removed from the coil, the collapsing magnetic field produces a momentary spike of high voltage which can be several hundred volts depending on the current and the number of turns of wire in the coil. The diode is forward-biased for the power generated in the coil and the high voltage spike is dissipated through the diode. A varistor is used in place of the diode on some relays.



As annunciator relay has a connection on the material contact to indicate by a light or annunciator panel when the relay is energized.

The contacts of a time delay relay do not move to the energized position usually when power is applied.



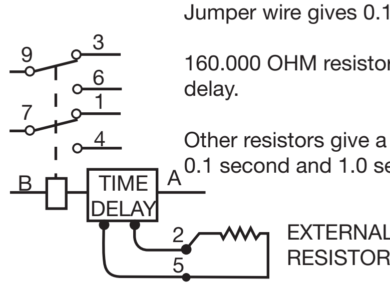
For some time delay relays, the delay time is part of the relay design.

For some time delay relays, the delay time is controlled by the size of an external resistor.

Jumper wire gives 0.1 seconds of delay.

160.000 OHM resistor gives a 10-second delay.

Other resistors give a delay time between 0.1 second and 1.0 second.



RESISTOR

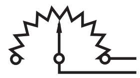
REGULAR — Resistance does not change.

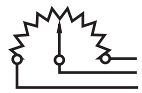
TEMPERATURE CONTROLLED — Resistance change with the temperature.

VARIABLE OR ADJUSTABLE — Resistance changes with mechanical input.

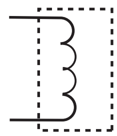
Figure 1-1. Symbology List and Description (Sheet 2 of 4)

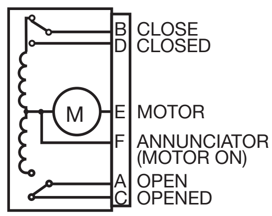
RESISTOR (Cont.)


RHEOSTAT — Type of variable resistor with two wires.

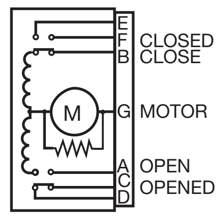

POTENTIOMETER — Type of variable resistor with three wires.

SOLENOID/SOLENOID VALVE


MOTOR OPERATED — Limit switches stop power when limit of travel is reached.



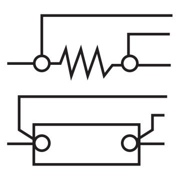
SHOWN OPEN



SHOWN OPEN

MOTOR OPERATED — Limit switches stop power when limit of travel is reached.

SHUNT

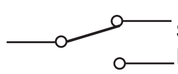


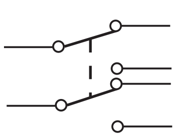
SPLICE

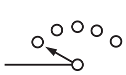

ENVIRONMENTAL

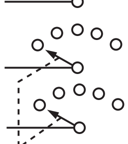
SWITCH

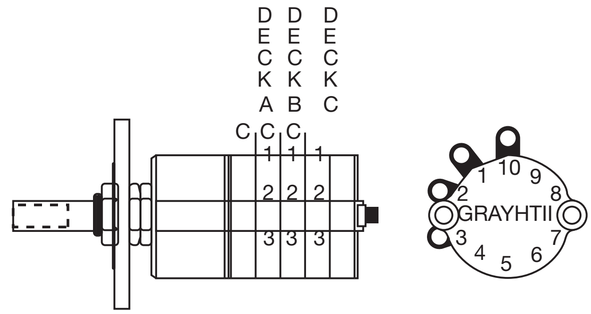

SINGLE-POLE/SINGLE-THROW (SPST)


SINGLE-POLE/DOUBLE-THROW (SPDT) — May have OFF position in the center.


DOUBLE-POLE/DOUBLE-THROW (DPDT) — May have OFF position in the center. Dashed line indicates all parts move simultaneously.


ROTARY OR MULTIPOSITION

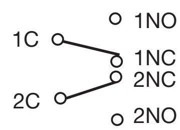

TWO-POLE ROTARY — On rotary or multiple pole switches controlled by a knob, the poles (or decks) are identified on wiring diagrams as A, B, C, with A being the part on the knob or shaft end.



SIDE VIEW
WITH TYPICAL TERMINAL NUMBERING
REAR VIEW

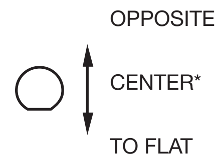
C = Common terminal for each deck
1 – 10 = Switch Terminal Position

EXAMPLE:
AC = Common Terminal of deck A
B1 = Switch position 1 on deck B



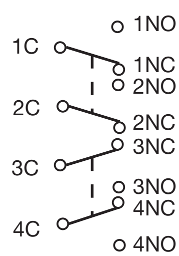
2-POLE

TWO-STAGE — Two-pole, four-pole, or six-pole switch designed so that all moveable switch contacts do not move simultaneously.

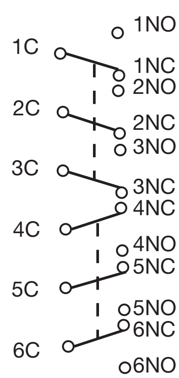


Handle position is reference to flat side of the mounting threads.

*NOTE: Some switches of this type do not have a center position.



4-POLE

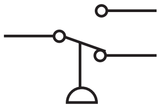


6-POLE

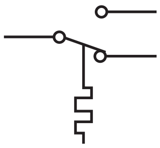
	TO FLAT	CENTER*	OPPOSITE
Contact 1 of 2-Pole Contacts 1 and 2 of 4-Pole Contacts 1, 2, and 3 of 6-Pole	NO C NC	NO C NC	NO C NC
Contact 2 of 2-Pole Contacts 3 and 4 of 4-Pole Contacts 4, 5, and 6 of 6-Pole	NC C NO	NC C NO	NC C NO
C IS THE COMMON TERMINAL NC IS THE NORMALLY CLOSED TERMINAL NO IS THE NORMALLY OPEN TERMINAL			

Figure 1-1. Symbology List and Description (Sheet 3 of 4)

SWITCH (Cont.)

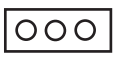


PRESSURE-OPERATED

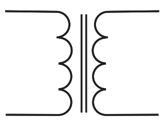


TEMPERATURE-OPERATED

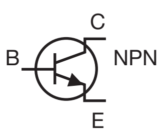
TERMINAL STRIP



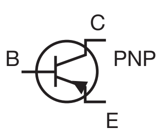
TRANSFORMER

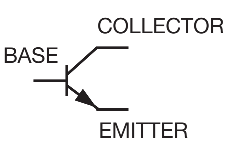


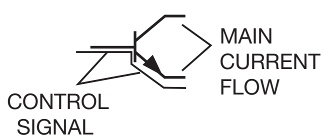
TRANSISTOR

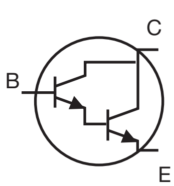


Transistor contacts are identified as base, collector, and emitter. Flow of current through a transistor is controlled by the signal applied to the base. The control current (3 to 5% of total current) flows between base and emitter. The main flow of current (95 to 97%) is between the collector and emitter. Transistors may be drawn without the circle.



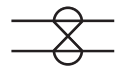




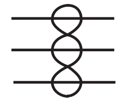


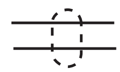
DARLINGTON

WIRE

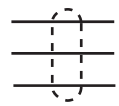


TWISTED





SHIELDED



Q12A20 HIGH-TEMPERATURE WIRE

Figure 1-1. Symbology List and Description (Sheet 4 of 4)

CHARTS

Chapter 91 has connector charts, ground charts, printed circuit board charts, and logic module charts.

Connector charts are provided for those connectors, such as pressure bulkhead connectors, that contain wires for several different systems. All contact pins are shown for the complete connector. The wire number for each wire to a pin is shown with reference to the system where the complete circuit will be found. (Connector maintenance practices in 20-10-04 have insert arrangement charts for connectors.)

Ground charts are provided for all numbered grounds. The grounds are arranged in numerical order with the wire number for each wire and with reference to the system where the complete circuit will be found. The location of each ground is given by zone, description and station, water line, and buttock line.

Printed circuit board or logic module charts show the complete circuit and part number for components on the board. Reference is given to the system where the wires and the complete circuit will be found.

STRUCTURAL REPAIR MANUAL

The *Structural Repair Manual* contains material identification for structure subject to field repair, typical repairs applicable to structural components, information relative to material substitution and fastener installation, and a description of procedures that must be performed with structural repair, such as protective treatment of the repair and sealing.

The manual serves as a medium through which Citation operators are advised of actual repairs of general interest. As service records indicate a requirement, this manual is revised to include additional specific repairs, repairs designed by Cessna, or those designed by the Citation operator.

The *Structural Repair Manual* is prepared in accordance with the Air Transport Association Specification 100 for manufacturers' technical data.

NOTES

SERVICE INFORMATION LETTERS

Technical information that becomes available between revisions to the previously covered publications is announced to operators and maintenance facilities in the field in the form of Service Letters, Service Letter Alerts, Service Bulletins, and Field Notes.

SERVICE LETTER

A "Service Letter" is a technical publication that communicates to those organizations responsible for servicing Cessna/ Citation products the latest up-to-date service information, specific inspection/ maintenance requirements, or parts or product improvements.

Service Letters are written by the Cessna/ Citation Customer Service Department with the knowledge of the Wallace Engineering Department.

SERVICE LETTER ALERT

A "Service Letter Alert" is another form of technical publication that communicates to those organizations responsible for servicing Cessna/Citation products the latest up-to-date service information, specific inspection/maintenance requirements, or parts or product improvements.

The Service Letter Alert is issued on blue paper indicating that a more serious product condition exists and that compliance with instructions listed is essential to continued product safety and reliability.

Service Letter Alerts are written by the Cessna/Citation Customer Service Department with the knowledge of the Engineering Department.

NOTE

Service Letter Alerts on DVD are not designated in color.

NOTES

SERVICE BULLETIN

A "Service Bulletin" is a technical publication that communicates to those organizations responsible for servicing Cessna/ Citation products the latest up-to-date service information, specific inspection/maintenance requirements, and/or parts/product improvements requiring specific part change-out, replacement, or installation.

The Service Bulletin is written and issued by the Cessna/Citation Customer Service Department along with Cessna Aircraft Company, Engineering Department with the concurrence and involvement of the FAA/DER. FAA approval has been obtained on technical data in the Service Bulletin publication that affects airplane type design.

Record of Service Bulletins

A record of service bulletins prepared for the Citation airplane is listed on the Service Bulletin page of the applicable manual.

The list of service bulletins utilizes four columns to summarize service bulletin information:

- Service Bulletin Numbers—The reference data column identifies the service bulletin by number. Service bulletins are numbered consecutively.
- Service Bulletin Date—The issue date column indicates the date the service bulletin displays.
- Title—The title column identifies the service bulletin by nomenclature. It is the same title displayed on page one of the service bulletin.
- Catalog Incorporation Date—The incorporation date column indicates the status of the service bulletin:
    - Date—If a date appears in the column, this indicates the service bulletin information is incorporated into the catalog.
    - No Effect—If "No Effect" appears in the column, this indicates the service bulletin does not affect the catalog.
    - Other self-explanatory statements may appear in this column (i.e., replaces, replaced, superseded, etc.)

NOTES

VENDOR SERVICE BULLETIN OR SERVICE LETTER

"Vendor Service Bulletins" or "Service Letters"are issued as necessary by the vendor when a service condition problem exists on a product used on the Cessna/Citation. At various times, as the condition warrants, the Cessna/Citation Customer Service Department will release a Service Bulletin or Service Letter Alert as the cover page and reference the attached Vendor Service Letter or Service Bulletin to correct a condition on a vendor item affecting the Cessna/Citation product.

FORMAT

The Service Bulletins and Service Letter Alerts are written in the following format:

- Date and type of technical publication used and revision number, if revised.
- Effectivity (unit number affected)
- Reason for issue
- Description
- Compliance:
    - Mandatory—Cessna's statement of expected action normally concerning safety of flight and/or certification items.
    - Recommended—Cessna's statement of expected action for modification or changes normally affecting aircraft performance, utility, or operation.
    - Discretionary—Eligible Citations exhibiting conditions described in this service bulletin may demonstrate improved operation by incorporation of the work described herein. This bulletin is to be accomplished at the discretion of the owner. Eligible aircraft may qualify for parts and labor coverage, as described.
    - Optional—Cessna's statement of expected action for items which may be incorporated at the discretion of the owner/operator.

Although not normally used in the publication of technical data, there are two additional categories that may be used:

- Regulatory—Refers to those items required by the regulating authority having jurisdiction over the aircraft regulatory requirements and that always supersede Cessna's requirements.
- Informational—Refers to those items that provide information general in nature.

NOTE

Component life limits, overhaul and/or repair times, and scheduled maintenance listed in Cessna maintenance manuals, other than Airworthiness Limitation items, are considered "Recommended" unless otherwise stated in the manual or superseded by a regulatory requirement.

- Approval (if FAA/DER approved)
- Manpower requirements (time involved, inspection/modification, and warranty, if any)
- Material (cost and availability)
- Tooling (a reference to any special tools required to complete the Service Letter, Service Letter Alert, or Service Bulletin)
- Change in weight and balance
- Reference
- Other publications affected (e.g., *Maintenance Manual, Illustrated Parts Catalog, Structural Repair Manual*)
- Accomplishment instructions/directions

QUESTIONS

1. Where would control surface balancing information be found?
   A. Chapter 27—“Flight Controls” of the *Maintenance Manual*
   B. *Structural Repair Manual*
   C. Chapter 51—“Structures” of the *Maintenance Manual*
   D. Chapter 12—“Servicing” of the *Maintenance Manual*

2. Which chapter contains access panel locations?
   A. Chapter 20 “Standard Practices” of the *Maintenance Manual*
   B. Chapter 6 “Dimensions and Areas” of the *Maintenance Manual*
   C. Chapter 5 “Time Limits/ Maintenance Checks” of the *Maintenance Manual*
   D. Chapter 51—“Structures” of the *Maintenance Manual*

3. Information for proper lubrication of the engine thrust reversers is found in:
   A. Chapter 78 “Exhaust” of the *Maintenance Manual*
   B. Chapter 20 “Standard Practices” of the *Maintenance Manual*
   C. Chapter 12—“Servicing” of the *Maintenance Manual*
   D. Chapter 5 “Time Limits/ Maintenance Checks” of the *Maintenance Manual*

4. Service Bulletins are:
   A. Always mandatory
   B. Written by Cessna and approved by the FAA
   C. Written by committee of service center personnel
   D. Written by the FAA and approved by Cessna

5. The (*) symbol on some connector pin letters, on a wire schematic, indicates:
   A. A capped wire to that pin
   B. Large case letters
   C. Small case letters
   D. A note on the schematic for that pin

CHAPTER 5-12
AIRCRAFT GENERAL

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 5
AIRCRAFT GENERAL

INTRODUCTION

This chapter includes illustrations and statistical information concerning the Citation XL/XLS/XLS+ aircraft. Provided are overall aircraft dimensions, station locations, aircraft zoning, location of major structural members and components. Information is also provided concerning ground handling, servicing information and inspection requirements.

GENERAL

Information is provided on airworthiness and limitations, time limits and checks, continuous inspection program, dimensions, areas, locations and zoning. Information is also provided concerning access panels and plates, jacking practices, leveling and weighing, towing, taxiing and parking. A section on servicing and replenishing is provided for components,valves, filters, fuel, batteries, pneumatics, hydraulics, lubricants , cleaning materials, and deicing fluids.

TIME LIMITS/ MAINTENANCE CHECKS

DESCRIPTION

This chapter provides the time limits and maintenance checks for the Citation XL/XLS/XLS+ aircraft. It is divided into sections, each with a specific purpose toward providing information necessary to establish inspection criteria:

- Inspections—This section lists (in chart format) all inspection and servicing requirements which must be performed.
- Unscheduled Maintenance Checks—This section includes inspections and checks which are required due to special or unusual circumstances that do not have regular repetitive intervals for accomplishment.

Federal Aviation Regulations Part 91.409(e) defines the inspection requirements for turbojet multi-engine aircraft. The inspection requirements defined in this chapter are the manufacturer recommended procedures and are tailored to satisfy the requirements of Part 91.409(e) and (f)(3).

For aircraft registered in countries other than the United States, the procedures specified by the Airworthiness Authority of that country must be followed.

All nondestructive testing procedures required in this chapter are defined in the *Nondestructive Testing Manual*. These procedures must be performed by personnel and at facilities that are certified by Cessna Aircraft Company. For details of certification program, refer to the *Nondestructive Testing Manual*.

Maintenance Steering Group—3 (MSG-3) Scheduled maintenance development scheduled inspection program is for Model 560XL aircraft 5717 and subsequent and Model 560XL aircraft 5002 through 5718 incorporating SB560XL-05-01.

The Cessna MSG-3 group used the concepts of the air transport authority (ATA) MSG-3 to make the time limits and maintenance checks for the Model 560XL aircraft.

An analysis of the aircraft maintenance significant items (MSI) and structural significant items (SSI) was made by the Cessna MSG-3 group. Each of these MSI were reviewed by working group(s) that had specialist representatives of operators, Cessna staff, and the regulatory authority.

After each MSI was approved by the working group, it was then given to an industry steering committee (ISC). The ISC made sure that the MSG-3 process identified all of the MSI and SSI and whether or not a task was made from the analysis.

The initial scheduled maintenance tasks and intervals have been specified in a maintenance review board (MRB) report completed by the Cessna MSG-3 group.

NOTES

Refer to the Citation 560XL/XLS (CE-560XL) maintenance review board report (MRBR) for additional information about the Model 560XL MSG-3 program. The MRBR also has information about MSG-3 concepts and methods that maintenance personnel must understand.

Inspection requirements for reduced vertical separation minimum (RVSM) certified aircraft are included in the regular continuous inspection program. The requirements include those inspections required by FAR 91.411 as defined for the specific air data computer installed, and verification that the static ports are within tolerances. Refer to Chapter 34—"Pitot/Static System" in the *Airplane Maintenance Manual (AMM)*.

The special inspection requirement only occurs if the aircraft has been damaged and repaired/painted or polished/buffed in the static port area. The static ports must be inspected to ensure they are within tolerances. Refer to Chapter 34—"Pitot/Static System" in the *AMM*.

As detailed in Part 91.409, paragraph (e), of the Federal Aviation Regulations, turbo jet aircraft must be inspected in accordance with an authorized inspection schedule. This section presents the basis for a continuous inspection program for the Citation XL/XLS/XLS+, recommended by Cessna Aircraft Company.

An operator may elect to perform the recommended inspections on a schedule other than that specified. Any inspection schedule requiring the various inspection items detailed in this chapter to be performed at a frequency equal to that specified herein or more frequently is acceptable. Any inspection item performed at a time period in excess of that specified herein must be approved by the appropriate regulating agency.

Inspection requirements for Supplemental Type Certification (STC) installations are not included in this manual. When an STC installation is incorporated on the aircraft, those portions of the aircraft affected by the installation must be inspected in accordance with the inspection program published by the owner of the STC. Since STC installations may change:

- Systems interface
- Operating characteristics
- Component loads or
- Stresses on adjacent structures

Cessna provided inspection criteria may not be valid for aircraft with STC installations.

NOTES

Table 5-1. INTERVAL AND PHASE CROSS-REFERENCE FOR INSPECTION TIME LIMITS

A	Phase 6	Every 14 days for airplanes based and operated in a corrosive environment (coastal areas).
B	Phase 7	(Not currently used) Every 50 hours.
D	Phase B	Every 150 hours.
E	Phase 1,2, 3, or 4	Every 300 hours or 24 calendar months, whichever occurs first.
F	Phase 5	Every 1200 hours or 36 months, whichever occurs first.
G	Phase 8	Every 6 calendar months.
H	Phase 9	Every 900 hours, ± 50 hours, or 24 calendar months, whichever occurs first.
I	Phase 28	Every 100 hours.
J	Phase 30	Every 600 hours.
K	Phase 18	Every on year.
L	Phase 19	(Not currently used) Two years from data of manufacture and every two years thereafter.
M	Phase 20	Every 24 calendar months as required by 14 CFR 91.411, 91.413 and RVSM certification. (No grace period.)
N	Phase 21	Every 2 years.
O	Phase 22	Every 3 years.
P	Phase 23	Every 5 years.
Q	Phase 48	Every 6 years.
R	Phase 47	Every 12 years.
S	Phase 11	(Not currently used) Every 3 to 6 calendar months.
T	Phase 51	Every 2400 hours.
U	Phase 52	Every 600 hours or 12 calendar months whichever occurs first and every 200 hours or 3 calendar months whichever occurs first, thereafter.
V	Phase 53	Every 900 hours or 12 calendar months, whichever occurs first.
W	Phase 49	First 6 years or 2400 hours, whichever occurs first, and every 1200 hours or 36 calendar months, whichever occurs first, thereafter.
X	Phase 54	First 10 years from date of manufacture and every 5 years therafter (based upon date of previous hydrostatic test).
Y	Phase 55	(Not currently used) First 6000 hours or 10 years, whichever occurs first, and every 2400 hours or 10 years, whichever occurs first thereafter.
Z	Phase 56	(Not currently used) Every 150 hours (no grace peroid).
AA	Phase 57	(Not currently used) First 150 hours and every 150 hours or 24 calendar months thereafter, whichever occurs first (no grace period)
AB	Phase 58	Every 500 operating hours of the auxiliary power unit (APU) starter-generator. Phase 58 is to be performed prior to the APU starter-generator overhaul as required in Chapter 5-11-00.
AC	Phase 59	Every 600 hours or 12 calendar months whichever occurs first.
AD	Phase 60	Inspect initally at 2400 hours and then every 1200 hours thereafter.
AE	Phase 61	Inspect every 1200 hours.

Table 5-1. INTERVAL AND PHASE CROSS-REFERENCE FOR INSPECTION TIME LIMITS (Cont)

AF	Phase 62	Thrust Reverser Lubrication. Refer to *Nordam Group Thrust Reverser Component Maintenance Manual* for the lubication intervals and procedures.
AG	Phase 63	Every 12 calendar months as required by 14 CFT 91.207. (No grace period).
AH	Phase 64	Vapor Cycle Cooling System Inspection. Refer to the *Enviro Systems, Inc. General Operating, Servicing and Maintenance Manual* for Airborne R-134a Air Conditioning Systems for inspection intervals and procedures.
MA	Phase MA	Every 500 hours. (This inspection is a Chapter 4 requirement, and as such, the interval limitation CANNOT be exceeded.)
MC	Phase MC	First 180 hours and every 180 hours or 24 calendar months therafter, whichever ocurs first. (This inspection is a Chapter 4 requirement, and as such,m the interval limitation CANNOT be exceeded.)
MD	Phase MD	First 6000 hours or 10 years, whichever occurs first, and every 2400 hours or 10 years, whichever occurs first thereafter. (This inspection is a Chapter 4 requirement, and as such, the interval limitation CANNOT be exceeded.)

Table 5-2. INSPECTION INTERVAL

INSPECTION INTERVAL	INSPECTION DOCUMENT
MSG-3 interval 1A/1C item(s), which are completed every 600 hours or 12 calendar months whichever occurs first.	5-92-01
MSG-3 interval 1C item(s), which are completed every 12 calendar months.	5-92-02
MSG-3 interval 2A/2C item(s), which are completed every 1,200 hours or 24 calendar months whichever occurs first.	5-92-03
MSG-3 interval 2C item(s), which are completed every 24 calendar months.	5-92-04
MSG-3 interval 2A/4C item(s), which are completed every 1,200 hours or 48 calendar months whichever occurs first.	5-92-05
MSG-3 interval 3A/3C item(s), which are completed every 1,800 hours or 36 calendar months whichever occurs first.	5-92-06
MSG-3 interval 3C item(s), which are completed every 36 calendar months.	5-92-07
MSG-3 interval 4A/4C item(s), which are completed every 2,400 hours or 48 calendar months whichever occurs first.	5-92-08
MSG-3 interval 4A/4C, 2A/4C thereafter item(s), which are completed at the first 2,400 hours or 48 calendar months whichever occurs first, then every 1,200 hours or 48 calendar months whichever occurs first thereafter.	5-92-09
MSG-3 interval 4C item(s), which are completed every 48 calendar months.	5-92-10
MSG-3 interval 4C, 2C thereafter item(s), which are completed at the first 48 calendar months, then every 24 calendar months thereafter.	5-92-11
MSG-3 interval 5A item(s), which are completed every 3,000 hours.	5-92-12
MSG-3 interval 6C item(s), which are completed every 72 calendar months.	5-92-13
MSG-3 interval 8C item(s), which are completed every 96 calendar months.	5-92-14
MSG-3 interval 12C item(s), which are completed every 144 calendar months.	5-92-15
MSG-3 interval 400 hours/1C item(s), which are completed every 400 hours or 12 calendar months whichever occurs first.	5-92-16
MSG-3 interval 500 APU hours/2C item(s), which are completed every 500 APU hours or 24 calendar months whichever occurs first.	5-92-17
MSG-3 interval 1,000 hours/2C item(s), which are completed every 1,000 hours or 24 calendar months whichever occurs first. Hours are based on APU operating hours.	5-92-18
MSG-3 interval 4A/2C item(s), which are completed every 2,400 hours or 24 calendar months whichever occurs first.	5-92-19

INSPECTIONS

General Inspection Criteria

While doing each of the specific inspections listed in this chapter, additional general inspections of surrounding areas must be performed while access is available. These general inspections are intended to detect obvious conditions which warrant further action.

When an area is exposed, wire bundles must be examined for chafing, proper security and support. Make sure that wire bundles are not attached to hydraulic tubes or lines.

Inspection items listed are for specific components and systems, however, the entire inspection program requires a high degree of professionalism and judgment by inspection personnel. This ensures that all components and systems are maintained and examined to the highest safety standards.

If a component or system is disturbed (due to maintenance) after an initially required operational or functional test is completed, then that specific test must be confirmed again after the completion of any maintenance, before returning the system or component to service. Refer to the appropriate chapter in this manual for removal, installation, operational tests and functional tests of components and/or systems.

Some items or components require lubrication. Refer to Chapter 12—"Scheduled Servicing" in the *AMM*, for lubricant, lubricating points and method.

Refer to Chapter 6—"Aircraft Zoning" in the *AMM*, for aircraft zone definition.

Do a preflight inspection after completion of the applicable inspection to make sure all required items are properly serviced. Refer to the approved *Airplane Flight Manual (AFM)*.

For record-keeping purposes, inspection items that are also required (Chapter 4—"Airworthiness Limitations" in the *AMM*) have been placed in intervals beginning with M. The interval for these items is defined by Chapter 4 and must be accomplished at or before the listed interval. Chapter 4 items do not have a grace period.

NOTE

Inspection requirements for the engine, vapor cycle cooling system, wheels and brakes are defined by component manufacturer. To ensure that the latest inspection requirements are performed as defined by manufacturer, refer to the requirements at intervals published in those inspection documents (Tables 5-1 and 5-2) and perform them as listed.

NOTES

Table 5-3. FUNCTION NUMBER IDENTIFICATION

10	CLEANING	64	Lubricating
11	Chemical	65	Fueling, Defueling
12	Abrasive	67	Disinfect, Sanitize
13	Ultrasonic	68	Drain Fluid
14	Mechanical	70	TESTING, CHECKING
15	Stripping	71	Operational
16	Miscellaneous Cleaning	72	Functional
17	Flushing	73	System
20	INSPECTION CHECK	74	Bite
21	General Visual	75	Special
22	Detailed Dimensional	76	Electrical
23	Penetrant	78	Pressure
24	Magentic	79	Leak
25	Eddy Current	80	MISCELLANEOUS
26	X-Ray	81	Fault Isolation
27	Ultrasonic	82	Adjusting, aligning, calibration, rigging
28	Specific, Special	87	Bleeding
29	Borescope	90	CHANGE, REMOVE, INSTALL
60	SERVICING, PRESERVING, LUBRICATION	96	Replace
61	Servicing		

Tasks

Inspection items are identified as having tasks associated with them, which are more complex than those described in *Inspection Time Limits*. These tasks are identified with an Air Transport Association (ATA) chapter–section–subsection–function number. Each task has a unique number. The chapter- section-subsection number identifies the specific location of the procedure in the *AMM*. The last three digits correspond to the task's specific functions.

The example illustrates a sample task number. Beginning with the word "task," the numbers that follow indicate where the inspection task text can be found by indicating the chapter–section–subsection. The two-digit function number indicates the general classification of the task. (See function number identification in Table 5-2). The last digit indicates the sequence number. If the same chapter–section–subsection and function is used for a different task, the sequence number increments change, by one.

CD-ROM users can link from a highlighted task number to the task text. Task numbers may be found in the Inspection Time Limits table or at any point throughout this manual where a task number is referenced.

NOTE

The third digit of function is assigned sequentially, beginning with zero. This ensures a unique nine-digit task number.

Supplemental Type Certificate Installations

Inspection requirements for supplemental type certificate (STC) installations are not included in this manual. When an STC installation is incorporated on the aircraft, those portions of the aircraft affected by the installation must be inspected in accordance with the inspection program (published by the owner of the STC). Since STC installations could change systems interface, operating characteristics and component loads or stresses on adjacent structures, Cessna-provided inspection criteria may not be valid for aircraft with STC installations.

Continuous Airworthiness Inspection Program

Turbojet aircraft must be inspected in accordance with an authorized inspection schedule. This section contains the continuous inspection program for the Citation XL/XLS/XLS+ recommended by Cessna Aircraft Company.

For aircraft registered in countries other than the United States, the procedures specified by the airworthiness authority of that country must be followed.

NOTES

Title 14 of the Code of Federal Regulations

General Operating and Flight Rules (Part 91.409)

No person may operate:

- A large aircraft
- A turbojet multi-engine aircraft
- Turbo-propeller powered multi-engine aircraft
- Or a turbine-powered rotorcraft

Unless the replacement times for life-limited parts specified in the aircraft specifications, type data sheets, or other documents approved by the administrator are complied with and the aircraft, or turbine-powered rotorcraft including the following, is inspected in accordance with an inspection program selected under the provisions of paragraph 91.409(f) of this section:

- The airframe
- Engines
- Propellers
- Rotors
- Appliances
- Survival equipment
- Emergency equipment

However, the owner/operator of a turbine-powered rotorcraft may elect to use the provisions of paragraph 91.409(a), (b), (c) or (d) in lieu of an inspection option of paragraph (f).

The registered owner or operator of each aircraft or turbine powered rotorcraft described in paragraph 91.409(e) of this section must select, identify (in the aircraft maintenance records), and use one of the following programs for the inspection of the aircraft:

- A continuous airworthiness inspection program that is a part of a continuous airworthiness maintenance program, which is currently in use by an individual holding an air carrier operating certificate that is issued under Part 121, 127, or 135 of this chapter that make and model aircraft must be operated under Part 121 of this chapter and maintained it under paragraph 135.411 (2) of this chapter.
- An approved aircraft inspection program approved under paragraph 135.419 of this chapter, currently in use by an individual who holds an operating certificate under Part 135 of this chapter.
- A current inspection program recommended by the manufacturer.
- Any other inspection program established by the registered owner/operator of that aircraft (or turbine-powered rotorcraft), approved by the administrator under paragraph (g) of this section. However, the administrator may require revision to this inspection program in accordance with the provisions of Part 91.415.

Each operator shall include (in the selected program) the name and address of the person responsible for scheduling the inspections required by the program. A copy of that program must be made available to the person performing inspections on the aircraft, and upon request to the administrator.

Each operator of an aircraft (or turbine-powered rotorcraft) desiring to establish or change an approved inspection program under paragraph (f)(4) of this section must submit the program for approval to the local FAA Flight Standards District Office (FSDO) that has jurisdiction over the area in which the aircraft is based. The program must include the following information:

- Instructions and procedures for the conduct of inspections for the particular make and model aircraft, including necessary tests and checks. The instructions and procedures must set forth in detail:

  - The parts and areas of the airframe
  - Engines
  - Propellers
  - Rotors and appliances
  - Emergency equipment that requires inspection
- A schedule for performing the inspections that must be performed under the program expressed in terms of the time in service, calendar time, number of system operations, or any combination of these.
- When an operator changes from one inspection program (under paragraph (f) of this section) to another, following must be applied to determine inspection times under the new program:
  - The time in service
  - Calendar times
  - Or cycles of operation accumulate dunder the previous program

Inspection and Maintenance Schedule

The recommended continuous inspection and maintenance schedule for Citation XL/XLS/XLS+ aircraft follows.

The program is divided into five primary phases (Phases 1 thru 5). These are the main repetitive phases that make up the basic requirements of the inspection program. The remaining phases include all of the remaining inspection items.

NOTE

Phase B, also provided, must be accomplished at 150-hour intervals when Phases 1 through 4 are combined at 300-hour intervals (Method 3 of accomplishing Phases 1 thru 4).

The inspection program is divided into phases that better enable continuous type inspection. Recommended continuous type inspection may be accomplished by one of three methods.

The remaining sections in the 5–12–XX series contain signoff sheets which are listings of inspection items in zone order. The last two digits (XX) of the chapter–section–subject correspond to the inspection phase.

When the continuous inspection program is selected, additional inspections shall be complied with as follows to ensure a complete inspection:

- Continuous—All phases shall be performed. Tasks are defined within the individual chapters.
- Component Time Limits—Refer to 5–11–00. Components that require maintenance at a specified time (not included on condition items).
- Unscheduled Maintenance—Refer to 5–50–00.

  This includes the following:
  - Hard or overweight landing check
  - Overspeed check
  - Severe turbulence and/or maneuver checks
  - Inadvertent in-flight thrust reverser deployment
  - Lightning strike check
  - Foreign object damage check
  - Towing with large fuel unbalance or high drag/side-loads due to ground handling check
  - Aircraft operation through deep standing water
  - Nose landing gear tow limits check

Table 5-4. METHOD 1

	75-HOUR	150-HOUR	225-HOUR	300-HOUR
PHASE 1	X			
PHASE 2		X		
PHASE 3			X	
PHASE 4				X

Table 5-5. METHOD 2

	75-HOUR	150-HOUR	225-HOUR	300-HOUR
PHASE 1		X		
PHASE 2		X		
PHASE 3				X
PHASE 4				X

Table 5-6. METHOD 3

	75-HOUR	150-HOUR	225-HOUR	300-HOUR
PHASE B		X		
PHASE 1				X
PHASE 2				X
PHASE 3				X
PHASE 4				X

Continuous Inspection Program Procedure

The five primary phases (Phases 1 thru 5) are the core of the continuous inspection program. The remaining phases have higher time inspection requirements and are not included in the Phase 1 through 5 continuous inspection cycle. There are three methods: Method 1, 2 and 3 (Tables 5-3 thru 5-5), which combine the different methods to complete Phase 1 through 5 in a continuous inspection program. Each plan provides different inspection intervals to adjust for the requirements of individual operators. Refer to paragraph 4.G. for the continuous inspection program chart that gives which phases are due at a specific hourly interval for each plan.

The five primary phases (Phases 1 through 5) constitute the core of the continuous inspection program.

- Phases 1 through 4 each have an interval of 300 hours. These phases must be accomplished in a continuous, repetitive 300-hour cycle. Three different optional methods can be used (as detailed in Item G).
- Phase 5 has an interval of 1200 hours. Phase 5 must be accomplished in a continuous, repetitive 1200-hour cycle.

The remaining phases can be accomplished when due or earlier to coincide with a convenient inspection or during maintenance downtime.

Calendar time limits (i.e., 36 months in Phase 5) must be taken into consideration for aircraft that have accumulated fewer hours than specified in the calendar period.

"Component Time Limits" (5–11–00) lists all components which must be replaced or overhauled on a scheduled basis. Those items are underlined and listed in Chapter 4—"Airworthiness Limitations." Replacement or overhaul of listed items must be accomplished when due or earlier.

An operator may elect to perform the recommended inspections on a schedule other than specified. Any inspection schedule requiring the various inspection items detailed in this chapter must be performed at a frequency equal to that specified here, or more frequently. Any inspection item performed at a time period in excess of that specific herein must be approved by the appropriate regulating agency.

Three optional methods of accomplishing Phases 1 through 4 are provided as follows:

NOTE

Operators changing from one method to another in performing Phases 1 through 4 must ensure that the timeframe from one inspection to the next inspection (on any given item) does not exceed intervals indicated in this manual.

METHOD 1

Phase 1 through Phase 4 inspections are based on 300-hour cycles, with one of the phase inspections accomplished every 75 hours of aircraft operation. Applicable additional phases are integrated at due times with the first four phases. At the completion of Phase 4, Phase 1 is due 75 hours later and the cycle is repeated.

METHOD 2

Phase 1 and Phase 2 are combined and accomplished at alternate 150-hour intervals, with Phase 3 and Phase 4 being performed at the next 150-hour interval. Applicable additional phases are integrated at due times with the first four phases. At the completion of Phases 3 and 4, Phases 1 and 2 are due 150 hours later and the cycle is repeated.

METHOD 3

Phase 1 through Phase 4 are all combined and accomplished at 300-hour intervals. Applicable additional phases are integrated at due times with the first four phases. Those that cannot be so integrated must be performed early or separately.

NOTE

When all of the phases are combined into 300-hour intervals, an inspection is required on certain items every 150 hours. These inspections are listed in Phase B. The 150-hour inspections are listed in two of the phases. Duplicated items only need to be accomplished once when combining phases.

Phase 5 is accomplished all at once, every 1200 hours concurrently with completion at the end of the fourth cycle of Phases 1 through 4.

Hour Interval Items

- Upon completion of Phases 1 through 4, a continued program repeats the cycle. When the 1200-hour interval is reached, Phase 5 is performed.
- Special inspections (Phases 6 and on) with hour intervals are performed with a phase that corresponds to that particular hour interval.

Time limit items components and/or systems with time limits of 12 months, 24 months, 36 months, 48 months, 60 months, etc., are performed at the specific time limit or with a phase inspection corresponding to the interval.

Inspection Time Limitations

NOTE

Any inspection time limitation required in both Chapter 5 and Chapter 4 must not be extended. Any inspection required by the Code of Federal Regulations (CFR) and reduced vertical separation minimums (RVSM) must not be extended.

Phases 1, 2, 3, 4 and 5 due points can be extended for maintenance scheduling purposes only as provided below:

- Phases 1, 2, 3 and 4 can be extended up to a maximum of 30 flight hours or two calendar months beyond the due point.
- Phases 5, 49, 51, 60, and 61 can be extended up to a maximum of 100 flight hours or two calendar months beyond the due point.
- The Phase B, when used, can be extended up to a maximum of 30 flight hours or two calendar months beyond the due point.
- Phase 59 can be extended up to a maximum of 50 flight hours beyond the due point.

All remaining phase due points can be extended for maintenance scheduling purposes only up to a maximum of ten flight hours from the due point (for hourly driven inspections and one calendar month for calendar driven inspections).

Any portion of the allowable extension used does not need to be deducted from the subsequent due point.

Any inspection program based upon the intervals of items in this chapter (or more frequent intervals) is acceptable.

Program Startup

NOTE

This procedure for program startup applies only to aircraft that have previously used an inspection program other than the Cessna recommended Continuous Inspection Program.

The following steps must be completed for aircraft which are not newly manufactured, in order to begin the Continuous Inspection Program:

1. Conduct a complete aircraft inspection by performing Phases 1 through 5.
2. Start the program at “Check Number 1.” Refer to the *Continuous Inspection Program Procedure*.
3. Continue performing inspections in the normal manner.

Logbook Forms, Pilot Information

The aircraft flight log (Tables 5-6 and 5-7) provides a convenient place to maintain:

- Individual and cumulative flight records
- Total hours on the aircraft and each engine
- Total landings for the aircraft and
- The cycles of each engine

The original of the Flight Log form is placed in Section 1 of the FAA approved *Aircraft Flight and Maintenance Logbook* and kept as part of the permanent aircraft log.

Form Completion

Complete the basic information at the top of the form. Enter the number of the page as it sequentially falls in the logbook section beginning with one. The line entitled "Accumulated Totals Brought Forward" is the record information carried forward from the previous page. This is a journey log. One line represents one flight.

Record the appropriate information under each numbered column:

1. The date of each flight
2. City where flight originated

   City of flight destination
3. The number of persons aboard, including crew
4. Distance of flight in nautical miles
5. The flight duration in hours and tenths from flight hour meter

Table 5-7. CITATION XL/XLS/XLS+ LOGBOOK

SECTION 1	Flight Time Information	SECTION 6	Left Engine Maintenance
SECTION 2	Pilot Squawk on Last Flight	SECTION 7	Left Engine Service Bulletin
SECTION 3	General Airplane Maintenance	SECTION 8	Right Engine Maintenance
SECTION 4	Service Bulletin	SECTION 9	Right Engine Service Bulletin
SECTION 5	Engine Change	SECTION 10	Service Condition Reports
		SECTION 11	Component Master Inventory Cross-Reference List

Table 5-8. NEW PAGES FOR THE AIRPLANE LOGBOOK

Airplane	SECTION 1	Airplane Flight Log
Airplane	SECTION 2	Airplane Discrepancies
Airplane	SECTION 3	Maintenance Transaction Report
Airplane	SECTION 4	Maintenance Transcation Report
Airplane	SECTION 5	Engine Change Record
Left Engine	SECTION 6	Maintenance Transcation Report
Left Engine	SECTION 7	Maintenance Transcation Report
Right Engine	SECTION 8	Maintenance Transcation Report
Right Engine	SECTION 9	Maintenance Transcation Report
Airplane	SECTION 10	Service Condition Report
	SECTION 11	Component Master Inventory Cross-Reference List

6. Accumulated total aircraft time in hours and tenths
7. Accumulated total time for engine one in hours and tenths
8. Accumulated total time for engine two in hours and tenths
9. Number of landings this flight
10. Accumulated total aircraft landing
11. Accumulated engine one cycles
12. Accumulated engine two cycle
13. The discrepancy sheet number column provides a reference to the *Aircraft Discrepancy Form*. The *Aircraft Discrepancy Form* details aircraft discrepancies on particular flight. If there are no discrepancies, enter NONE in this column. This provides the pilot with a convenient method of determining whether previous aircraft discrepancies have been corrected.
14. Record the last name of the pilot and copilot for each flight in the last column.
15. Fill in the line "Accumulated Totals" when the form is completely filled out or at the end of the CESCOM reporting period.
16. Carry the totals of these lines forward to the line entitled "Accumulated Totals Brought Forward" on the next aircraft flight log form.
17. Place the original white sheet in the *Aircraft Flight and Maintenance Logbook* in Section 1.
18. Mail the first copy (pink sheet) to the address at the top of the form. It is used by CESCOM to report aircraft utilization.
19. The second copy (yellow sheet) provides an extra customer copy and must be filed at the aircraft base.

Mechanic Information

1 Mechanics fill out the aircraft and engine maintenance records and use "*Maintenance Transaction Report*" for additional pages in the aircraft logbook in Sections 3, 4, 6, 7, 8 and 9.

2 Place copies of the inspection program selected (sample letter), inspection record of phase due and completed in the front of the aircraft log, for pilot and mechanic; information on inspections coming due.

Maintenance Transaction Report

Fill out the *Maintenance Transaction Report* as follows:

1. Maintenance log—Aircraft or engine entry.
2. Date—Write the month, day and year that maintenance is performed.
3. List the aircraft serial number and aircraft registration number.
4. Log the total aircraft hours.
5. Complete when applicable.
6. Write the three-letter identifier for the city in which step the maintenance transaction is performed.
   a. Item Name—The component for which maintenance is performed.
   b. Position—Left or right, inboard or outboard position.
   c List the part number.
   d. Serial number of the part installed.
   e. Removal Reason—Check the box which corresponds to the reason for component removal.
   f. Status of Installed Part—Check the status of the part installed. Page 3 (Yellow) should go to Aircraft base files.

g. Time between overhaul (TBO)—Check one of the boxes under check one. Next, list the TBO change in the blank and check one of the TBO controls. If the "OK as is" box is checked, nothing else needs to be done in this box.

h. Time since overhaul (TSO)—If an overhaul component is installed, list its present TSO in the block and check the type of control.

i. List section and page numbers.

j. Type of maintenance.

k. For information and other log entries—Enter pages in the logbook. Place page one (white) in the logbook. Place page two (pink) in CESCOM.

UNSCHEDULED MAINTENANCE CHECKS

During operation, forces can be applied to the aircraft that make it necessary to complete unscheduled maintenance. Here are some examples of these forces:

- Hard/overweight landings
- Overspeed—Speed greater than the placard speeds (of the flaps) or landing gear speed that is more than aircraft design speeds
- Dangerous air turbulence or dangerous maneuvers
- Accidental in-flight thrust reverser deployment
- Lightning strike
- Foreign object damage
- Aircraft towed with a large fuel unbalance or high drag/side loads due to ground handling
- Aircraft operation through deep standing water
- Nose landing gear tow limits

When any of these conditions is listed on a report given by the flight crew, complete a visual inspection of the airframe; and perform specific inspections of components or areas that might be affected.

The inspections are done in order to complete an analysis of the depth of damage:

- In local areas where damage can be seen
- To the structure and components adjacent to the area of damage

When there is a lightning strike, a full inspection of the aircraft exterior must be completed to discover possible damage.

If foreign object damage might have occurred, do a visual inspection of the aircraft before the aircraft is returned to service.

Unscheduled Maintenance Checks Defined

Hard landing—Any landing made by an aircraft at a sink rate greater than what is permitted.

Overweight landing—Landing the aircraft at any gross weight which is greater than the placard landing weights.

NOTE

If a hard/overweight landing is mixed with high drag/side loads, additional checks are required.

Overspeed—Any time an aircraft has done one or both of the conditions that follow:

- Aircraft speed is greater than the placard speed limits of the flaps and/or landing gear.
- Aircraft speed is greater than the aircraft's design limitations.

B4224

LIGHTNING STRIKE/STATIC DISCHARGE INCIDENT REPORTING FORM
Part 1

1. Flight Crew must complete Part 1.

NOTE: Entire report must be filled out following any lightning strike incident. If lightning strike is discovered after the fact, complete as much of report as possible. File form immediately following incident. Attach additional sheet(s) to provide complete description.

A. Flight Information:
Flight Number ________ Strike Date ________ Model ________ Unit/Serial Number __________
Altitude ________ ft Airspeed ________ knots Geographical Location ____________________

B. Airplane Orientation:
Takeoff ________ Climb __________ Cruise __________ Descent __________
Approach ________ Other ____________

C. At time of Strike, aircraft was:
Above Clouds _______ Within Clouds ________ Below Ceiling _________

D. Precipitation at Strike:
Rain _________ Sleet __________ Hail __________ Snow __________ None _______

E. Lightning in Vicinity:
Before ________ After _________ None _______

F. Static in Comm/Nav
Before ________ After _________ None _______

G. Was St. Elmo's fire (bluish electrical discharge or corona) visible before strike?
Yes ______ No ______

H. Interference (**I**) or Outage (**O**) report. Check all the following which apply, and list affected systems, such as dimming of cabin lights, total system outage, etc.

Engines	I ________________	O ________________
Navigation	I ________________	O ________________
Communication	I ________________	O ________________
Flight Instruments	I ________________	O ________________
Flight Control	I ________________	O ________________
AC Power System	I ________________	O ________________
DC Power System	I ________________	O ________________

I. Additional comments and descriptions:

Part 1 completed by: ________________________________ Date __________ Phone __________

Figure 5-1. Lightning Strike Reporting Form (Sheet 1 of 2)

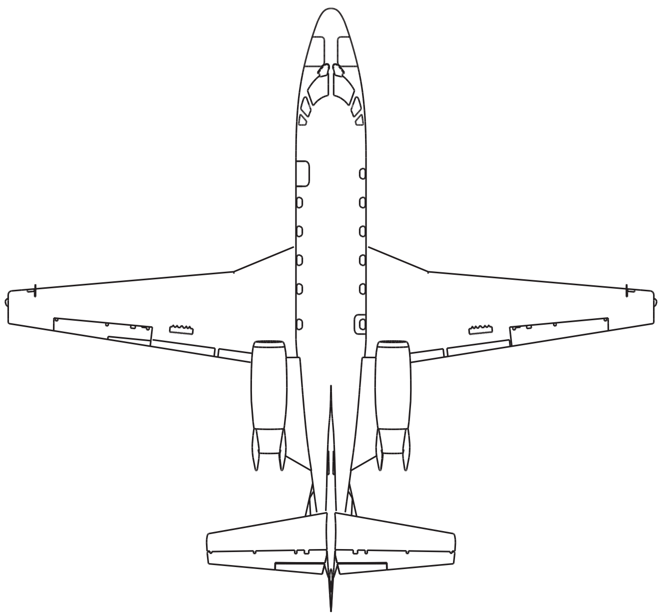
LIGHTNING STRIKE/STATIC DISCHARGE INCIDENT REPORTING FORM
Part 2

1. Ground Crew must complete Part 2.

NOTE: Attach additional sheet(s) to provide complete description. Photos and sketches of damage are recommended and must be itemized and referenced in their description.

NOTE: If damage is severe, please report the lightning strike as soon as possible. Inspection by Cessna Engineering Representative(s) may be required.

A. List any sweeping points, such as burn marks, divots, etc., and skin penetrations on airplane skin believed to be the result of the lightning strike. Itemize and reference location(s) of damage on drawing provided. Indicate top, bottom, left or right.



B. Describe damage to structure and external components caused by previously mentioned damage points. In the case of skin penetration(s), indicate hole diameter(s). List all damage to radome and any other composite structure, such as fairings, control surfaces, etc. If lightning diverter strips are damaged, include lightning diverter strip location(s) on radome. For damage to composite structure, paint thickness must be included in description.

C. List any damage to avionics and electrical components believed to be the result of the lightning strike, including damaged wiring, disengaged circuit breakers, etc. Include manufacturer, model number and serial number of damaged units where applicable.

D. Estimate cost of repair.

E. Mention severity of damage (light, moderate, heavy).

F. Additional comments and descriptions:

Part 2 completed by: ______________________________________ Date __________ Phone ____________

Figure 5-1. Lightning Strike Reporting Form (Sheet 2 of 2)

Dangerous air turbulence or dangerous maneuvers—Air conditions that can dangerously shake an aircraft. Severe maneuvers are any maneuvers greater than those permitted by *AFM*.

Lightning strike—If flown through an electrically charged area of atmosphere, where electrical release is transferred from cloud-to-cloud and from cloud-to-earth, the aircraft can become a part of this flow. During a lightning strike, the current goes into the aircraft at one point and out at another (usually at opposite sides of the aircraft). Damage is most likely to occur in the wing tips, nose and tail sections.

Burns and/or erosion on small surface areas of the skin and structure can be identified during inspection. Usually the damage is seen easily, but damage that is hard to see can occur. The function of the lightning strike inspection (Figure 5-1) is to find any damage to the aircraft before it is returned to service.

Foreign object damage—An aircraft engine can be damaged by the ingestion of slush, a bird, or by any other foreign object, whether the aircraft is operated on the ground or in flight. Damage can also be caused during maintenance operations by:

- Tools
- Bolts, nuts, washers, rivets
- Rags
- Pieces of safety wire left in the engine inlet duct

Dents, nicks or scratches in the engine inlet are an indication of foreign object ingestion. The function of the foreign object damage inspection is to locate any damage before to the aircraft is repaired or returned to service.

Safety precautions must be taken to prevent foreign objects from touching the aircraft during towing (and at all times when aircraft is not in service). The engine inlet and tailpipe must have the correct covers to prevent corrosion in the compressor stages and damage to the fan disc and blades. When wind turns the engines, the covers must be installed as soon as possible after engine shutdown.

Cleanliness of the aircraft's aerodynamic surfaces increases its smoothness, which improves performance. It is most important that surfaces are kept very clean, especially the engine inlet cowling area.

Contour and distortion of the aerodynamic surface can occur during normal operation or by incorrect maintenance operations. Surfaces that are very curved (i.e., the engine inlet lip and inlet ducting) are areas where minor distortions can have a large effect on aircraft performance. Doors and access panels can be easily distorted by incorrect movement. Be careful when you touch these items.

Fuel unbalance—A fuel unbalance condition is when one wing has a larger quantity of fuel than the other. This can be due to a fuel system malfunction, incorrect refuel procedure, etc. It is best not to move an aircraft in this condition. If it must be moved, an inspection must be completed before the aircraft is returned to service.

High drag/side-load conditions—A high drag/side load condition is when:

- The aircraft skids
- Or leaves the prepared surface onto an unprepared surface
- The aircraft lands before the prepared surface
- Or lands and with a blown tire(s)
- Or goes into a skid on a runway , thus endangering the safety of the aircraft

This can occur during takeoff, landing, or in unusual taxi conditions.

Nose landing gear limits exceeded when towing—Turns that are greater than the limits of the nose landing gear when the aircraft is towed (with either a towbarless vehicle or towbar) can cause damage to the nose landing gear.

The turn limits for the tow method that follow are:

- When the aircraft is towed with the control lock disengaged, the turn limit is 90°.

- When the aircraft is towed with the control lock engaged, the turn limit is 60°.

NOTE

The nose wheel can be turned more than the black limit marks when the torque links are disconnected and kept apart from each other, and the tire. This prevents damage to the nose gear centering mechanism and steering stops.

NOTE

Cessna Aircraft Company does not recommend that the aircraft be towed with the Lektro tow vehicle when the torque links are engaged.

Hard or Overweight Landing Check

Landing Gear Check:

1. The main landing gear upper barrel-to-trunnion attachment (bolts and braze)—Examine for correct installation and indications of structural damage.

2. Main landing gear actuator attachments and support structure—Examine for correct installation, loose or unserviceable fasteners and indications of structural damage.

3. Nose landing gear trunnion at supports and attach structure—Examine for correct installation, unserviceable fasteners and any indications of structural damage.

4. Nose landing gear actuator attachments and support structure—Examine for correct installation, unserviceable fasteners and any indications of structural damage.

Wings Check:

1. Lower wing surface in the main landing gear area—Examine for skin buckles, unserviceable fasteners, correct installation of landing gear rib and trunnion fittings, rear spar web and fuel leaks.

2. Wing-to-stub wing fittings—Examine for correct installation, and any indications of structure damage.

3. Trailing edge assembly—Examine for any deformation that has an effect on normal flap operation.

4. Leading edge—The skin attach rivets along the leading edge of the wing inboard of the landing gear for loose rivets.

Overspeed Check

Landing gear Check:

1. Trunnion and supports—Examine for cracks, correct installation and indications of structural damage.

2. Doors and Attachments—Examine for unserviceable fasteners, cracks, buckles and indications of structural damage.

3. Examine to make sure components are free to move and do an operational check.

Fuselage Check:

1. Nose section—Examine for buckles, dents, unserviceable fasteners and any indications of structural damage.

2. All hinged access doors—Examine the hinges, hinge attach points, latches and attachments and skins for deformation and indications of structural damage.

Nacelles and Pylons Check:

1. Skins—Examine for buckles, cracks unserviceable fasteners and indications of structural damage.

2. Fillets and fairings—Examine for buckles, dents cracks and loose or unserviceable fasteners.

Stabilizers Check:

Examine the skins, hinges and attachments, surfaces that can move, mass balance weights and attach structure for cracks, dents, buckles, unserviceable fasteners and indications of structural damage.

Wings Check:

1. Flaps—Examine for skin buckles, cracks, unserviceable fasteners, attachments and structure for damage.

2. Complete a check of the flaps for freedom of movement operation.

Severe Turbulence and/or Maneuver Checks

Fuselage Check:

1. Forward and Center Fuselage—Do an inspection of the aircraft skin surface for buckles, wrinkles and deformations. Do a check for unserviceable or missing fasteners. Do an inspection of the areas around the wing attachments, the cabin door, and the emergency exit door for structural damage.

2. Tail Cone—Do an inspection of the aircraft skin surface for buckles, wrinkles and deformations. Check for unserviceable or missing fasteners. Do an inspection of the areas around the baggage door and the engine beams for structural damage.

Stabilizer Check:

1. Horizontal tail hinge fittings, actuator fittings and stabilizer center section—Examine for correct installation, unserviceable fasteners and any indications of structural damage.

2. Vertical tail—Examine for indications of structural damage, skin buckles and correct installation at the primary attachments in the tail cone, unserviceable fasteners, damage to hinges and actuator fittings.

3. Elevator and rudder balance weight support structure—Examine for correct installation, unserviceable fasteners and indications of structural damage.

Wing Check:

1. Wing to body fittings and support structure—Examine for correct installation, unserviceable fasteners and indications of structural damage.

2. Trailing Edge—Examine for any deformation that will effect normal operation of flap and aileron.

3. Leading edge—Examine the skin attach rivets along the leading edge of the wing inboard of the landing gear for loose rivets.

Inadvertent In-Flight Thrust Reverser Deployment

Fuselage Check:

Examine the tail cone skins, stringers and frames aft of the aft pressure bulkhead for wrinkles, cracks, unserviceable or loose fasteners or bonds, or other damage. Examine the strakes and their installation for damage.

Nacelles Check:

Examine the nacelles, pylons, engine beams and related attach fittings, and thrust reversers for buckles, cracks, unserviceable fasteners and indications of structural damage.

Stabilizer Check:

1. Horizontal stabilizer and elevator hinges and hinge fittings, actuator hinge fittings connections and idlers—Examine for correct installation, unserviceable fasteners and any indications of structural damage. Examine the forward spar area of the stabilizer at the stabilizer actuator attachment and the stabilizer aft spar at the pivot for damage and unserviceable fasteners.

2. Vertical stabilizer and rudder—Examine the stabilizer for indications of structural damage, skin buckles and correct installation at the primary attachments in the tail cone. Complete a check for cracks, or unserviceable fasteners at the attach points. On the rudder, complete a check for damage to the hinges, hinge fittings, actuators, and actuator fittings.

Lightning Strike Check

As the checks are done that follow, complete the Lightning Strike/Static Discharge Incident Reporting Form. Mail the completed form to:

Cessna Citation Customer Service
P.O. Box 7706, Wichita, KS 67277
Attn: Avionics Customer Service.

Communications Check:

Antennas—Examine all of the antennas for indications of burns or erosion. If damage is found, complete a functional check of the applicable system.

Navigation Check:

1. Radar reflector, feed horn, motor box assembly and mount structure—Examine for damage. If damage is found, complete a bench check of the system. If pits or burns in the surface of the mount structure only is found, complete a functional check of the radar system.

2. Glideslope antenna—Examine for burns and pits. If damage is found, complete a functional check of the glideslope system.

3. Standby compass—Must be thought of as serviceable if the corrected heading is within ±10° of the heading indication from the remote compass system. If the remote compass is not within tolerance, remove, repair or replace, as necessary.

Fuselage Check:

1. Radome—Examine for indications of burns or erosion.

2. Skin—Examine the surface of the fuselage skin for indications of damage.

3. Stinger—Examine the static discharge wicks for damage.

Stabilizers Check:

1. Rudder—Examine the static discharge wicks for damage.

2. Elevator—Examine the static discharge wicks for damage.

Wings Check:

1. Skins—Examine for indications of burns and erosion.

2. Wing Tips—Examine for indications of burns and pits. Examine the static discharge wicks for damage.

3. Flaps, ailerons and speed brakes—Examine for burns and pits. Examine the static discharge wicks for damage.

Engine Check:

Refer to the manufacturer's *Engine Maintenance Manual* for lightning strike inspection.

Foreign Object Damage Check

Landing Gear Check:

Doors—Examine for dents, cracks, and indication of structural damage. Also make sure the door is not incorrectly aligned.

Fuselage Check:

1. Radome—Examine for dents, cracks, punctures, scratches, etc.

2. Skin—Examine the forward and belly areas for dents, punctures, cracks and any indications of damage.

Nacelles/Pylons Check:

Skins—Examine for dents, punctures, unserviceable fasteners, cracks and indications of structural damage.

Stabilizers Check:

Leading Edge Skins—Examine for dents, cracks, scratches and any indications of structural damage.

Windows Check:

Examine windshield for chips, scratches and cracks.

Wings Check:

Examine the leading edge skins for dents, cracks, punctures and indications of possible structural damage.

Powerplant Check:

1. Cowling—Examine for dents, cuts, tears, scratches, blood and feathers.

2. Air inlet section—Examine for dents, cracks, scratches, punctures, blood and feathers.

3. Fan—Examine for bent, broken or cracked blades. Make sure the blades do not have nicks or rubs.

Refer to the manufacturer's *Engine Maintenance Manual* for additional inspection procedures that are necessary after a bird strike or ingestion.

Towing with Large Fuel Unbalance or High Drag/Side Loads Due to Ground Handling Check

Landing Gear Check:

1. Main landing gear and doors—Examine for unserviceable fasteners, buckles, correct installation, cracks, and indications of structural damage.

2. Nose landing gear and doors—Examine for unserviceable fasteners, cracks, correct installation, buckles, and indications of structural damage.

Aircraft Operation Through Deep Standing Water

Visually examine the elevator trim tabs for delamination or peeling along the trailing edge.

Nose Landing Gear Tow Limits Check

Control Lock Disengaged Check:

Complete the checks that follow, if the aircraft nose landing gear is turned past the limit with the control lock disengaged.

1. Nose wheel steering gear attach bolts on top of the upper barrel—Examine for correct installation, loose or unserviceable fasteners, or indications of damage.

2. Nose wheel steering gear stops on top of the trunnion—Examine for correct installation, loose or unserviceable fasteners, or indication of damage to the structure.

3. Steering bungee, bell crank, steering cable brackets and steering cables—Examine for correct installation, loose or unserviceable fasteners, or indication of damage to the structure.

Control Lock Engaged Check:

Complete the checks that follow, if the aircraft is turned past the limit with the control lock engaged.

1. Nose wheel steering gear attach bolts on top of the upper barrel—Examine for correct installation, loose or unserviceable fasteners, or indications of damage.

2. Nose wheel steering gear stops on top of the trunnion—Examine for correct installation, loose or unserviceable fasteners, or indication of damage to the structure.

3. Steering bungee, bell crank, steering cable brackets and steering cables—Examine for correct installation, loose or unserviceable fasteners, or indication of damage to the structure.

4. Rudder cables and cable brackets, rudder pass-thru sector, and rudder control lock system—Examine for correct installation, loose or unserviceable fasteners, or indication of damage to the structure.

NOTES

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CHAPTER 6
DIMENSIONS AND AREAS

DIMENSIONS AND AREAS

This chapter includes illustrations and statistical information concerning the Citation XL/XLS/XLS+ aircraft. Provided are overall aircraft dimensions, surface areas, station locations, location of major structural members, access plates, panels, floorboards, fairings, aircraft zoning and aircraft drain locations.

Dimensions and measurements are presented to aid the operator and/or maintenance personnel in ground handling the aircraft and locating the components. Measurements are expressed in feet (meters), inches (millimeters), and degrees. (Figure 6-1).

DESCRIPTION

This section identifies dimensions and areas of the aircraft and aircraft components in tabular form. Dimensions are selected for pertinent information of measurements that will aid the operator in providing storage, passing through hangar doors, covering isolated areas of the aircraft, and building/ordering maintenance stands. The dimensions are expressed in feet, inches, degrees, and minutes. Aircraft assembly areas are expressed in square feet (Table 6-1, Sheets 1 and 2).

NOTES

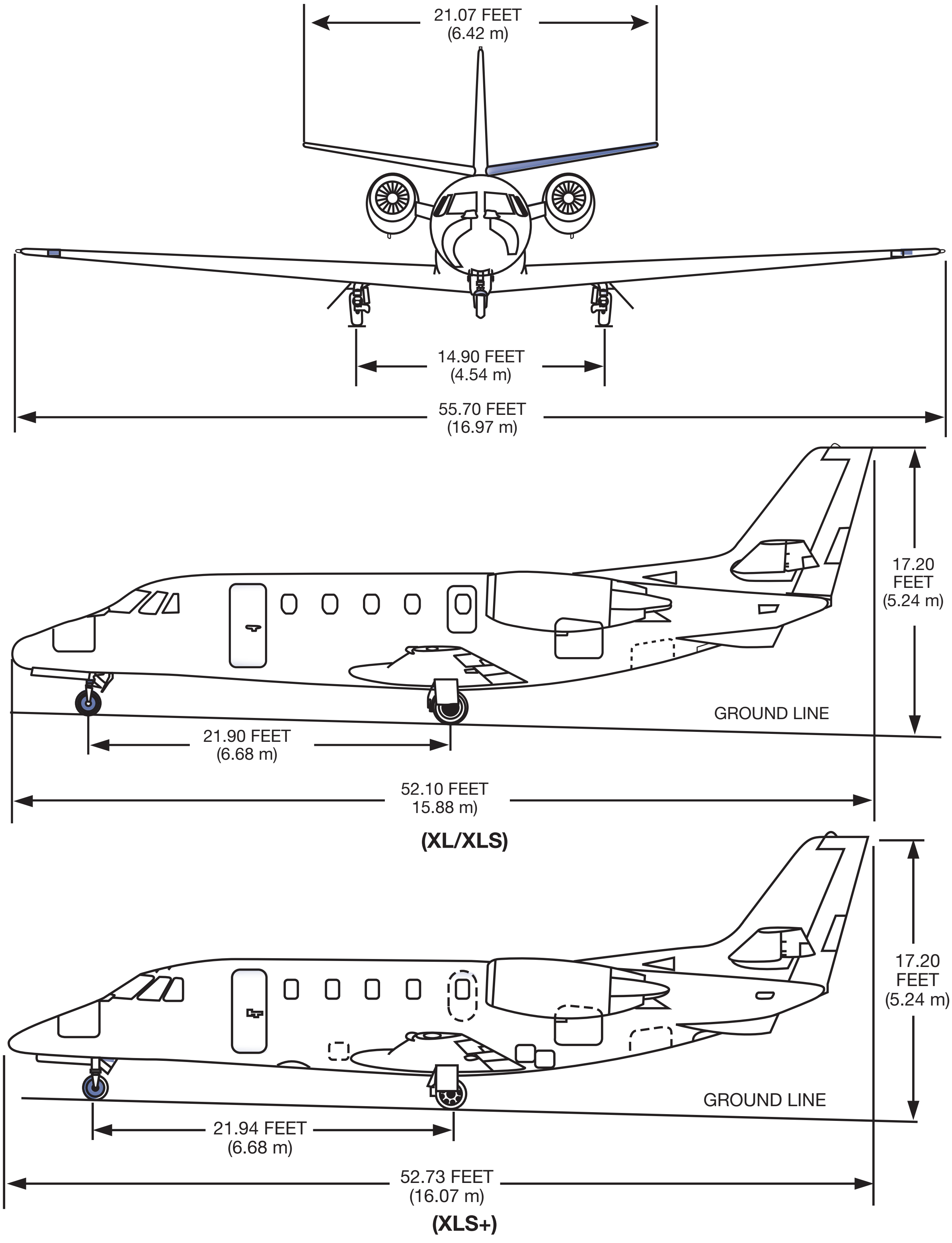


Figure 6-1. Airplane Views

Table 6-1. AIRPLANE DIMENSIONS

Airplane (Overall)	
Wing span	55.70 feet (16.98 m)
Length	52.10 feet (15.88 m) (Aircraft -5001 thru -6000) 52.73 feet (16.07 m) (Aircraft -6001 and Subsequent)
Height	17.20 feet (5.24 m)
Wing	
Chord	
WS 34.00	10.80 feet (3.29 m)
WS 101.073	7.58 feet (2.31 m)
WS 335.023 (construction tip)	2.78 feet (849.4 mm)
WS 136.685 (mean aerodynamic)	6.85 feet (2.09 m)
Dihedral	4°
Sweep back (35% chord)	0°
Ailerons	
Span	8.46 feet (2.58 m)
Root chord (aft of hinge line)	22.58 inches (573.5 mm)
Tip chord (aft of hinge line)	14.60 inches (370.84 mm)
Trim tab span (along hinge line)	3.74 feet (1.14m)
Trim tab chord root	6.06 inches (153.92 mm)
Trim tab chord tip	3.45 inches (87.63 mm)
Flaps	
Span (per wing)	11.21 feet (3.42 m)
Percent wing chord	25%
Horizontal Tail	
Span	21.07 feet (6.42 m) (Aircraft -5001 thru -6000) 21.50 feet (6.55 m) (Aircraft -6000 and Subsequent)
Root chord (BE 0.00)	5.44 feet (1.65 m)
Tip chord (BE 126.42)	2.61 feet (795.8 mm)
Sweep back (leading edge)	10.23°
Sweep back (trailing edge)	–4.86°
Dihedral	9°
Incidence (nose up)	1° or +0.1°
Incidence (nose down)	2° or –0.1°
Elevator Trim Tab	
Span (at hinge line)	4.21 feet (1.28 m)
Root chord (Aft of hinge line)	10.84 inches (275.3 mm)
Tip chord	6.51 inches (165.4 mm)

Table 6-1. AIRPLANE DIMENSIONS (Cont)

Vertical Tail	
Span (for equivalent exposed area	8.71 feet (2.65 m)
Root chord (water line 138.90)	8.50 (2.59 m)
Tip chord (water line 254.75)	3.67 feet (1.12 m)
Sweep back (25% chord)	32.90°
Rudder Trim Tab	
Span	2.09 feet (637 mm)
Root chord (WL 171.45)	10.73 inches (272.5 mm)
Tip chord (WL 196.55)	6.96 inches (176.8 mm)
Fuselage	
Fuselage constant section inside diameter	6.04 feet (1.84 mm)
Forward pressure bulkhead to aft pressure bulkhead	23.31 feet (7.10 m)
Length forward divider to aft pressure bulkhead	17.48 feet (5.33 m)
Height aisle floor to ceiling	5.67 feet (1.73 m)
Areas	
Wing (total)	370.60 square feet (34.43 square meters) (Aircraft -5001 thru -6000)
	369.70 square feet (34.16 square meters) (Aircraft -6001 and Subsequent)
Aileron (aft of hinge) each	20.25 square feet (1.88 square meters)
Aileron (total) each	27.68 square feet (2.57 square meters)
Aileron trim tab (aft of hinge line)	3.54 square feet (0.33 square meters)
Flaps (per wing)	20.48 square feet (1.90 square meters)
Horizontal tail (total)	84.84 square feet (7.51 square meters)
Elevator (aft of hinge line)	25.50 square feet (2.37 square meters)
Elevator trim tab	6.10 square feet (0.57 sqaure meters)
Vertical tail (total, exposed area above tail cone)	50.88 square feet (4.73 square meters)
Fin (exposed)	36.00 square feet (3.34 square meters)
Rudder (aft of hinge line) (exposed)	14.88 square feet (1.38 square meters)
Rudder trim tab	1.57 square feet (0.15 square meters)
Speedbrake (total per wing)	5.37 square feet (0.50 square meters)

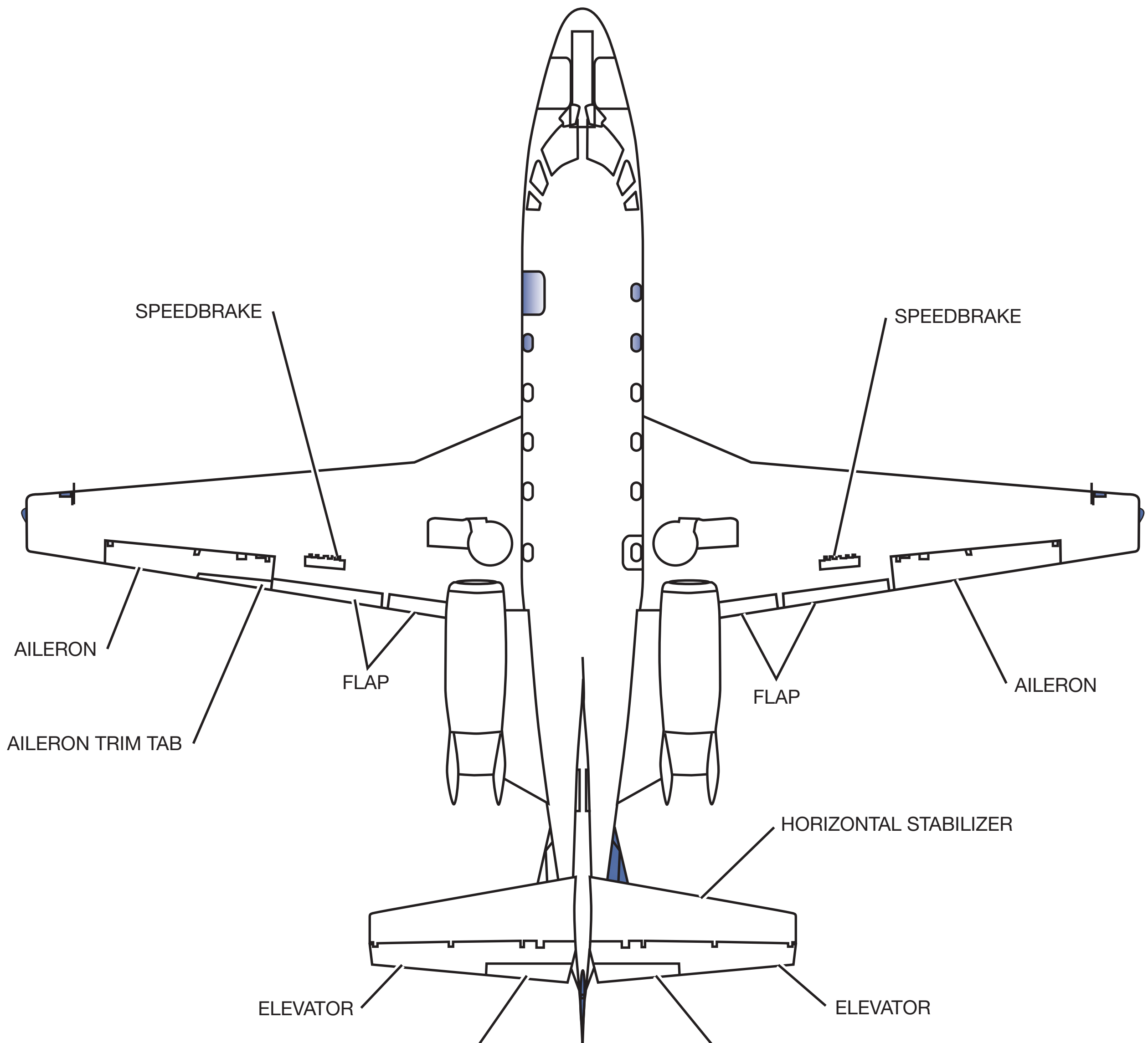


Figure 6-2. Airplane Areas (Sheet 1 of 2)

AIRCRAFT LOCATIONS

Description

This section describes the aircraft reference points to facilitate in locating specific areas of the aircraft. Throughout the *AMM*, component locations and locations of assemblies or major structures are identified by a particular station, water line and/or buttock line. To assist the maintenance personnel in visualizing a location on the aircraft, illustrations are provided for comparing the component, assembly or major structure location with the pictorial view provided. (Figure 6-2).

Abbreviations and Terminology.

The Citation XL/XLS/XLS+ aircraft utilizes a reference point of 30.70 inches in front of the nose (radome) for its fuselage station (FS) datum line.

Datum Line

A datum line is an imaginary plane or line from which distances are measured. The distance to a given fuselage station is measured in inches from the datum line, in front of the aircraft aft perpendicular to the center line.

Center Line

The center line of the aircraft is the imaginary vertical plane extending lengthwise through the middle of the fuselage.

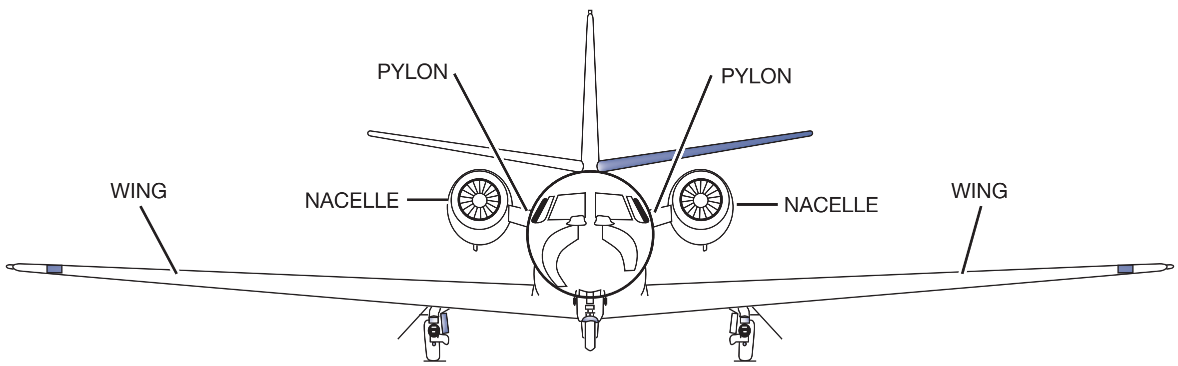
Buttock line

The buttock line (BL) is a width measurement to the left or right of, and parallel to, the vertical center line. Measurements in inches to the left of the aircraft center line are identified as left buttock lines (LBL) and measurements to the right are identified as right buttock lines (RBL).

Water line

The water line (WL) is the measurement of height in inches from a horizontal plane at a fixed number of inches below the bottom of the fuselage. For the Citation XL/XLS/XLS+, the zero water line is 91.00 inches below the bottom of the fuselage.

NOTES



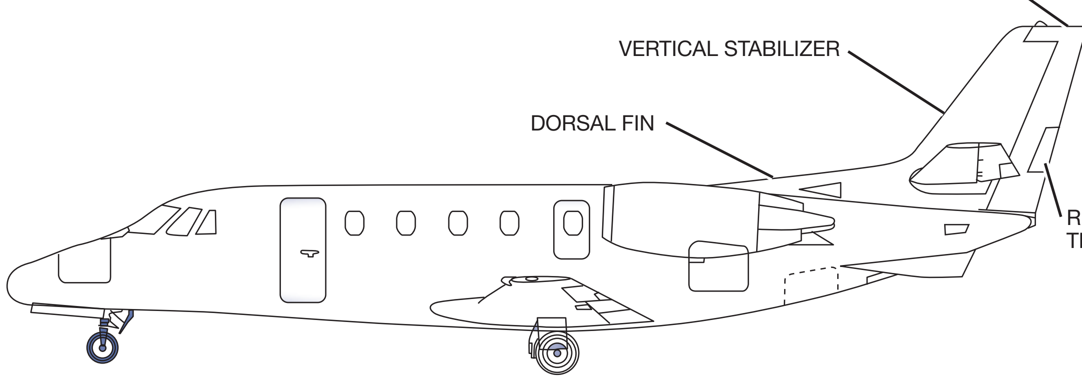


Figure 6-2. Airplane Areas (Sheet 2 of 2)

The horizontal stabilizer station (HSS) is a length-measurement (in inches) of the horizontal stabilizer from the aircraft center line, outboard to the left or right stabilizer tip (parallel to the fuselage).

Vertical Stabilizer

The vertical stabilizer structure locations are identified by the fuselage stations and water lines extending through the vertical stabilizer, sometimes called the vertical fin.

Wing Station

The wing station (WS) is a length-measurement (in inches) of the wing, from the aircraft center line, outboard to the left or right wing tip (parallel to the fuselage).

Chord

Chord is defined as a straight line intersecting or touching an airfoil profile at two points. A chord line is usually a datum line joining the leading and trailing edges of an airfoil. Points or stations along a chord are designated in percentages or fractions of the chord, or the chord length from the leading edge.

- A root chord is the length of the chord where the wing and fuselage join.
- A chord is the length of the chord at or near the wing tip.
- A mean chord is the gross airfoil surface area divided by the span.

NOTES

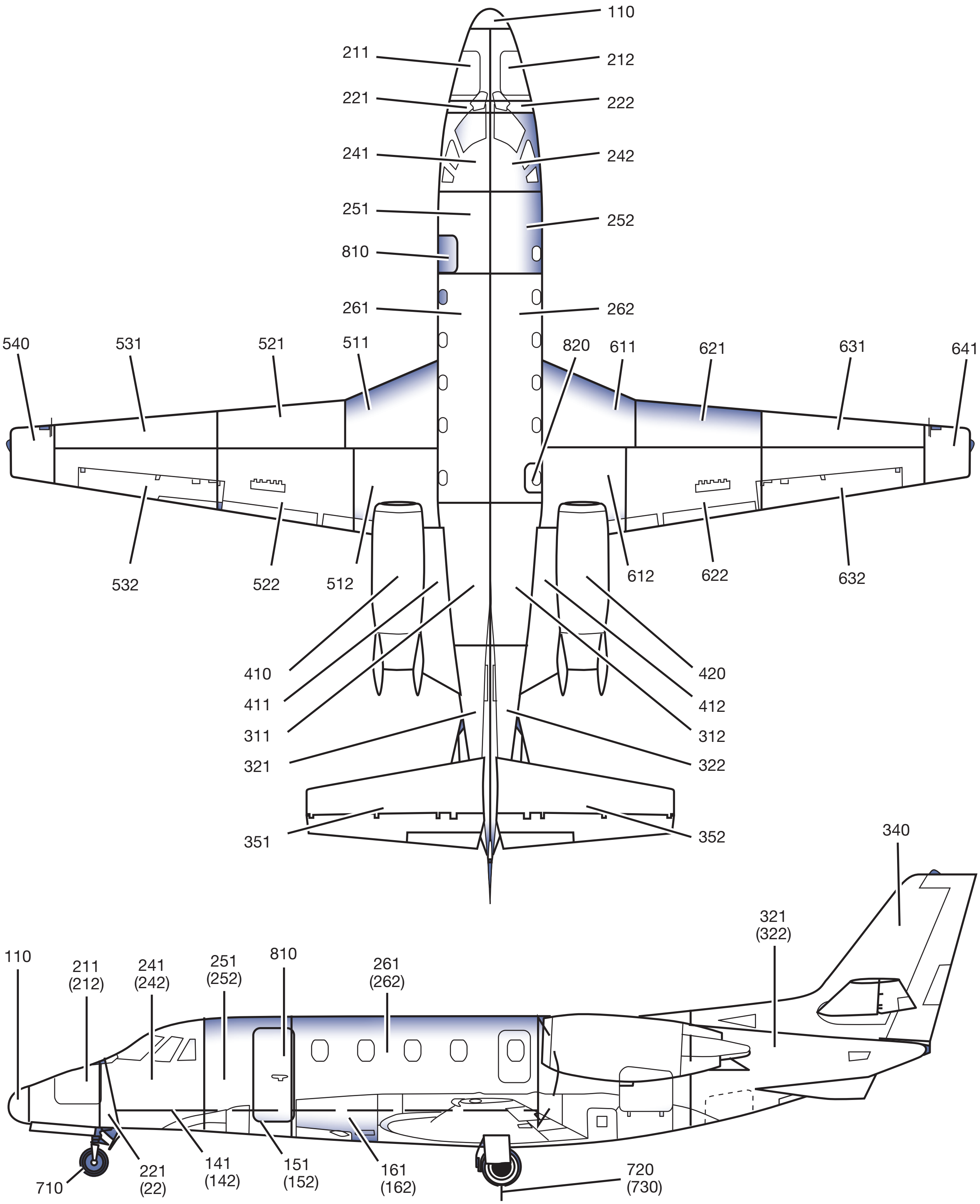


Figure 6-3. Airplane Zones (Sheet 1 of 3)

AIRCRAFT ZONING

The Citation XL/XLS/XLS+ is divided into numbered zones to provide a method for locating components. The zones are identified by a three-digit number as shown in the example below. Each digit designates a zone category: major, sub-major or subdivision (Figure 6-3).

Major Zones:

- 100—Radome and area below nose compartment shelves and below cabin floorboards to rear pressure bulkhead
- 200—Area above the nose compartment shelves and cabin floorboards to behind the pressure bulkhead
- 300—Empennage
- 400—Nacelle area outboard of firewall
- 500—Left wing
- 600—Right wing
- 700—Landing gear and landing gear doors
- 800—Entrance door and emergency exit door

Description

Aircraft zones may be utilized to locate work areas and components before beginning maintenance or servicing tasks on the aircraft. Aircraft zones are used in this manual to locate items such as placards and markings that are displayed on interior and exterior surfaces of the aircraft.

NOTES

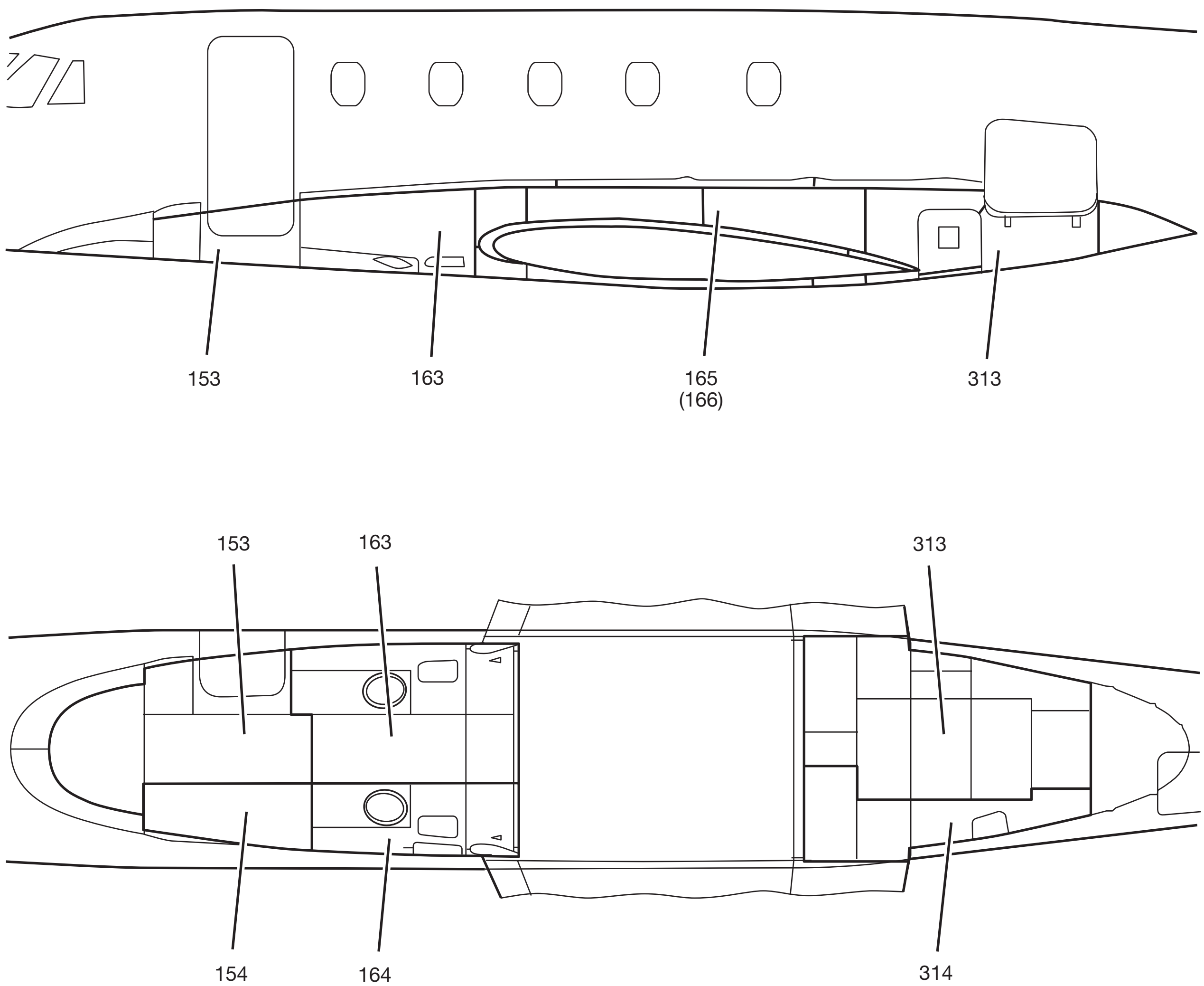


Figure 6-3. Airplane Zones (Sheet 2 of 3)

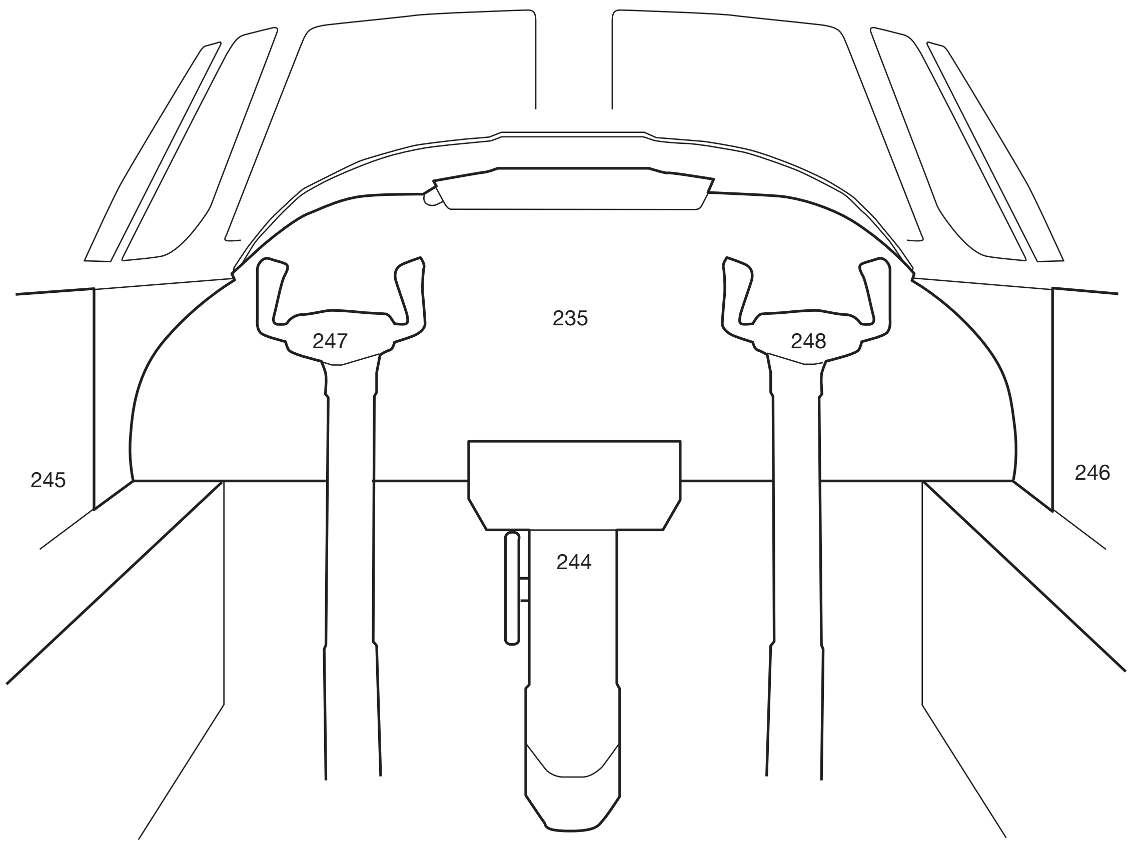


Figure 6-3. Airplane Zones (Sheet 3 of 3)

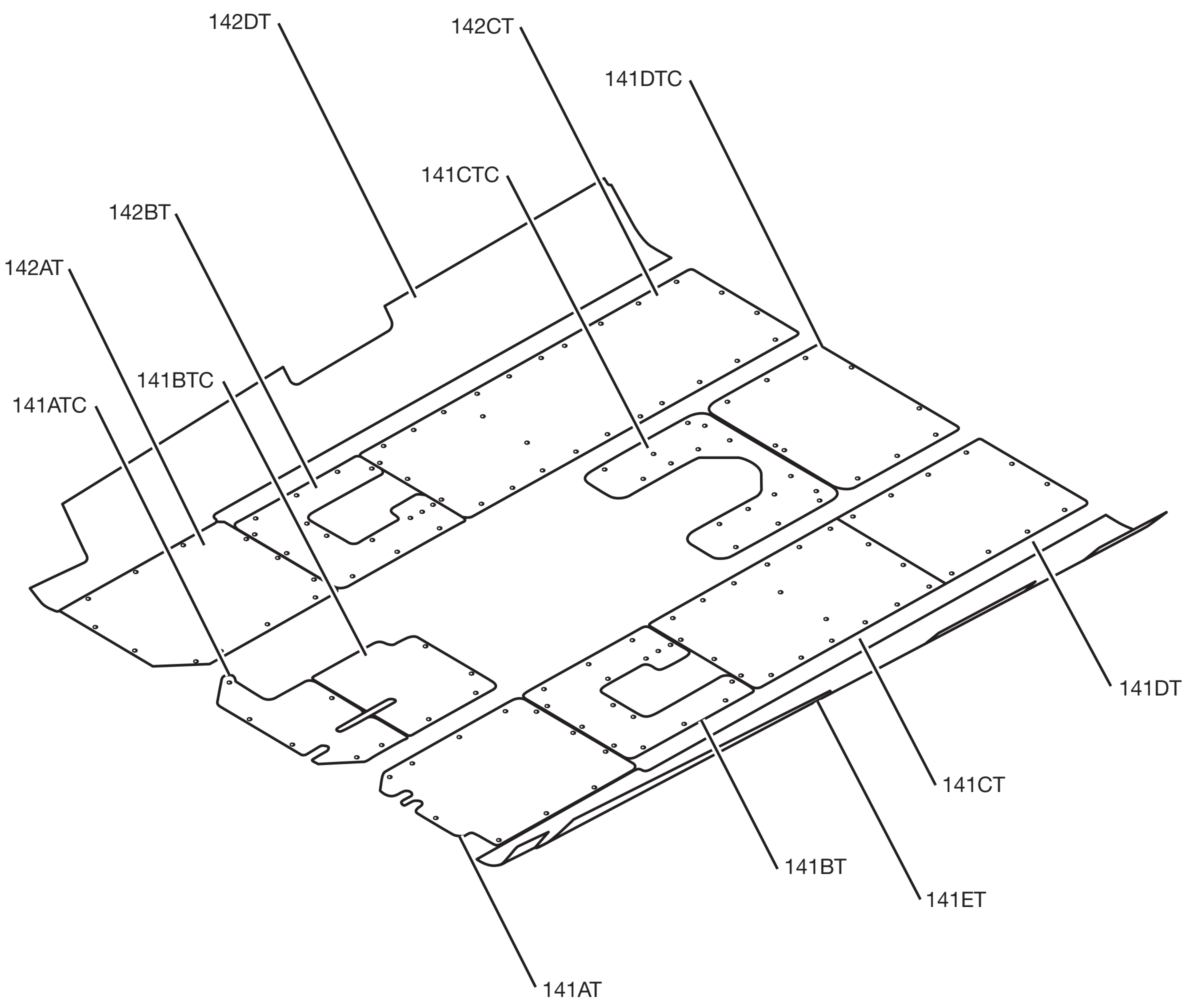


Figure 6-4. Cockpit Floorboard Panels

ACCESS PLATES AND PANELS IDENTIFICATION

Description

All access plates, panels, and doors are identified by using the aircraft zone number plus one or two suffix letters as shown in the example below. The first suffix letter is the primary identifier. The primary identifier identifies the plate, panel, or door in a logical sequence (i.e., inboard, outboard, forward, or aft; starting with the letter in each zone).

The second suffix letter identifies the plate, panel, or door in its relation to the aircraft (i.e., top, bottom, left, right, or internal).

Remove and install the access plates, panels, and doors in accordance with the applicable chapter. Refer to Figure 6-4 and Table 6-2.

NOTES

Table 6-2. EQUIPMENT IN AREA

PANEL	EQUIPMENT LOCATED IN AREA
141ATC	Nose wheel steering cables and pulleys, nose wheel steering mixer assembly, rudder interconnect cable, fuselage pressure seal
141AT	Nose wheel steering cables and pulleys, pedestal, hydraulic lines to nose gear
141BTC	Cockpit warm air crossover duct, forward elevator control sectors, aural warning system, structural ground, hydraulic lines, electrical connectors (PC517 and PF517), wire bundles, control lock cable
141BT	Emergency brake hydraulic lines left throttle switch assembly, left throttle RVDT
141CTC	Aileron brake hydraulic lines, left throttle switch assembly, left throttle RVDT
141CT	Emergency brake hydraulic lines, warm air distribution ducts, service air line to throttle bell crank, rudder cable turnbuckle, right 80% throttle switch, rudder trim cables
141DTC	Aileron forward sector, rudder forward sector, elevator forward sector
141DT	Control cables and pulleys, wire bundles, cockpit warm air crossover duct, fuselage pressure seal, service air line to throttle bell crank
141ET	Cockpit warm air duct, cockpit warm air duct control cable, wire bundles, static ports
142AT	Nose wheel steering mixer assembly
142BT	Electrical connector (PF906), ground (GF042), control column torque tube, flap position transmitter, electrical connector (PF041), right throttle switch assembly, right throttle RVDT
142CT	Cockpit warm air crossover duct, control cables and pulleys
142DT	Cockpit warm air duct control cable, cockpit warm air duct, wire bundles, control cables, static ports

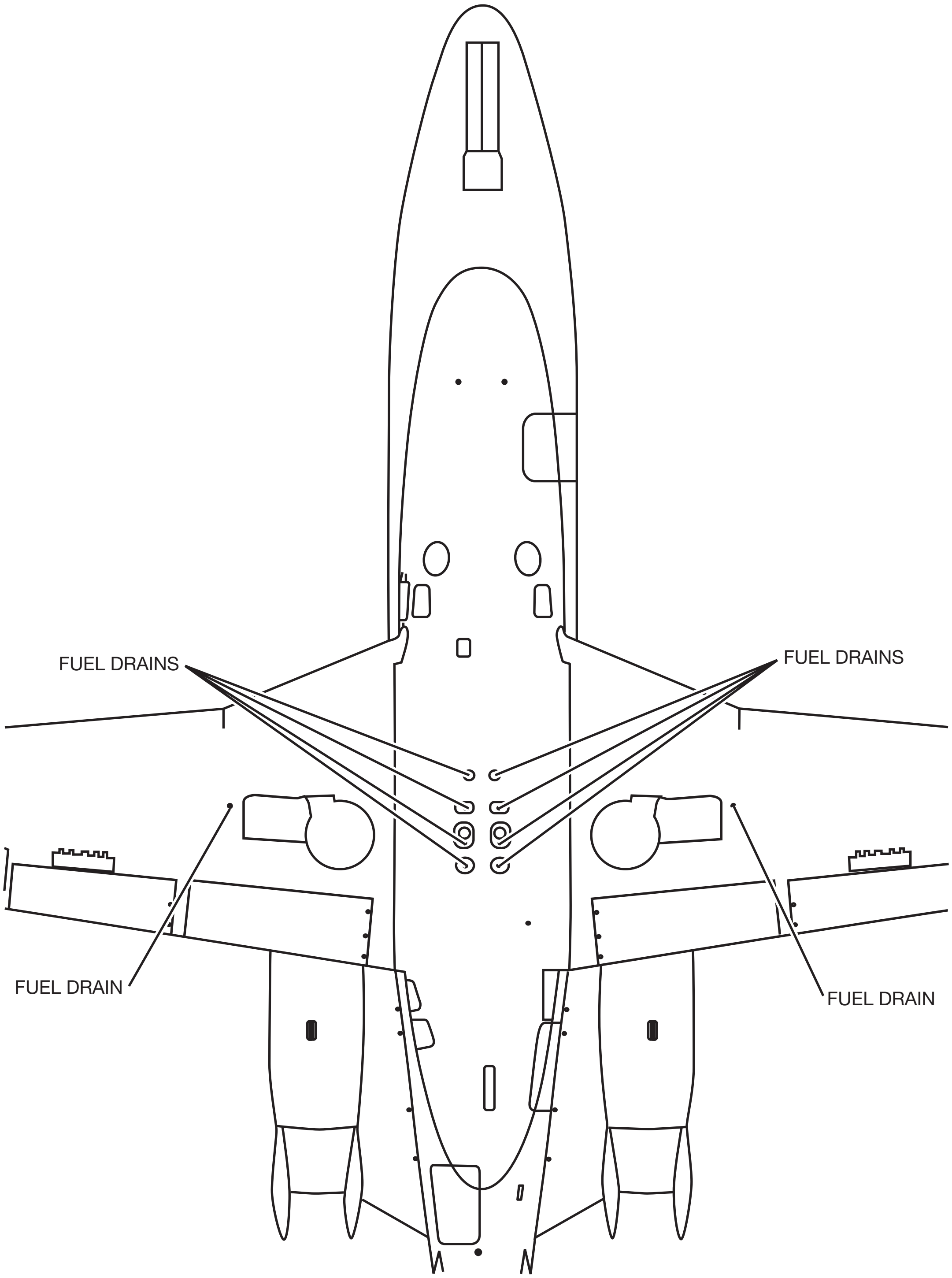


Figure 6-5. Airplane Drain Line and Vent Locations

AIRCRAFT DRAIN LOCATIONS

This section describes and locates the drains of various systems throughout the aircraft. (Figure 6-5).

Description

Drain locations are specified by the fuselage station, water line or buttock line where the drain protrudes through the fuselage. For removal and installation of drain lines, refer to the applicable chapter.

NOTES

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CHAPTER 7
LIFTING AND SHORING

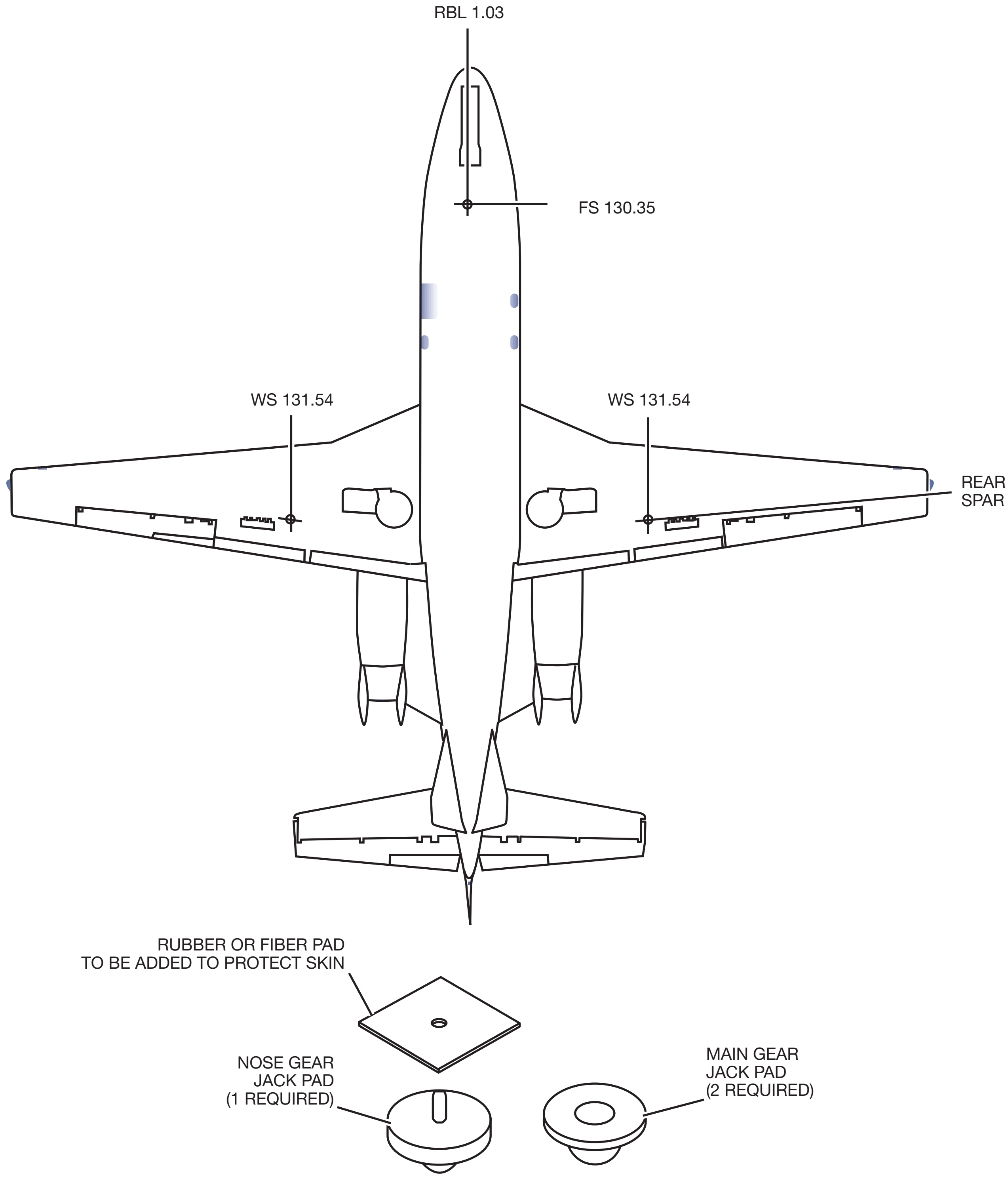

Figure 7-1. Wing and Fuselage Jack Points

LIFTING AND SHORING

DESCRIPTION

This chapter describes the standard methods of lifting and shoring the aircraft. This includes the standard method of lifting during an emergency condition. A method of shoring is also described in this chapter. Refer to Figure 7-1 for wiring and fuselage jack points.

Lifting

CAUTION

Disengage the TAS HEATER circuit breaker on the left CB panel prior to jacking or lifting the aircraft. The TAS probe heater becomes active when weight is removed from the wheels. Disengaging the TAS HEATER circuit breaker removes 28VDC power from the heater, preventing possible injury to personnel and/or damage to aircraft components.

NOTE

Disengage FLT HOUR METER circuit breaker on the left CB panel prior to jacking or lifting aircraft. The flight hour meter becomes active when weight on wheels is removed.

Standard jacking of the aircraft utilizes tripod-type jacks. There is one jack point adjacent to the nose gear area and a jack point outboard of each main gear wheel well.

Individual jacking of a main landing gear is accomplished by utilizing a hydraulic jack.

Emergency Lifting

Emergency lifting is accomplished by utilizing air bags. Air bags are normally ground support emergency equipment. The air bags are used to lift the aircraft enough that jacks can be placed under the aircraft, or enough that a dolly can be placed under the aircraft.

LIFTING

Description

The entire aircraft may be lifted at wing and fuselage jack points to:

- Perform landing gear tests
- Remove and install nose and/or main gear assemblies
- Level the aircraft for major repairs

One wheel may be lifted for tire and landing gear repairs using axle jack.

If possible, position the aircraft on a level surface when jacking. The jacking site must be protected from the wind, preferably in a hangar.

The aircraft is limited to 18,700 pounds ramp weight when jacking.

Tools and Equipment

For tool and equipment listing, refer to “Lifting and Shoring” in the *AMM*.

Jacking Instructions

CAUTION

The aircraft must be jacked inside a hangar. If conditions require jacking the aircraft outdoors, jacking must be done in calm or light wind conditions with the aircraft headed into the wind. The aircraft must be on a level surface when jacking and only approved jacks can be used.

Jacking Aircraft (Three Wheels)

1. Disengage the following circuit breakers on the CB panel:
   - GEAR CONTROL
   - L PITOT STATIC
   - R PITOT STATIC
   - STDBY P/S HTR
   - TAS HEATER
   - AOA HEATER
   - L START
   - R START
   - L IGNITION
   - R IGNITION
   - L FUEL BOOST
   - R FUEL BOOST
   - FLT HR METER
2. Ensure that the aircraft is electrically (static) grounded.
3. Insert jack pads at each jack point.
4. Position a jack below each jack pad.
5. Ensure that the jack cylinders are vertical at start of jacking operations to prevent side loads and possible gear strut binding.

   a. Check the ballast in the nose avionics compartment. Removal of equipment or furnishings from the aircraft forward of the aircraft center of gravity will unbalance the weight distribution and may cause a hazardous condition (aircraft may tilt and rest on the tail assembly). Likewise, adding weight aft of the center of gravity may produce the same weight distribution hazard. If equipment is removed from the aircraft or weight added aft of the center of gravity, an appropriate amount of ballast is to be placed in the nose compartment.

WARNING

Make sure that the parking brake is released before the aircraft is lifted. The main gear wheels must turn when the aircraft is lifted. If the wheels cannot turn, structural damage to the aircraft and injury to personnel can occur when the aircraft is lifted.

6. Make sure that the parking brake is released after the jacks are set and before the aircraft is lifted.
7. At the same time, raise the wing and the fuselage jacks. Keep the aircraft level until the tires are off of the ground. Keep the follower nut of each jack against the jack shoulder.

   a. Do not raise the tires more than the necessary distance to do the maintenance.
8. Put the tail stand under the tail cone below the forward canted bulkhead 10.75 inches (273 mm) forward of the leading edge of the access panel 321ABC. Refer to Chapter 6—“Access Plates and Panels Identification”.

Lowering aircraft (Three Wheels)

1. Remove tail stand from under tail cone.

WARNING

Make sure that the parking brake is released before lowering the aircraft. The main gear wheels must turn when the aircraft is lowered. If the wheels cannot turn, structural damage to the aircraft and injury to personnel can occur when the aircraft is lowered.

2. If necessary, release the parking brake.
3. Turn the main gear wheels with your hands to make sure that the parking brake is not set.

4. Before lowering the aircraft, make sure that there are no maintenance stands, or other equipment under the aircraft (or near the engines or wings) that could touch the aircraft.

5. Loosen the jack follower nuts; lower the fuselage and wing jacks simultaneously.

6. Remove the jacks and jack pads; stow jack pads.

7. Engage the following circuit breakers on the CB panel:
   - GEAR CONTROL
   - L PITOT STATIC
   - R PITOT STATIC
   - STANDBY P/S HTR
   - TAS HEATER
   - AOA HEATER
   - L START
   - R START
   - L IGNITION
   - R IGNITION
   - L FUEL BOOST
   - R FUEL BOOST
   - FLT HR METER

Jacking One Wheel

CAUTION

If nose gear is being jacked, remove the tail stands and maintenance equipment (aft of the main gear) that may cause damage when the aft portion of the aircraft is lowered. If the ground is level enough to permit, release the aircraft brakes while jacking to avoid pulling aft on jack as nose rises. After jacking is complete, install the tail stand.

CAUTION

Ensure that the antiskid transducer wire bundle is clear of jack and adapter. Ensure that the wire bundle is not contacted or damaged during the jacking procedure.

1. Ensure that the aircraft is electrically (static) grounded.

2. Position the hydraulic jack under the trailing link jack point for gear to be jacked.

3. Extend jack until tire is clear of the ground.

4. Position the tail stand under tail cone below the forward canted bulkhead at 10.75 inches (273 mm) forward of the leading edge of access panel 321ABC. Refer to Chapter 6—"Access Plates and Panels Identification."

Lowering One Wheel

1. Remove the tail stand from under the tail cone.

CAUTION

When lowering the aircraft ensure that the parking brake is off. If the parking brake is set, the aircraft may roll forward off the jacks as trailing link gear compresses.

2. Release the parking brake.

3. On completion of maintenance, retract the jack until weight is assumed by tire.

4. Remove the jack.

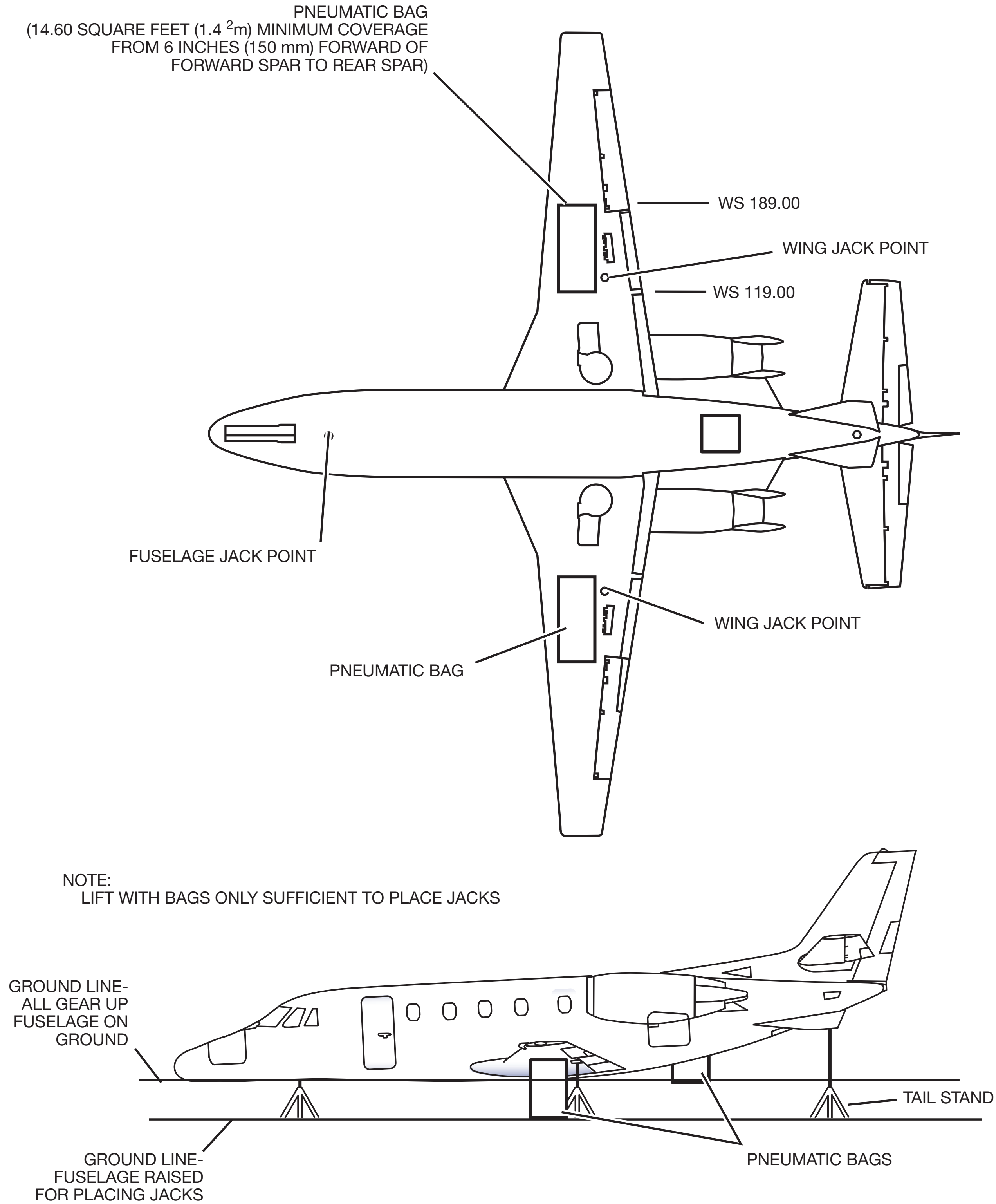

Figure 7-2. Emergency Lifting Airplane

EMERGENCY LIFTING

Description

Emergency lifting is a procedure to lift the aircraft from an abnormal position while subjecting the airframe to the least amount of damage (Figure 7-2). The methods of emergency lifting described in this section do not limit emergency procedures; all approved alternate methods may be used when emergency conditions warrant such action.

An aircraft that has belly-landed or one with collapsed landing gear can be lifted using pneumatic bags and jacks. For an aircraft resting on the runway (or equivalent hard surface) in the nose-down condition, a pneumatic bag may be placed under the fuselage to lift the aircraft enough to place a jack on the nose jack point. For an aircraft landing on soft surface (plowed field), it may be necessary to dig sufficient clearance below the fuselage and/or wing to place the pneumatic bag.

A tail pull-down method may be used for lifting a nose-down aircraft. However, damage to the tail cone structure may result.

NOTE

The tail pull-down method should be used as a last choice alternate method to pneumatic bags and jacks.

NOTES

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CHAPTER 8
LEVELING AND WEIGHING

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LEVELING AND WEIGHING

DESCRIPTION

This chapter describes procedures for aircraft leveling and weighing. Alternate methods of leveling and weighing, not in this chapter, can be used to weigh and level the aircraft.

The aircraft must be operated within definite weight and balance limits. Therefore, weight and center-of-gravity must be calculated accurately.

Aircraft leveling is necessary for specific maintenance functions. These leveling requirements are defined in the particular system chapter.

Aircraft weighing procedures must include a weight and balance manual and scales to weigh the aircraft. The empty weight and center-of-gravity are calculated from information when the aircraft is weighed.

NOTES

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CHAPTER 9
TOWING AND TAXIING

TOWING AND TAXIING

DESCRIPTION

This chapter describes procedures used in towing and taxiing the aircraft. Observe local requirements that involve the operation of taxiing and towing (designated ramps, ramp speed, etc.).

TOWING

The aircraft can be towed forward or aft, on hard surfaces, using a yoke-type towbar attached to the upper fork buckets on the fork assembly. Towing can also be done for fuel loading with no passengers aboard (except for one flight or ground crew member).

Towing can be done with the nose wheel properly cradled on the lift platform of the correct model Lektro towbarless vehicle. Refer to "Towing" in the *AMM*, for nose wheel turning limits.

Towing the aircraft with a flat tire is not recommended. However, at times, the aircraft may have to be moved from an active runway or taxiway. Tow the aircraft forward a minimum distance to clear the runway or taxiway. Avoid sharp turns. The tire must be considered destroyed, and the wheel must be inspected in accordance with the manufacturer's overhaul manual.

Towing Procedure

To tow the aircraft normally, the nose gear upper fork buckets (on the fork assembly) connect to a yoke-type towbar. A towbarless tow vehicle with the nose wheel on the lift platform may be used.

- While the aircraft is being towed, the vehicle operator must make sure that the nose gear does not turn more than the specified turning limits. If the nose wheel is turned more than the specified limit, the steering gear attaching bolts are destroyed.
- During nose gear wheel towing, all turning is done with the towbar or towbarless vehicle.
- If the aircraft is off the runway or taxiway in sand, soft ground, or mud, towing may be accomplished with the aid of cables or ropes attached to each main gear towing adapter. When towing is done by attaching cables or ropes to the main landing gear assemblies, the rudder pedals are used to steer the aircraft.

A qualified person may be stationed in the aircraft during towing operations to be prepared for hazardous conditions. For example, if the towbar breaks or becomes detached between the aircraft and the towing vehicle.

In congested areas, station wing and/or tail walkers must be stationed to make sure there is enough clearance between the aircraft and adjacent equipment and structures.

Safety Precautions for Towing

CAUTION

Do not force the nose gear beyond the towing stop (90° limit). The bolts that attach the steering gear assembly to cylinder are sheared when this occurs. The maximum nose gear towing turning angle limit is 90° either side of center.

CAUTION

When towing the aircraft, make sure the recommended Lektro tug tow vehicle is used. To make sure the towbar is serviceable, do a periodic inspection of the towbar for cracks and condition.

1. Make sure the towbar and vehicle (Lektro Tug tow) is attached correctly to the aircraft.

2. Do not operate the engine(s) during towing operations.

3. Always tow the aircraft at a walking speed and avoid quick stops and starts.

4. Always have someone walking at each wing tip and tail section to prevent a possible collision. Keep visual or communication contact between the walking crew members(s) and the brake/vehicle operator(s).

5. Do not turn the nose gear beyond the black 90° turn limit decals on the nose gear while towing. Turning beyond 90° can damage nose gear turning stop.

6. Replace the turn limit decals if chipped, worn or deteriorated. Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog—Chapter 11*.

7. Never let anyone enter or leave the aircraft (or ride on the external portions of the aircraft) while it is moving.

8. Remove the tail stand before winching the aircraft on the lift platform or towing.

9. Remove the chocks before winching the aircraft on the lift platform or towing.

10. Disconnect the grounding cable before towing.

11. Raise the main entrance door out of the full extended position before towing, for adequate ground clearance during towing.

12. After towing, make sure the entrance door is fully extended before stepping on the door.

13. When disconnecting the towbar, do not let the towbar fall on the nose gear fork.

Towbar Draw Force

The towbar draw force for the Citation XL/XLS/XLS+ aircraft is 1,200 pounds under the following conditions:

- Ramp—Smooth concrete surface which is dry, clean and level within 2°.
- Aircraft Weight—Empty weight plus full fuel.
- Wind—Aircraft towed into 16-knot (gust 25 knots) wind.

The towbar draw force (1,200 pounds) represents the minimum amount of force necessary to start to move the aircraft with the conditions listed above. This towbar draw force can increase with different conditions, like rough surface or sod, an uphill/downhill grade and improperly serviced tires.

NOTES

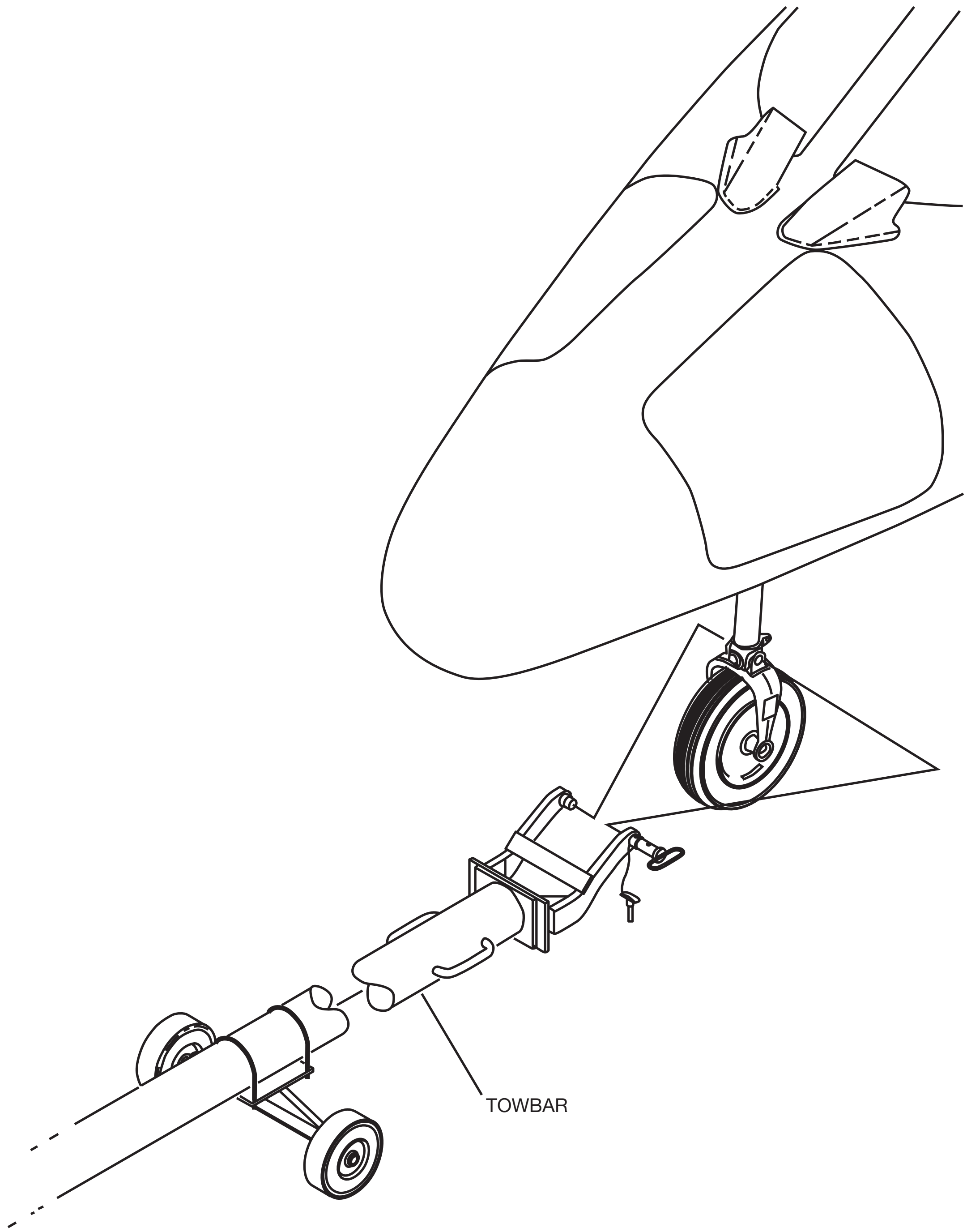

Figure 9-1. Towbar

NOSE GEAR TOWBAR TOWING

NOTE

It is permissible to disconnect the nose gear torque links. The nose wheel may then be turned beyond the black limit marks, eliminating the possibility of damage to the nose gear centering mechanism or steering stops.

Towing the nose gear with a towbar. Refer to Figures 9-1 and 9-2.

1. Attach the towbar to the upper fork buckets on nose landing gear.
2. Connect the towbar to the towing vehicle.
3. Station a person in the pilot seat.

NOTE

The aircraft may be towed without entering the aircraft if the parking brake is not set. Towing can be done with the control locks engaged. When towing the aircraft with control locks engaged to prevent unnecessary loads on the control system, limit the nose wheel turning angle to 60°. When extreme turning angles are necessary, release the control lock system.

4. Make sure that the wheel chocks, tailstand, static ground cables and mooring ropes are removed.
5. Disengage the parking brake.
6. If the area is congested, station wing and/or tail walkers to ensure adequate clearance between aircraft and adjacent equipment or structures.

CAUTION

Do not turn the nose landing gear wheel more than 90° from centered position in either direction. Damage to the turning stop results if the torque links are connected.

7. Tow the aircraft. Make smooth starts and stops with the tow vehicle. Refer to Table 9-1 for turn limitations.
8. After the towing operation is complete, do the following:
   a. Engage the parking brake.
   b. Lock the controls.
   c. Chock the wheels.
   d. Connect the static ground cables.
   e. Remove the towbar from the aircraft.
   f. Connect the nose gear torque links if they were disconnected.

Table 9-1. TURN LIMITATIONS

CONDITION	TURN LIMITATION
Torque links connected	90° (If the control lock is not engaged)
Torque links connected	60° (If the control lock is engaged)
Torque links disconnected	The nose wheel can be turned more than the 90° limit if stated on the placard. The control lock can be engaged or disengaged.

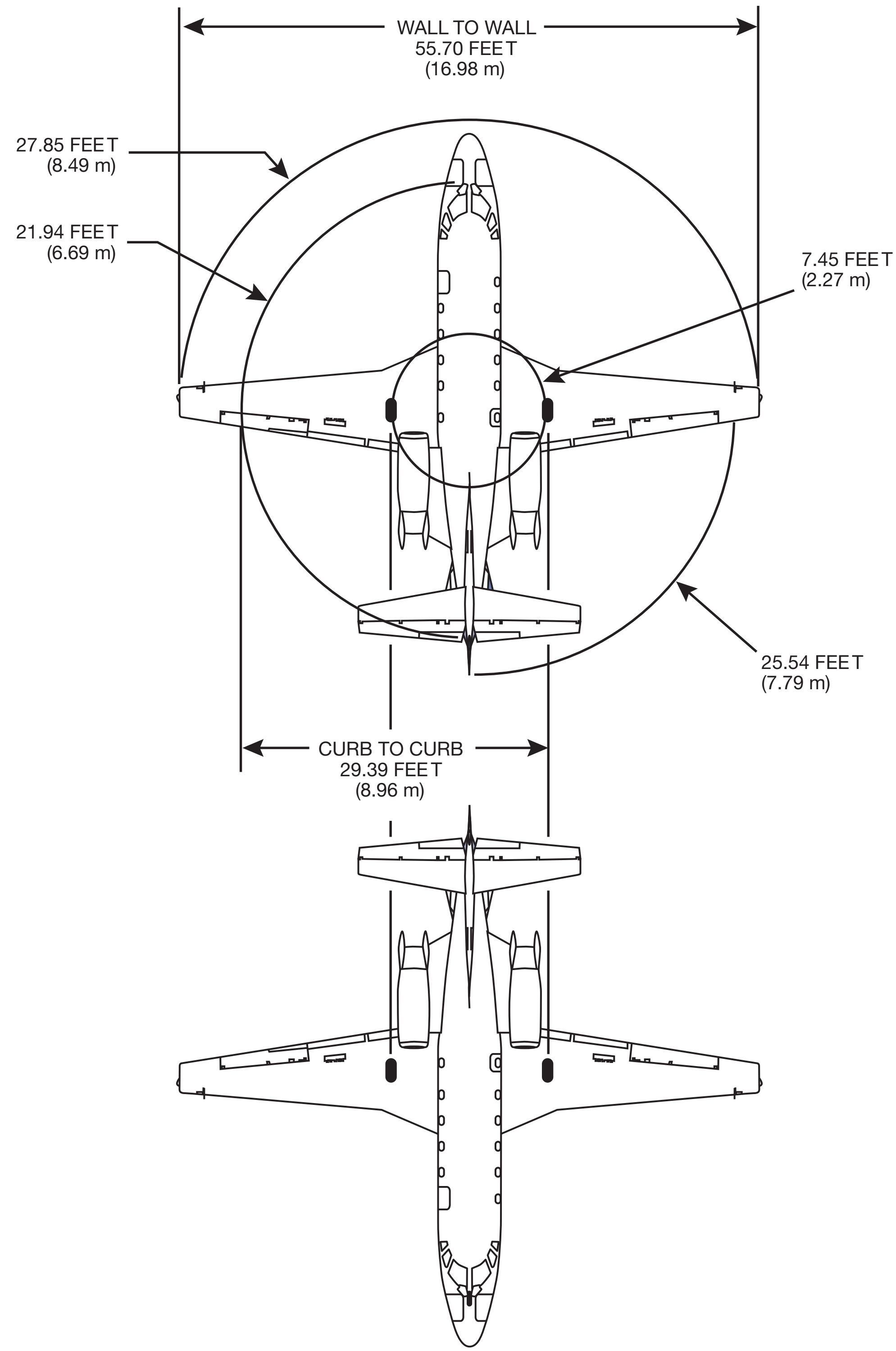

Figure 9-2. Towbar Turning Distance

NOSE GEAR ELECTRIC TOWING VEHICLE (WITHOUT A TOWBAR)

Towing With Lektro Tow Vehicle

1. Wrap the winch strap around the nose gear strut just above the fork.
2. Station a qualified person in the pilot seat.
3. Make sure that the wheel chocks, tailstand, static ground cables and mooring ropes are removed.
4. Disengage parking brake.
5. Winch the nose gear on the tow vehicle lift platform. Attach the nose gear to the vehicle. Follow the procedures for the specific Lektro tow vehicle. Refer to *Lektro Operations, Service and Parts Manual*.
6. If the area is congested, station wing and/or tail walkers to ensure adequate clearance between aircraft and adjacent equipment or structures.

CAUTION

Do not turn the nose landing gear wheel more than the limits specified in Table 9-1. Damage to the nose landing gear structure results if the torque links are connected and the aircraft is turned more than the 60° turn limit.

NOTE

It is not recommended to tow the aircraft with the torque links connected.

7. Tow the aircraft. Make smooth starts and stops with the towing vehicle. See Table 9-1 for turn limitations.
8. When the towing operation is complete, center the nose wheel and remove the nose wheel from the lift platform.
9. Engage the parking brake.
10. Engage the control lock.
11. Install chocks around the wheels.
12. Connect the static ground cables.
13. Connect the nose gear torque links if disconnected.

MAIN GEAR TOWING

NOTE

This procedure is done only in an emergency situation, such as an off-runway incident or when the aircraft must be pulled out of water or mud.

Tow the Aircraft with the Main Landing Gear

1. Station a qualified person in the pilot seat.
2. Install towing adapters on the main gear.

CAUTION

Use care to prevent damage to the wiring, brake plumbing or linkage rods in the area.

CAUTION

Do not wrap cables around the main gear. Use towing adapters when you attaching tow cables to main gear trunnions.

3. Attach the two cables to the towing adapters and the towing vehicle. Make sure that the cables are long enough to clear aircraft, and that the towing vehicle is on a hard surface.
4. Disengage the parking brakes and control lock.

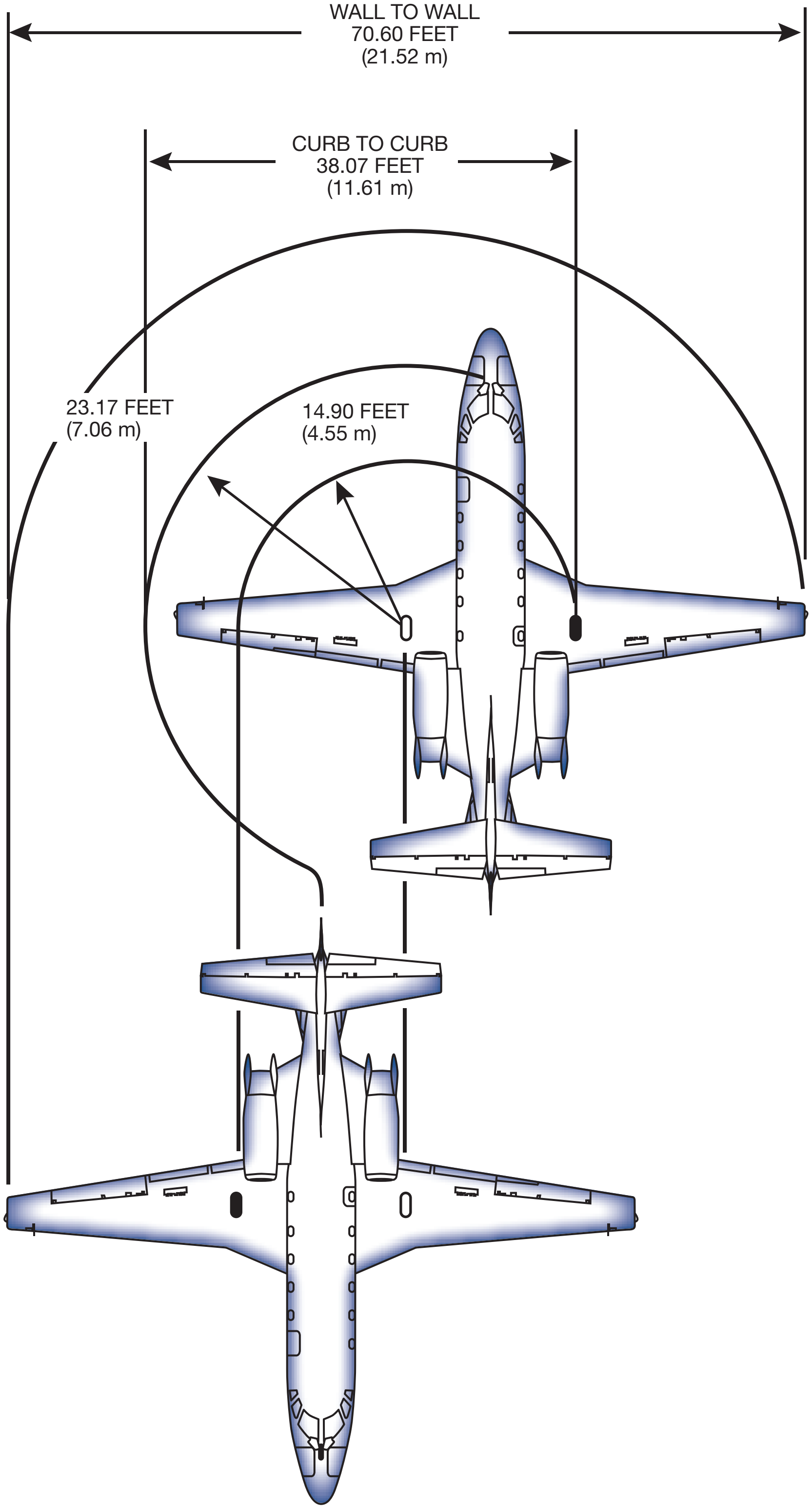

Figure 9-3. Taxi Turning Limits

5. Steer the aircraft with the rudder pedals. Use the aircraft brakes with smooth and even pressure.

6. When towing is complete, do the following:
   a. Center the nose wheel.
   b. Engage the control lock.
   c. Set the parking brakes.
   d. Chock the aircraft wheels.
   e. Connect the static ground cables.
   f. Connect the mooring cables.
   g. Disconnect the tow cables and remove the towing adapters.

TAXIING

Description

Taxiing the aircraft for ground movement is more desirable than towing when great distances are involved or when moving to a remote engine run-up area for engine test/adjustments.

Safety Precautions for Taxiing

1. Personnel involved with taxiing a Citation XL/XLS/XLS+ must be familiar with the aircraft and limits of turning. (Figure 9-3).

2. Ensure that the hydraulic system and brakes are in proper working condition.

3. Use only the required engine thrust to begin roll and approximate taxi speed. Do not use the brakes continuously to maintain desired speed. Adjust the engines accordingly.

4. Clear the taxi route of all obstructions such as maintenance stands, vehicles, etc.

5. In congested areas, station observer(s) to ensure wing tip clearance. Wing tip observer(s) must maintain visual contact with taxi operator at all times and must be familiar with taxi and parking signals.

6. On slick or icy surfaces, when nose wheel steering does not respond, do not permit the nose wheel to become cocked left or right. Damage to the nose gear results if the aircraft hits a dry area with the wheel cocked.

The aircraft can be taxied on hard surfaces, gravel or sod taxiways, and runways. The aircraft has a nose wheel steering system. When taxiing, rudder pedal movement operates the nose steering system.

CAUTION

When taxiing with a flat tire, do not use more engine thrust than needed. Monitor the inter-turbine temperature (ITT) indicator for possible engine overtemperature.

Taxiing the aircraft with a flat tire is not recommended. However, under emergency conditions, the aircraft may be taxied a short distance to clear the active runway or taxiway.

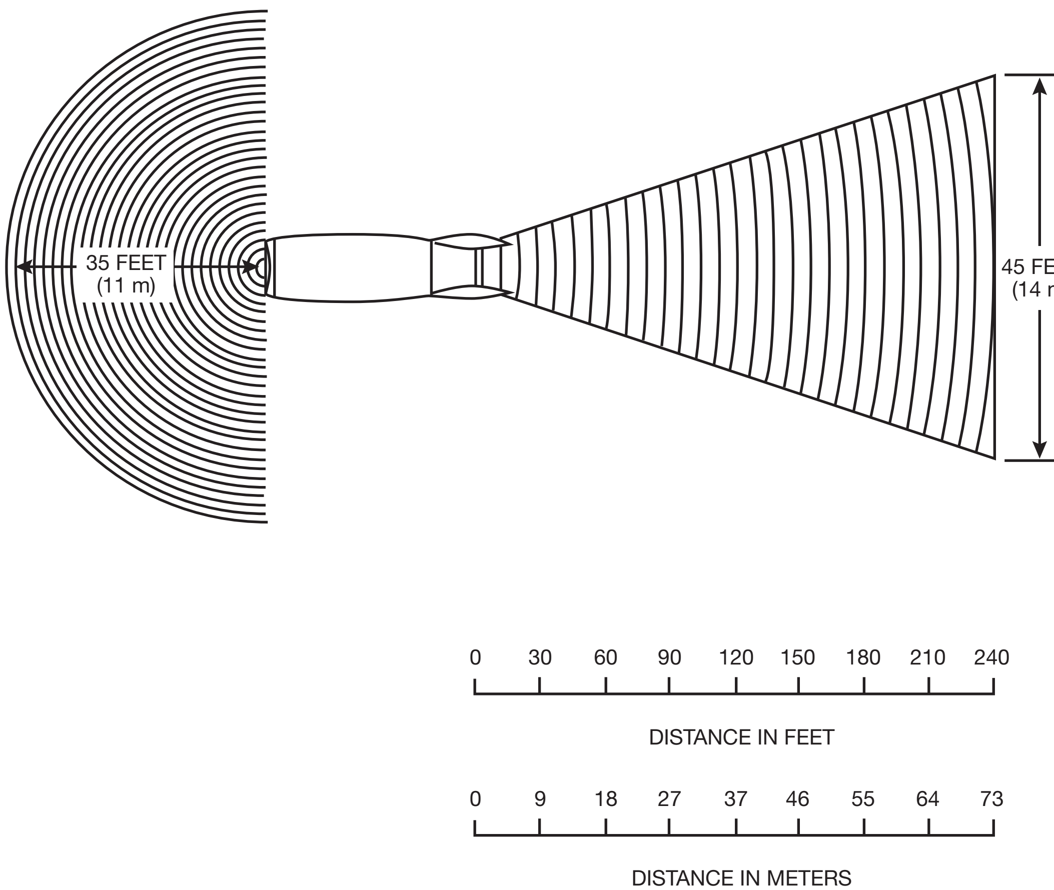

Figure 9-4. Engine Hazard Area

Preliminary Procedures

Refer to Figure 9-4.

1. Clear the following away from the area around the aircraft: maintenance stands, removed cowling, and other articles that might be damaged from engine exhaust blast.

2. Check the main gear wheels and remove static ground cables.

3. Ensure that fuel in the left and right wing tanks is balanced within 600 pounds.

4. Close all access and baggage doors.

Taxiing Procedure

1. Station two qualified persons in the flight compartment, one in the pilot seat to maneuver the aircraft, and one in the copilot seat to assist and act as an observer.

2. Engage the parking brake.

WARNING

Ensure that personnel and equipment are clear of engine inlet and exhaust.

3. Start the engines. Refer to the *FAA Approved Aircraft Flight Manual*. Verify that the antiskid system is OFF.

4. Remove the wheel chocks and release the parking brakes.

5. Begin taxi roll, applying only enough thrust to start roll. Roll forward before making a turn.

6. When braking is required during taxi, it is very important to use the brakes intermittently rather than dragging the brakes continuously. Let the aircraft accelerate and brake down speed to an acceptable taxi level rather than applying constant use of the brakes to maintain the desired taxi speed.

7. Steer the aircraft using the rudder pedals. Nose wheel steering is operated by the rudder pedals.

8. Use wing walkers to clear congested areas.

9. Taxi the aircraft to desired area. On the last wheel roll, ensure that the nose wheel is straight forward.

CAUTION

Do not set the parking brake while brakes are hot, since irregular friction surface mix transfer can result in brake chatter, noise and vibration.

10. Apply the parking brake.

11. Shut down the engines. Refer to the *FAA Approved Aircraft Flight Manual*.

12. Chock the main gear wheels. If aircraft is to be parked or moored, refer to Chapter 10—“Parking” or “Mooring.”

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CHAPTER 10
PARKING AND MOORING

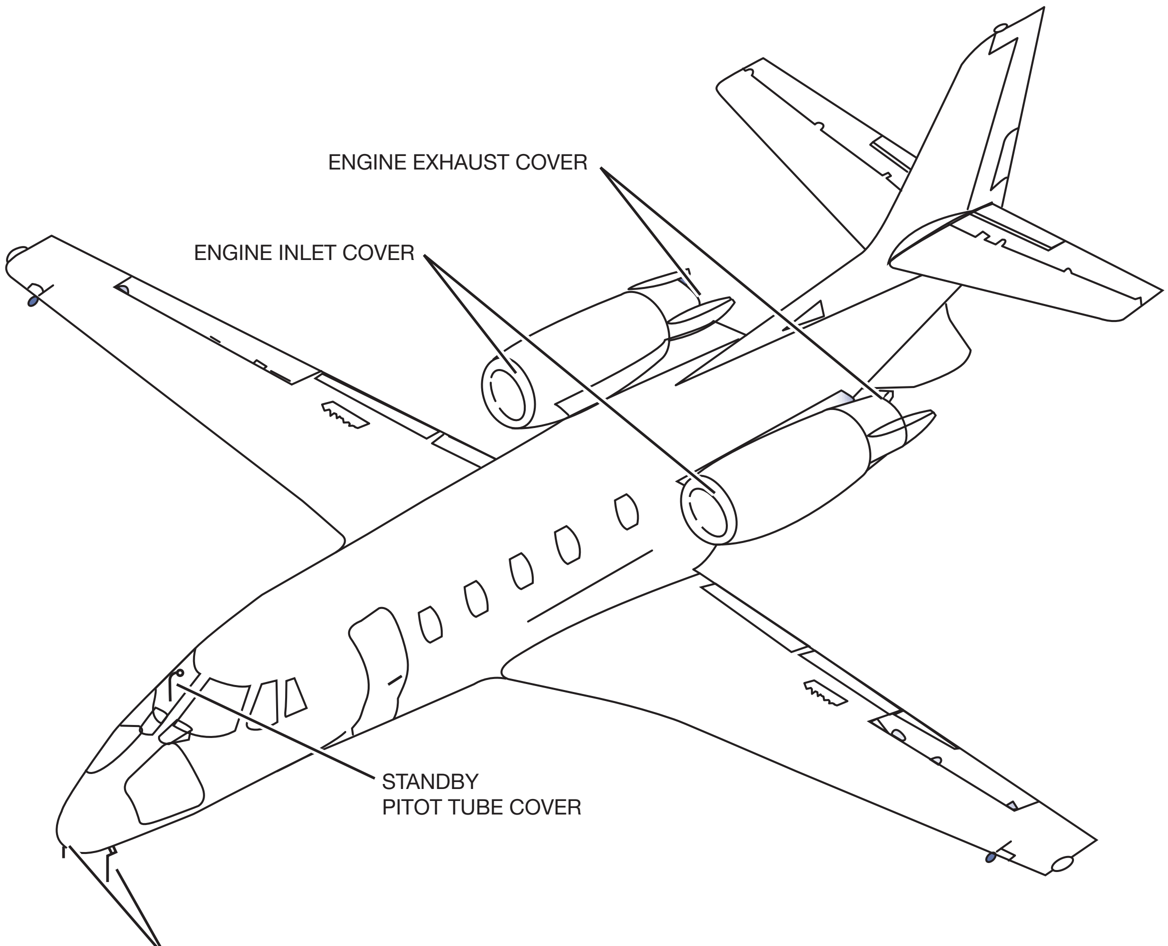


Figure 10-1. Engine Cover Installation

PARKING, MOORING, STORAGE AND RETURN TO SERVICE

DESCRIPTION

This chapter provides maintenance instructions for parking and mooring the aircraft on aprons when necessary mooring accommodations are available (Figure 10-1). No instructions are provided for parking or mooring on surfaces other than prepared parking aprons.

This chapter is divided into sections to aid maintenance personnel in locating information. A brief description of the section follows:

- The parking section describes procedures to secure the aircraft during normal weather conditions and short periods of time.
- The mooring section describes procedures used when servicing the aircraft when adverse weather conditions are present and anticipated for long periods of time.
- The storage section provides the recommended procedures for storing the aircraft. Recommendations vary with the length of time the aircraft is stored.

NOTES

Parking

Aircraft parking procedures are similar to those for other aircraft having tricycle landing gear. The wheels are chocked, parking brake and control lock engaged, and the aircraft ground cables attached. Under normal weather conditions, the aircraft may be parked and headed in a direction that will facilitate servicing without regard to prevailing winds.

Parking procedures are generally used during good weather conditions. If bad weather conditions exist or are expected, the aircraft must be moored.

General Procedures

1. Position aircraft on level surface

CAUTION

Do not set the parking brake while brakes are hot since irregular friction surface mix transfer can result in brake chatter, noise, and vibration.

2. Set parking brake and control lock.
3. Chock main gear wheels.
4. Connect static ground cable to aircraft.
5. Install protective covers as determined by expected weather conditions. The covers are stored in the tailcone baggage compartment.
6. Close foul weather window and doors as necessary.

Mooring

Mooring aircraft to the parking apron is accomplished by tying down at main gear and nose gear. With aircraft headed into wind, tie down using hemp rope or equivalent around gear. Mooring procedures are used during extended parking and expected adverse surface wind.

CAUTION

Make certain rope does not contact sharp edges and will not damage equipment.

General Procedures

1. Park aircraft on level surface.

CAUTION

Do not set the parking brake while brakes are hot since irregular friction surface mix transfer can result in brake chatter, noise, and vibration.

2. Set parking brake and engage the surface control gust locks.

NOTE

Do not set parking brakes for extended parking.

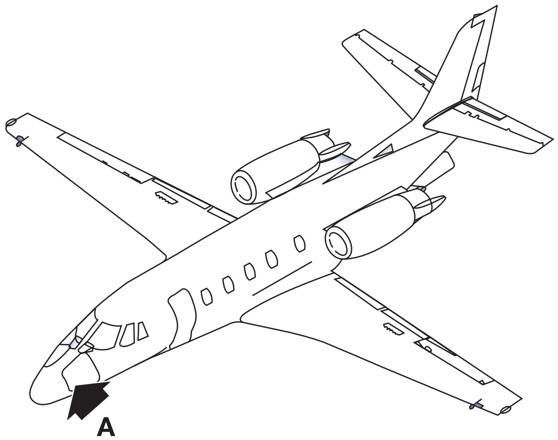
3. Chock main wheels and secure forward and aft chocks together.
4. Connect static ground cable.
5. Install protective covers (refer to Parking—Maintenance Practices).
6. Attach ropes to main landing gear and nose landing gear and secure to parking apron.

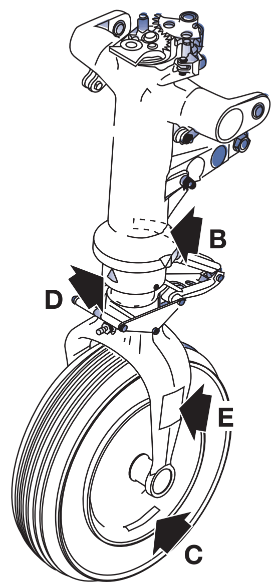
NOTE

Do not wrap rope around hydraulic lines or electrical wiring when securing the gear strut.

7. Close foul weather window and doors as necessary.

CHAPTER 11
REQUIRED PLACARDS



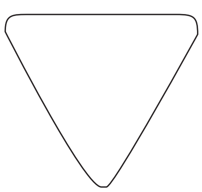


DETAIL A

TOWING WARNING

DO NOT TURN STRUT PAST 90° WHILE TOWING (BLACK POINTERS)

▲ 12
DETAIL E
ON NOSE LANDING GEAR FORK



13
DETAIL D
ON NOSE LANDING GEAR BARREL

▲ **MARKINGS REQUIRED BY GOVERMENT REGULATIONS**

FOR NOSE GEAR SERVICING REFER TO EXCEL MAINTENANCE MANUAL

15
DETAIL B
ON NOSE LANDING GEAR TRUNNION

INFLATE TIRE TO 130 ± 5 PSIG (UNLOADED)

▲ 14
DETAIL C
AIRPLANES -5001 THRU -5292
ON NOSE WHEEL BELOW VALVE STEM

INFLATE TIRE TO
130 ± PSIG (UNLOADED)
135 ± 5 PSIG (LOADED)

193
▲ **DETAIL C**
AIRPLANES -5293 AND ON
ON NOSE WHEEL BELOW VALVE STEM

Figure 11-1. U.S. Exterior Placards and Markings—Nose Landing Gear

PLACARDS AND MARKINGS

INTERIOR AND EXTERIOR PLACARD AND DECAL INSPECTION

This section describes inspection of the interior and exterior placards.

NOTE

This inspection is intended to be an overall inspection of all placards, decals, and markings on the aircraft.

INSPECT PLACARDS, DECALS AND MARKINGS

1. Examine the interior of the aircraft, including the nose and aft baggage areas, for the presence of all required placards, decals and markings.
   a. For required placards, decals and markings, refer to the *Citation XL/XLS/XLS+, Illustrated Parts Catalog.*
2. Examine the exterior of the aircraft, including the nose and aft baggage areas, for the presence of all required placards, decals and markings (Figure 11-1 and Table 11-1).
   a. For required placards, decals and markings, refer to the *Citation XL/XLS/XLS+, Illustrated Parts Catalog.*
3. Examine the aircraft identification plate.
   a. The ID plate is found on the forward post of the cabin entry doorway opening (Zone 251) or on the aft empennage (Zone 321) Refer to the *Citation XL/XLS/XLS+, Illustrated Parts Catalog* and Chapter 6—"Aircraft Zoning."

Table 11-1. U.S. EXTERIOR PLACARDS AND MARKINGS—NOSE LANDING GEAR

PART ITEM NUMBER	NOMENCLATURE 1 2 3 4 5 6 7	EFFECTICITY FROM	TO	UNITS PER ASSY
	U.S. exterior placard and markings - nose landing gear			
1 6640002-2	• Placard servicing			02
2 6640002-9	• Placard tire inflation FSO 6640002-13	5001	5292	NP R
3 6640002-13	• Placard tire inflation	5293	& ON	02
4 6640002-3	• Placard tow indicating			02
5 664002-1	• Placard tow warning			02

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CHAPTER 12
SERVICING

REPLENISHING

This section provides servicing information for replenishing aircraft fluid and gaseous systems, including capacity of the various systems. Applicable chapters throughout the *AMM* apply to this chapter for servicing procedures on entire systems, assemblies or components.

Replenishing tables in the *AMM* provide data applicable to the Citation XL/XLS/XLS+. Tables identify the system, system capacity, and type of fluids/gases suitable for the system.

FUEL AND ENGINE OIL

Description

This section provides maintenance personnel with servicing procedures on the aircraft fuel system and the engine oil system. It is subdivided into the fuel system and the engine oil system.

The fuel system servicing procedures include:

- Adding fuel
- Mixing anti-icing additives to the fuel
- Checking anti-icing concentration in fuel tanks
- Defueling procedures
- Purging fuel storage areas

The engine oil system servicing procedures provide information on:

- Adding oil to the engine
- Draining oil from the engine
- Descriptive information on synthetic turbine engine oil

Fuel Capacities and Additives

The wing fuel tank capacities and acceptable fuel specifications are shown in the fuel replenishment chart, in “Replenishing” of the *AMM*.

Biocidal Fuel Additive—Fuel requiring additives.

The additive has a biocidal chemical which inhibits growth of fungal and bacterial organisms in fuel storage reservoirs.

For instructions on how to mix biocidal fuel additives, contact the manufacturer of the additive and only use as directed by the manufacturer.

For the most current approved manufacturer of the biocidal fuel additive product, contact Cessna Citation Support at 1-877 483-2695 or FAX 1-316 517-8500.

Safety Precautions

1. Ground the fueling/defueling equipment (vehicle or fuel hydrant equipment) to the aircraft with designated grounding cable(s).

2. Make sure fueling/defueling equipment is grounded to an approved static ground.

3. Ground the aircraft to an approved static ground with the grounding cable.

4. Ground the fuel nozzle to appropriate ground near the fuel filler.

5. Ground aircraft as follows:

   a. Ground the aircraft first.

   b. Ground the vehicle (or hose cart) to the same ground as the aircraft.

   c. Bond the vehicle (or hose cart) to the aircraft.

   d. Bond the refuel nozzle to the aircraft

   e. Make sure that the fire-fighting equipment is set and made available.

6. Do not wear clothing that generates static electricity, such as nylon or synthetic fabrics.

7. Do not wear shoes made with metal taps or toes.

8. Make sure the aircraft is in a fuel loading/unloading area.

9. Do not operate high-wattage, pulse transmitting avionics equipment near the fueling/defueling operation. Turn on only the power needed to fuel the aircraft.

Maintenance Precautions

1. Use designated equipment for fuel loading or unloading to prevent contamination.

2. If the fuel and anti-icing additive are not blended correctly, deterioration of the integral fuel tank's interior finish results, which promotes corrosion.

3. Proper anti-ice additive blending procedures must be followed. Manufacturer instructions must also be followed.

4. Use only authorized types of fuel and anti-ice additive.

NOTE

While defueling, make sure that anti-ice additive blended fuel and unblended fuel are not mixed together.

FUEL LOADING

CAUTION

Make sure the correct grade and type of fuel is used. Refer to the approved aircraft flight manual for a list of approved fuels.

Approved fuels for the Citation XL/XLS/XLS+ aircraft do, or do not contain an anti-ice additive. The additive has a biocidal chemical that prevents growth of fungal and bacterial organisms in fuel storage reservoirs.

If fuel reservoirs become contaminated with fungi or bacteria, refer to Chapter 28—"Fuel Contamination."

Mixing anti-ice additive with fuel during refueling involves utilization of an aerosol or proportioner dispenser. Refer to "Tools and Equipment" in the *AMM*.

When fuel and anti-ice additive are mixed, a concentration test must be performed with the HB–P–C B/2 Anti-Ice Concentration test kit. To test the anti-ice additive concentration refer to the instructions provided with the kit.

Overwing Tank Filling Procedures

WARNING

Observe all safety and maintenance precautions when handling fuel.

WARNING

Perform fuel loading in areas where free movement of fire equipment is permitted.

WARNING

Make sure that the fuel supply unit is grounded and ground to the aircraft is connected.

1. Connect the fueling nozzle ground to the aircraft grounding receptacle, on the lower side of the wing outboard of the filler cap.

2. Place a protective pad on the wing by the fuel filler and remove the filler cap.

NOTE

Due to the position of the key holes, lock freezing may be encountered on aircraft with locking-type filler caps. Heating the key before inserting it into the lock normally thaws the lock. Jet fuel, anti-ice spray, or liquid can also be injected into the lock during inclement weather to reduce the possibility of freezing.

3. Fill the aircraft wing tanks.

4. Remove the fuel nozzle and protective pad.

CAUTION

Make sure that the filler cap is installed.

5. Disconnect the fueling nozzle ground and install the fuel filler cap.

Single-Point Pressure Refueling

CAUTION

Make sure the correct grade and type of fuel is used to service the aircraft. Refer to the approved aircraft flight manual for a list of approved fuels.

The single-point refueling control panel is on the right side of the fuselage fairing, forward of the wing leading edge. The control panel consists of the refuel/defuel adapter (receptacle) and a refueling precheck panel. For access to the refueling control panel, open the control panel access door.

WARNING

Obey all safety and maintenance precautions when handling fuel.

Single-Point Refuel Procedure:

NOTE

Single-point fuel pressure must not exceed 55 psi maximum.

1. Make sure that fire-fighting equipment is ready and available.
2. Open the single-point refueling control panel access door.
3. Prepare the aircraft for refueling by correctly grounding the aircraft and refueling vehicle/equipment together with an approved static grounding source. Refer to the "Safety and Maintenance Precautions" in this section.
4. Make sure that the aircraft fuel vents are not obstructed.
5. Remove the adapter cap.
6. Put the refueling nozzle into the receptacle, and turn clockwise to latch in place.
7. Open the nozzle.

CAUTION

Do a refueling precheck before each single-point refueling.

8. Start the fuel flow and do a system precheck to make sure that the pilot valves and/or fuel shutoff valves are operating properly.
   a. On the precheck panel, open the left and right precheck valves. Within 10 seconds, the refueling operation should shut down as indicated by the refueling equipment flowmeter or the flow totalizer.

   NOTE

   Each high level pilot valve needs a maximum of 3 GPM for precheck. Therefore, fuel flow rate during precheck must be 6 GPM for the left and right wing tanks.

   b. If refueling does not stop, stop the refueling operation and correct the malfunction. Refer to Chapter 28—"Single-Point Refueling/Defueling System" in the *AMM*.
   c. Close the precheck valves and continue the refueling operation.
9. When the aircraft fuel reservoirs are full, the high level pilot valves cause the fuel shutoff valves to close and fuel flow is stopped automatically.

NOTE

Fuel flow stoppage is indicated when the pumping equipment flowmeter or flow totalizer indicates no fuel flow.

10. Stop the pumping equipment (vehicle or hydrant equipment).
11. Make sure the aircraft fuel reservoirs are full. Look at the fuel quantity indicators.
12. Disconnect the refueling nozzle from the adapter (receptacle), and install the adapter cap.
13. Close and attach the single-point refueling control panel access door.
14. Remove all grounding cables.
15. Move the aircraft or refueling vehicle from the area.

Fuel Check in Wing Tank

The main function of the poppet-type drain valves on the lower side of the fuel tank is to sample fuel and to check for and drain sediment from the tanks. The valves are by the fuel tank sump area.

The poppet-type valve is a spring-loaded poppet housed in the drain valve body. The poppet is spring-loaded in the closed position.

A cross in the end of the poppet allows for screwdriver operation. To open the valve, depress the cross end and rotate it, to lock the valve in the open position. To close the valve, push the cross end, turn the lock, and release the screwdriver from the cross end, to seat the valve in the closed position.

During cold weather, if more than one hour elapses between removal from a heated shelter and takeoff, all fuel sumps must be drained through the drain valves during the preflight inspection. Enough fuel must be drained from each drain point to ensure that the fuel is free from water and/or other contaminants. At least 30 minutes must elapse between fueling and checking for contamination. The fuel should be drained into a suitable clear, clean container to allow a careful visual examination for water and other contaminants. To help distinguish water from fuel, add one or two drops of water-soluble food coloring in the container before draining fuel samples. The food coloring will mix easily with the water but not with the fuel.

For procedures to take a fuel sample, refer to Chapter 28—"Fuel Contamination" in the *AMM*.

Defueling

WARNING

Obey all safety and maintenance precautions when handling fuel.

WARNING

Before defueling an aircraft (for any maintenance checks) drain samples from the fuel sumps and examine the fuel for any obvious contaminates (i.e., water, discoloration, sediment, etc.). Do not refuel any aircraft with contaminated fuel. Do not mix contaminated fuel with any fuel supplies that might be used for aircraft refueling.

Single-point defueling of the aircraft is the recommended method, which must be used whenever possible. In the event single-point refueling/defueling equipment is not available, or a system malfunction prevents single-point defueling, an alternate defueling method can be utilized.

Defuel—Single-Point Method:

1. Access the refuel/defuel control panel by opening the access door on the right side of the fuselage fairing, forward of the wing leading edge.

2. Make sure that the aircraft and defueling equipment are properly grounded together, and to an approved static ground.

3. Remove the adapter (receptacle) cover.

4. Insert the nozzle into the receptacle; turn clockwise and latch in place. Open the nozzle.

NOTE

Each wing fuel tank has a defuel shutoff valve. The defuel shutoff valves are connected to manual defuel select shutoff valves, which can be used to deactivate defueling of either wing tank during the defueling operation.

5. To close the defuel shutoff valve (at the left or right wing tank) open the access door on lower fairing panel and pull the handle(s) on the manual defuel select shutoff valves. Pull outward to the extended/ horizontal position.

6. Start the defueling equipment and monitor operation.

7. When the tank(s) are empty, stop the defueling equipment and remove the nozzle from the receptacle.

8. Close the manual defuel select shutoff valve access door.

9. Install the receptacle cap and close the refuel/defuel control panel access door.

10. Remove the grounding cables and remove the aircraft or vehicle from the area.

Defuel—Force Method:

1. Remove the lower engine cowl. Refer to Chapter 71—"Engine Cowling."

2. Disconnect the fuel supply line at the rigid tube assembly. Refer to Chapter 73—"Engine Fuel Distribution."

3. Attach a suction line. Select one of the following:

   a. Attach the suction line (from fueling/defueling unit) to fuel supply line.

   b. Place a large container (five-gallon can) below the engine, and attach one end of a line to the fuel supply line and place the other end into the container.

   c. Put the suction line (from fueling/defueling unit) into the container.

NOTE

If both wing tanks are to be defueled at the same time, make sure there is enough capacity to contain the fuel.

4. Apply external electrical power to the aircraft and operate the fuel boost pump. Refer to Chapter 28—"Fuel Distribution."

5. Operate the suction line pump (in the defueling unit) and aircraft fuel boost pump until the wing tank(s) are empty.

CAUTION

To prevent possible damage to the fuel boost pump, do not operate the fuel boost pump after the low fuel pressure annunciator illuminates.

NOTE

Do not rely on the fuel boost pump sound as an indicator of cavitation, since sound varies with fuel depth. The fuel boost pump must be fully immersed in fuel during operation to make sure the pump has adequate cooling and lubrication.

6. Drain residual fuel from the tank using the wing tank poppet-type drain valves.
7. Remove the suction line from the defueling unit.
8. Install the lower cowl. Refer to Chapter 71—"Engine Cowling."
9. Remove the line attached to the fuel supply line.
10. Remove the line from the containers.
11. Connect the fuel supply line to rigid tube assembly. Refer to Chapter 73—"Engine Fuel Distribution."

Defuel—Suction Method:

1. Remove the fuel filler cap.
2. Insert a suction line from the defueling unit into the fuel filler opening.

NOTE

The suction line must have sufficient length to move the open end of the hose toward the fuel tank sump area.

3. When defueling flow stops, continue the defuel process with the forced defueling method.
   a. Transfer fuel from one wing to the opposite wing.
   b. Drain the remaining fuel through the poppet-type drains.

Transfer Method of Defueling (One Tank at a Time):

NOTE

Determine whether adequate space is available in the left or right fuel tank to accept the quantity of fuel to be transferred (defueled). The fuel is transferred through the crossfeed fuel system.

1. To defuel (transfer) fuel from the left tank to the right tank, perform the following:
   a. Connect external electrical power to the aircraft.
   b. With the FUEL BOOST L and R switches (SI006 right and SI007 left) to NORMAL, and both throttle levers in CUTOFF, put the CROSSFEED switch (SI004) to L TANK to R ENG.

NOTE

When the crossfeed valves open the left electric boost pump automatically activates.

CAUTION

To prevent possible damage to the fuel boost pump, do not operate the fuel boost pump after the low fuel pressure annunciator illuminates.

   c. Operate the left boost pump until the left engine low pressure light illuminates.

NOTE

Do not rely on the fuel boost pump sound as an indicator of cavitation, since sound varies with fuel depth. The fuel boost pump must be fully immersed in the fuel during operation to make sure the pump has adequate cooling and lubrication.

   d. Place the CROSSFEED switch (SI004) in the OFF position.

NOTE

Both crossfeed valves close within approximately 5 seconds and the boost pump shuts off automatically.

e. Put the FUEL BOOST L and R switches (SI006 right and SI007 left) to OFF.

f. Disconnect the external electrical power from aircraft.

g. Drain residual fuel from the left tank with the wing tank poppet-type drain valves.

2. To defuel (transfer) fuel from the right tank to the left tank, perform the steps listed above, but put the CROSSFEED switch (SI004) to R TANK to L ENG.

Purging

The following purging procedure is recommended when it is necessary to keep an aircraft in buildings not approved for fueled aircraft:

CAUTION

Fuel the aircraft, and purge air from the fuel lines to the engine, in order to return the aircraft to service. After this procedure, the fuel tanks are safe for 10 to 15 days.

1. Defuel the aircraft. Refer to "Defueling" in the *AMM*.
2. Drain the remaining fuel with the poppet-type drain valves and the filter drain.
3. Fill the aircraft fuel tanks with purging fluid MIL–PRF–38299 (JP-5 fuel may be used as an alternate purging fluid).
4. Let the purging fluid remain in the tanks for 15 minutes.
5. Defuel aircraft. Refer to "Defueling" in the *AMM*.
6. Move the aircraft to a hangar, if desired.

ENGINE OIL SYSTEM

Servicing the engine oil system consists of:

- Initial filling after engine installation
- Normal servicing (adding oil)
- Draining the system

Servicing the left and right engines is typical. The engine operates on oils that meet requirements. See Chapter 12—"Replenishing" in the *AMM*.

The oil tank/reservoir is an integral part of the intermediate case and is comprised of a main tank (on the right side of the engine), interconnected with a smaller auxiliary tank (on the left side) by a tank to tank cored passage. Both tanks are equipped with sight glasses. A drain cover is fitted on the main oil tank to permit drainage for both tanks. An oil filler cap is provided for oil service on the outboard side of each engine. The oil filler cap is accessed through the oil access door on the lower engine cowling.

Some synthetic oils may change color after a few hours time. The color change is not harmful unless accompanied by oil sludge formation and viscosity or acidity increase.

The maximum oil consumption rate specified for the engine is 0.2 pounds per hour (0.000025 kg/sec) measured over a 10-hour period. Engine maintenance is necessary when the oil consumption rate is surpassed. The oil consumption limit allows for some expected increase in service, due to possible seal deterioration, high altitude operating environment, etc. Specific oil brands have minor density variations.

The approximate weight of turbine engine oil is:

- One quart (0.94 Liter)= 1.9 pounds (0.86 kilogram)
- One Imperial quart = 2.3 pounds

This means that in order to exceed the field limit of 0.2 pounds per hour, oil consumption would be in excess of one U.S. quart every 9.5 hours or one imperial quart every 11.5 hours.

Safety and Maintenance Precautions

Safety Precautions:

1. Wash hands/skin thoroughly after exposure to jet engine oil, to prevent skin irritation.
2. Clean up oil spills on the floor/ramp area.

Handling Precautions for used oil:

WARNING

Hot oil can cause severe burns. Wear protective gloves and clothing if service is required on a hot engine.

It is essential that the precautions be practiced to minimize the amount of skin exposed and the length of time that used oil stays on the skin.

WARNING

Thoroughly wash used oil off skin as soon as possible with soap and water. A waterless hand cleaner can be used when soap and water are not available. Do not use kerosene, thinner, or solvents to remove used engine oil. Always apply skin cream after using waterless hand cleaner. Do not overuse waterless hand cleaners, soaps or detergents. They can remove the skin protective barrier oils.

WARNING

Do not put oily rags in pockets or tuck them under a belt. This can cause continuous skin contact. Wash oil-soaked clothing before wearing again. Discard oil-soaked shoes. Use gloves made from nitrile, neoprene, viton or other material that oil cannot penetrate. Do not use kerosene, thinners or solvents to remove used engine oil. They remove the skin's natural protective oils and can cause dryness, irritation and possibly more serious toxic effects.

CAUTION

Do not pour used engine oil on the ground or down drains and sewers. It is a violation of federal law (chapter 40 code of federal regulations section 110). The EPA encourages collection of used motor oil at collection points in compliance with appropriate state and local ordinances.

CAUTION

Always use specified type of oil to service engines. Clean oil spills on the engine, accessories, electrical wiring, and nacelle skin. Use proper oil servicing techniques/procedures. Do not mix oils with oils that do not meet the requirements listed in Chapter 12—"Replenishing" in the *AMM*.

Servicing the Oil Reservoir

Oil Servicing After Engine Installation or During Oil Change:

1. Open the oil access door. Refer to Chapter 71—“Engine Cowling.” in the *AMM*.
2. Remove reservoir oil filler cap.
3. Refer to the *Pratt and Whitney Canada PW545A Maintenance Manual* Chapter 72—“Engine, General Servicing”.
4. Install filler cap.
5. Start the engine. Refer to the *Aircraft Flight Manual (AFM)*.
6. Operate the engine at idle for 15 minutes.
7. Stop the engine.
8. Do a check on the oil level 10 minutes after engine shutdown.
9. Check reservoir sight glass oil level.

NOTE

If oil of the same brand (as tank contains) is unavailable, then other oils listed in the replenishment chart may be intermixed, when the total quantity added does not exceed 2 U.S. quarts in any 400 hour period.

10. Top off reservoir to the required level (if required).

NOTE

Do not overfill the oil tank/system.

11. If more than 2 quarts of dissimilar oil brands have been intermixed in any 400 hour period, drain and flush the oil system. Refer to the manufacturer’s *Engine Maintenance Manual*.
12. If unapproved brands of oil or oil of different viscosities are intermixed, drain and flush the oil system. Refer to the manufacturer’s *Engine Maintenance Manual*.
13. Install the filler cap in the filler neck.
14. Verify that the cap is correctly installed and locked.
15. Close oil access door.

Between/After Flight Oil Servicing:

NOTE

To reduce the possibility of over servicing and ensure accurate readings for oil consumption measurement, it is recommended that the oil level always be checked within 10 minutes after shutdown.

1. Open oil access door. Refer to Chapter 71—“Engine Cowling.” in the *AMM*.
2. Check reservoir sight glass oil level. Top off the reservoir to the required level (if needed).

NOTE

If the same brand of oil (as existing oil in the tank) is unavailable, then other oils listed in the replenishment chart may be intermixed, if the total quantity added does not exceed 2 quarts in any 400 hour period.

3. If more than 2 quarts of dissimilar oil brands have been intermixed in any 400 hour period, drain and flush the oil system. Refer to the manufacturer’s *Engine Maintenance Manual*.
4. If unapproved oil brands or oil of different viscosities are intermixed, drain and flush the oil system. Refer to the manufacturer’s *Engine Maintenance Manual*.
5. Close oil access door.

Draining/Filling Engine Oil System

Draining Engine Oil System:

NOTE

Accomplish oil draining as soon as practical after engine shutdown.

WARNING

Hot oil can cause severe burns. Wear protective gloves and clothing if service is required on a hot engine.

1. Open lower engine cowling. Refer to Chapter 71—“Engine Cowling.” in the *AMM*.
2. Open oil access door and remove the filler cap.
3. Position container(s) under the oil tank drain cover and oil filter drain cover.
4. Remove nuts and washers securing the oil tank drain cover.
5. Remove cover using puller (PWC66103).
6. Discard preformed packing (O-ring).
7. Remove nuts and washers securing oil filter drain cover. Remove cover using puller (PWC66103).
8. Discard preformed packing (O-ring).
9. Allow system to drain completely.

Oil System Filling:

1. Lubricate and install new preformed packing (O-ring) on oil tank drain cover.
2. Install oil tank drain cover using nuts and washers. Refer to the *Citation XL/XLS Illustrated Parts Catalog*.
3. Lubricate the threads on nuts and studs with anti-seize compound (PWC06–009).
4. Install nuts and washers.
5. Torque nuts per manufacturer specification. Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*.
6. Lubricate and install a new preformed packing (O-ring) on the oil filter drain cover.
7. Lubricate the threads of nuts and studs with anti-seize compound (PWC06-009).
8. Install oil filter drain cover using nuts and washers.
9. Torque nuts on studs. Refer to the Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*, Introduction—List of Vendor Publications.
10. Replace oil filter element if necessary. Refer to the Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*, Introduction—List of Vendor Publications.
11. Refill oil tank with engine oil.
12. Install filler cap in filler neck.
13. Verify cap is correctly installed and locked.
14. Start engine. Refer to the *Aircraft Flight Manual (AFM)*.
15. Operate at idle for 15 minutes.
16. Check for oil leaks.
17. Verify correct engine oil level. Refer to the Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*.
18. Close oil access door.
19. Close lower engine cowling. Refer to Chapter 71—“Engine Cowling” in the *AMM*.

5–12 AIRCRAFT GENERAL

ONBOARD AUXILIARY POWER UNIT

Description

This section describes the servicing of the auxiliary power unit (APU) oil system. The APU is in the tail cone. The oil sump is accessed by removing the APU tail cone access panels. Refer to Chapter 6—"Access Plates and Panels Identification" in the *AMM*.

The APU oil system provides pressurized and mist lubrication for all gears, shafts and bearings within the engine. Refer to the *Allied Signal Component Maintenance Manual* for oil servicing, oil filter replacement, draining and replenishment of oil.

Servicing the onboard APU oil system consists of periodic oil changes and normal between-oil-change servicing (adding oil). The engine operates on oils that conform to MIL–L–23699 specifications (the same oil utilized in the main engines). Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*.

The oil sump is under the reduction drive assembly. Filling/servicing the oil system is accomplished through the filler neck and cap. Draining the oil sump and oil system is accomplished by removing the drain plug on the bottom of the oil sump. Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*.

WARNING

Jet engine oil may cause skin irritation. Wash the skin thoroughly after any exposure to oil. To avoid personal injury, the proper personal protection must be worn when handling jet engine oil.

CAUTION

Use only approved jet engine oil for servicing the APU oil system. Do not mix oils that do not meet MIL-L-23699 specifications.

NOTE

For engine preservation and depreservation, refer to the *Citation XL/XLS Illustrated Parts Catalog*.

Oil Discoloration

Some synthetic oils may change color within a few hours of engine operation. The color change is not harmful unless it is accompanied by oil sludge formation and viscosity or acidity increase.

Safety and Maintenance Precautions

Safety Precautions:

1. Wash hands/skin thoroughly after exposure to jet engine oil to prevent skin irritation.

2. Clean up oil spills on floor/ramp area.

Maintenance Precautions:

Refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*.

Always use specified type of oil to service the APU.

Wipe up oil spills on engine, accessories, electrical wiring and fuselage area.

Never overfill the oil sump. An overfull oil sump results in oil foaming, low oil pressure and abnormal gear wear.

Do not mix oils meeting the specifications of MIL-L-23699 with other oils.

Servicing/Draining the Oil Sump

NOTE

Check the engine oil level ten minutes after shutdown. For between oil change servicing, refer to the *Citation XL/XLS/XLS+ Illustrated Parts Catalog*.

HYDRAULIC FLUID SYSTEMS

Description

This section is subdivided into the hydraulic power system and anti-skid brake system:

- The aircraft hydraulic power systems servicing procedures include servicing the system with a hydraulic power service unit and a portable service unit.
- The anti-skid brake system servicing procedures describes filling the brake reservoir.

Before performing any operation on the hydraulic system, personnel must read, thoroughly understand, and observe the following when working with hydraulic fluid.

WARNING

Observe the following safety precautions when working on systems containing phosphate ester base fluid. Long exposure to phosphate ester base fluids can cause skin dehydration and chapping.

1. Wash hands thoroughly with soap and water before starting work.
2. Apply panthoderm cream or equivalent (silicone hand cream) to hands, wrists and forearms at beginning of work period. Rub cream under fingernails and into creases of skin. Apply kerodex or equivalent frequently during work period. Reapply the panthoderm cream only after skin has been cleansed by washing.
3. Wear goggles when pressure testing components or systems, and any time there is possibility of fluid splashing into eyes.
4. If fluid splashes into eyes, treat eyes immediately by irrigating thoroughly with clear, cold water.
5. Wash hands, wrists and forearms with soap and hot water when there is contact with hydraulic fluid.
6. If clothing becomes soaked with fluid, remove it as soon as possible. Thoroughly wash the skin and put on clean clothing.

Before any maintenance is performed on the hydraulic system, personnel must read and thoroughly understand the following precautions. Careful adherence to these instructions aids in maintaining a functional and trouble-free system.

CAUTION

Observe the following technical precautions when working on the hydraulic systems. Phosphate ester fluids will attack a wide range of materials including rubber, copper, various plastics and paints and dyes in clothes.

CAUTION

Skydrol hydraulic fluid, when heated to approximately 270°F (132°C), decomposes into acids and other products that can cause damage to metal structure.

Ensure that the fluid does not come into contact with any part of aircraft outside of the hydraulic system. Keep spillage to an absolute minimum, place rags under fittings before disconnecting lines. Clean up spilled hydraulic fluid immediately to prevent entry into adjacent areas of the aircraft and to prevent future false hydraulic leak reports. If spillage occurs, wipe up the fluid with a dry cloth and wash area with naphtha, Federal Specification PD-680 (Type 1) or a high flash Stoddard solvent.

When lines are disconnected and/or components are removed, provide suitable protection to prevent foreign material from entering the lines or components by use of caps or covers.

When electrical connectors are disconnected, install caps or other suitable protectors to prevent entry of hydraulic fluid, moisture and foreign objects.

Always check the position and angle of all fittings removed from components to ensure placement and alignment on installation or replacement components.

When washing metal parts before assembly, use only naphtha, Federal Specification PD-680 (Type 1) or a high flash Stoddard solvent. Ensure that all traces of the solvent are removed before assembly.

Use only clean phosphate ester base fluid for flushing or testing hydraulic components.

Use only clean phosphate ester base fluid when filling reservoir.

Do not unpack packings and seals until they are required. Ensure that only approved rings and seals are used.

When assembling hydraulic system packings and seals, lubricate only with hydraulic fluid. Always lubricate packings and seals immediately before installation. Threaded fittings must be assembled without the use of lubricants whenever possible. If a lubricant is required to prevent galling or to otherwise ease installation, use hydraulic fluid.

CAUTION

Take special care to avoid contamination of packings after lubrication. Take care to prevent contamination of hydraulic fluid with other oils, water or dirt.

CAUTION

The aircraft hydraulic systems are designed for use with phosphate ester base hydraulic fluid. If a petroleum-based oil or solvent is introduced into the system or component, rapid deterioration of all seals, packings, O-rings takes place causing multiple leaks. This also requires the overhaul or replacement of all components containing such seals. Additionally, particles of deteriorated seals, packings and o-rings can be deposited within the orifices or valve from which they are dislodged, causing failure of the component during operation.

If a system becomes contaminated with any petroleum-based oil or solvent, drain the system, perform maintenance on the components and replace the seals involved. Then flush with clean phosphate ester base hydraulic fluid.

Discoloration from original color may be observed with some brands of phosphate ester base hydraulic fluids. Color change alone in hydraulic fluid has not been considered a significant criteria for evaluating fluid performance capability.

The clean hydraulic fluid used for flushing, testing and filling of a hydraulic system must meet the requirements of NAS 1638, Class 5. A NAS 1638, Class 5 fluid is defined as a fluid which contains a maximum of 1731 particulate contaminates greater than 15 microns (approximately 0.006 inch in diameter) in a 100 milliliters sample. Of these, a maximum of 50 particulates can be greater than 50 microns (approximately 0.017 inch in diameter).

HYDRAULIC POWER SYSTEM

There are two methods of servicing the hydraulic system reservoir. Although alternate equipment may be used, it is recommended that the servicing procedure is adapted to prevent damage to the hydraulic reservoir.

Personnel must be familiar with the safety and technical precautions of phosphate ester base hydraulic fluid.

Specific capacities and fluid specifications are shown in the hydraulic system replenishment chart in the *AMM*.

Hydraulic Reservoir Servicing

Servicing the reservoir with a hydraulic service unit:

CAUTION

Skydrol hydraulic fluid, when heated to approximately 270°F (132°C), decomposes into acids and other products which can damage the metal structure.

NOTE

Both hydraulic firewall shutoff valves must remain open during hydraulic ground service unit operation, to prevent excessive pump shaft seal-back pressure and to prevent excessive pressure build-up in the right pump suction/supply system.

1. Start the hydraulic service unit and adjust to the following settings.

CAUTION

Failure to adjust the hydraulic service unit to reduced settings results in damage to the hydraulic reservoir.

a. Adjust the GPM flow to 1 GPM or less.

b. Adjust the service unit relief pressure to 100 psi, ± 50 psi (689 kPa ± 345 kPa).

c. Close the outlet and return valves at the stand.

d. Shut down the service unit after adjustments have been made.

2. Open the access door and connect the hydraulic service unit hoses to the ground operations couplings.

CAUTION

Ensure that the aircraft ground suction source quick-connect fitting is securely connected to the service cart return line or hyd purifier inlet line fittings. Failure to do so could cause damage to the reservoir by overpressurizing it.

3. Start the hydraulic service unit and open the return valve at the hydraulic service unit.

4. Slowly open the hydraulic service unit outlet valve and cycle the hydraulic fluid in the system, to bleed air from the system and lines.

NOTE

If pressure is indicated on the hydraulic service unit gauge, one or more of the hydraulic system components have been actuated. Check to ensure that all hydraulic valves are in neutral, the speedbrakes are closed, the gear extended, flaps retracted, and thrust reversers are stowed. Cycle the fluid in the hydraulic system from 2 to 5 minutes, to properly bleed air from the system and lines.

5. After cycling the fluid in the system, close the outlet valve on the service unit. When the outlet valve has been closed, close the return valve.

WARNING

If the return valve is closed first and the hydraulic service unit is not adjusted to the lower settings (referred to earlier) the hydraulic reservoir ruptures.

6. With the hydraulic service unit return valve closed, slowly crack the outlet valve to fill the reservoir to the overfill position on the sight gauge.

NOTE

Fluid drains from the reservoir relief valve if the reservoir is overfilled.

7. Shut down the hydraulic service unit and remove the lines.
8. After reservoir has stabilized, recheck the reservoir. If overfill is indicated on the sight gauge, proceed as follows:
   a. Using the manual pressure relief valve, slowly drain fluid from reservoir.
   b. When full is indicated on the sight gauge, close the manual relief valve.
9. After the reservoir is properly serviced, disconnect the hydraulic service unit, replace the dust caps and close the access door.

Service the Hydraulic Reservoir with Portable Service Unit:

1. Ensure that all hydraulic valves are in neutral, that the speedbrakes closed, the gear extended, flaps retracted, and thrust reversers stowed.

CAUTION

When these conditions are not met, overservicing of the reservoir results. The system may be damaged if the reservoir is overserviced.

NOTE

Care must be taken not to introduce air into the system during these servicing procedures.

2. Open the hydraulic ground operations service panel access door and remove the overboard line attached to drain valve.
3. Place a drain pan under the ground operations service panel.
4. Connect the hand pump service hose to the drain valve, but do not tighten.
5. Momentarily crack the drain valve to release any air trapped in the drain line.

WARNING

Follow safety precautions. Do not allow hydraulic fluid to be sprayed into the face and eyes.

6. Operate the hand pump to flush the service line. Tighten the line when escaping fluid is clean and clear of air bubbles.
7. After the service line has been tightened, open the reservoir drain valve and pump fluid into the reservoir until it indicates overfill at the sight gauge.

NOTE

Fluid escapes from the relief valve drain tube if the reservoir is overfilled.

8. After the reservoir is properly serviced, close the drain valve and open the service line relief valve.

WARNING

Fluid in the service line is under pressure. If the service unit does not have a line relief valve, fluid sprays when loosened.

9. Remove the service line. Replace the overboard tube and close the access door.

BRAKE RESERVOIR

The hydraulic brake system uses a reservoir to hold a supply of hydraulic fluid. On SNs 5001 thru 5500, the brake reservoir is found inside the left nose bay door. On SNs 5501 and on, the brake reservoir is found inside the brake service door on the left side of the aircraft, next to the battery service door.

On Aircraft 5001 thru 5500, the brake reservoir has:

- A filler plug
- A top and bottom sight gauge
- A vent tube

On Aircraft 5501 and on, the brake reservoir has:

- A filler plug
- A single sight gauge
- A vent tube

Hydraulic Brake Reservoir Servicing

CAUTION

Do brake reservoir servicing with an approved hydraulic fluid as shown in the hydraulic replenishment chart. The fluid must also be clean in accordance with NAS 1638, Class 5 "Clean hydraulic fluid requirements". Refer to "Hydraulic Fluid System" in the *AMM*.

CAUTION

When the temperature of Skydrol hydraulic fluid is approximately 270°F (132°C) or higher, the fluid is broken down into acids and other products which can damage metal structures or surfaces.

Examine the Fluid Level In the Hydraulic Brake Reservoir:

1. On aircraft 5001 thru 5500, hydraulic fluid must be added when the fluid level is seen in the bottom sight gauge. If hydraulic fluid is not added when the fluid is at this level (or lower), operating the brake system can result in high brake fluid temperature and possible brake failure. Hydraulic fluid is usually added to the brake reservoir when the fluid level is below the top sight gauge. Operation of the brake system is permitted until the hydraulic fluid level is seen in the bottom sight gauge.

2. On aircraft 5501 and on, hydraulic fluid must be added when the fluid level is below the "add" marks on the right side of the sight gauge for the current temperature of the hydraulic fluid. If hydraulic fluid is not added when the fluid is at this level (or lower), operating the brake system can result in high brake fluid temperature and possible brake failure.

To Complete Servicing the Hydraulic Brake Reservoir:

1. On aircraft 5001 thru 5500, open the left nose bay door.

2. On aircraft 5501 and on, open the brake service door on the left side of the aircraft next to the batter service door.

3. Remove the safety wire on the filler plug.

4. Remove the filler plug.

CAUTION

Do not spill phosphate ester hydraulic fluid when servicing the brake reservoir. Damage to the avionics equipment can occur if the hydraulic fluid touches the equipment.

5. Use a hydraulic service hand pump that contains clean, (per NAS 1638, Class 5) approved hydraulic fluid to fill the brake reservoir to within 1/2-inch (12.7 mm) below the filler port.

NOTE

If too much fluid is put in the hydraulic brake reservoir, the unwanted fluid will be pushed out the vent tube during brake operation.

6. Install the filler plug.
7. Safety wire the filler plug. Refer to Chapter 20—"Safetying."
8. On aircraft 5001 thru 5500, close the left nose bay door.
9. On aircraft 5501 and on, close the brake service door.

PNEUMATIC SYSTEMS

This section is subdivided into: landing gear and brake pneumatic system, anti-skid brake accumulator and tires.

The aircraft landing gear and brake pneumatic system is an emergency system. One storage cylinder is utilized to supply high-pressure gas on demand to the landing gear pneumatic extend system and/or the pneumatic brake system.

The anti-skid brake accumulator utilizes high-pressure gas to precharge the accumulator.

The tires are included in the pneumatic systems because the same gas can be used to service the tires as used in other aircraft pneumatic systems.

Precautions

High-pressure gas, including air pressure, is dangerous when precautions are not exercised. A few of the basic precautions are listed below:

- Follow procedures. Short cuts may be dangerous.
- Maintain tools in good serviceable condition. Using wrong equipment or worn tools may be dangerous.
- When discharging pressure vessels, prevent the exhausting stream of gas from contacting the body in a direct line. Position the body on the opposite side or at the best possible angle from the escaping gas, including the hands.
- When the pressure vessel contains a liquid or gas that is harmful to inhale, discharge the vessel in an assigned area.
- Use protective equipment. This includes shields, cages, goggles or other equipment when specified.
- The escaping gas must not be directed toward equipment that could rotate or spin.

Pneumatic Service Valves – MS Type

Discharging Pressure:

1. Remove the service valve dust cap.
2. Turn 3/4-inch hex swivel nut counterclockwise to open valve.
   a. Release torque on swivel nut and turn past the free-play area until all play is taken up.
   b. Slowly turn the swivel nut an additional 1/4 turn, to allow pressure to escape.

WARNING

Both the swivel nut and body nut are 3/4-inch hex. Turning the body nut removes valve from service port, allowing valve to be blown out.

WARNING

Do not cover the valve with hands or allow pressure to be blown into face.

CAUTION

Do not open valve more than 1/2 turn. A pressure release that is too rapid causes frosting and possible blockage of valve stem. Refer to "Blocked/Damaged Pneumatic Service Valve" in the *AMM*.

3. When all system pressure has been released, remove the valve from the service port for replacement of seal or when further system maintenance is to be performed.

Charging (Filling) Pneumatic System:

1. Install servicing valve, torque and safety wire.
   a. When reinstalling old valve, replace the seal and check the seat for damage. Replace when seat is damaged.
   b. When a new valve is being installed, use the new seal provided and remove the dust cap.

CAUTION

Attempting to install and torque valve into the service port with a 3/4-inch hex swivel nut damages the valve seat and strips the threads on the valve stem.

2. Open the service valve by turning the swivel nut counterclockwise to between 1/2 to 3/4 turns.
3. Connect the service line hose to the charging stem of service valve.
4. Slowly charge the system according to the service placard.
5. Close the service valve by turning the swivel nut clockwise past the free-play area and tighten. Disconnect the service hose after tightening the swivel nut to ensure correct system charge is maintained.

CAUTION

Excessive torque damage the service valve seat and may strip the stem or swivel nut threads. Refer to "Removal of Damaged/Blocked Service Valve" in the *AMM*.

6. Replace the dust cap.

Removal of Blocked/Damaged Pneumatic Service Valves:

WARNING

Only experienced personnel should attempt to remove a service valve when system pressure cannot be released. All procedures for releasing the pressure must have failed before proceeding.

NOTE

Pressure can sometimes be released by cracking a system line. This is dangerous and must be the last resort.

1. Remove the safety wire from body of valve.

WARNING

Turning the valve body more than necessary puts excessive strain on valve and port threads. If the threads fail, the valve blows out.

2. Slowly turn 3/4-inch hex body nut 1 to 1-1/2 turns counterclockwise until an emergency pressure release notch in the threads of valve allows pressure to bypass the threads and escape between the seal and port surface.
3. Hold the valve in this position with a wrench until all system pressure has been released.

WARNING

Do not attempt to remove the valve until all pressure has been released.

4. When pressure has been released, slowly remove the valve, stopping at any indication of pressure release.
5. Install a new valve and destroy the damaged valve.

GEAR AND BRAKE PNEUMATIC SYSTEM

The gear and brake pneumatic bottle is inside the left nose bay door.

The pressure in the pneumatic bottle must be maintained at 2000 psi.

Servicing/Deflating Gear and Brake Pneumatics

Refer to “Pneumatic Systems” in the *AMM*, for operating instructions.

Service (Fill) Pneumatic Bottle:

1. Open the left nose bay door.
2. Return the emergency gear release handle to normal position.

NOTE

The handle is positioned to the discharge position if the pneumatic system was used to operate the landing gear. Moving the handle to normal position releases the trapped high-pressure air and vents the air overboard through the pneumatic system vent. Venting the high-pressure air may produce a phosphate-ester base fog.

3. Safety wire the pneumatic bottle control valve discharge arm with MS20995CY15 copper safety wire. Refer to Chapter 20—“Safetying.”
4. If the emergency gear release handle is in normal position, proceed with servicing.

NOTE

The handle will stay in the normal position if the pneumatic system was used to operate the brake system. Also, the handle will stay in the normal position if pressure bleed off due to a system leak.

5. Attach adapter assembly and nitrogen supply. Refer to "Pneumatic Systems" in the *AMM*, for servicing high pressure gases through the service valve. Charge the bottle to 2000 psig at 70°F.

NOTE

After cold soak at extreme altitudes, the indicated pressure may be low. Allow storage bottles to warm to ambient temperatures before servicing.

6. Close service valve. Close nitrogen supply; remove servicing adapter assembly.
7. Check for leaks around service valve and install valve cap.
8. Close left nose bay door.

BRAKE ACCUMULATOR

Description

On aircraft 5001 thru 5500 the following are found inside the left nose bay door:

- Brake accumulator
- Brake accumulator service valve
- Pressure gauge

The air side of the brake accumulator is connected to the brake accumulator service valve and pressure gauge with a tube. The brake accumulator service valve and pressure gauge are outboard and aft of the accumulator. There is a brake accumulator bleed-valve on the aircraft structure adjacent to the brake accumulator service valve. The brake accumulator bleed-valve gives the maintenance personnel a way to remove pressure from the hydraulic system before service is done.

On aircraft 5501 and on, the brake accumulator service valve and pressure gauge are found behind the brake service door on the left side of the aircraft, next to the battery service door. The brake accumulator is found under the aft fuselage fairing. A brake accumulator bleed-valve is on the aircraft structure below and to the left of the brake hydraulic reservoir. The brake accumulator bleed-valve gives the maintenance personnel a way to remove pressure from the hydraulic system before service is done on the brake accumulator.

Service Brake Accumulator

Service the Brake Accumulator (aircraft 5001 thru 5500):

1. Disengage the SKID CONTROL circuit breaker on the left CB panel.
2. Open left nose bay door.
3. Push the brake accumulator bleed-valve for four or five seconds to remove hydraulic fluid pressure from the accumulator.
4. Connect the adapter and hose assembly from the pressure source to the brake accumulator service valve.
5. Open the brake accumulator service valve to add nitrogen to the brake accumulator. Refer to "Pneumatic Systems" in the *AMM*.
6. Push the brake accumulator bleed-valve for four or five seconds again to make sure that all of the hydraulic fluid is removed from the accumulator.
7. Close the brake accumulator service valve and disconnect the hose assembly. Refer to "Pneumatic Systems" in the *AMM*.
8. Hold a rag in front of the brake accumulator service valve and open the valve to see if brake hydraulic fluid comes out.
9. If hydraulic fluid comes out of the brake accumulator service valve, the accumulator is leaking and must be replaced. Refer to Chapter 32—"Hydraulic Pack Assembly and Accumulator."

10. If no hydraulic fluid comes out of the brake accumulator service valve, connect the adapter and hose assembly from the pressure source to the brake accumulator service valve.

11. Open the brake accumulator service valve and add nitrogen until the accumulator is charged to a pressure of ± 675 psi (4654 kPa, ± 172.4 kPa) as shown on the pressure gauge. Refer to "Pneumatic Systems" in the *AMM*.

12. Close the brake accumulator service valve and disconnect the hose assembly. Refer to "Pneumatic Systems" in the *AMM*.

13. Make sure that the brake accumulator service valve does not leak.

14. Install the valve cap on the brake accumulator service valve.

15. Engage the SKID CONTROL circuit breaker on the left CB panel.

16. Close the left nose bay door.

Do the Service for the Brake Accumulator (aircraft -5501 and on):

1. Disengage the SKID CONTROL circuit breaker on the left CB panel.

2. Open the brake service door on the left,aft side of the aircraft.

3. Turn the brake accumulator bleed-valve handle clockwise for four or five seconds to remove hydraulic fluid pressure from the accumulator.

4. Connect the adapter and hose assembly from the pressure source to the brake accumulator service valve.

5. Open the brake accumulator service valve to add nitrogen to the brake accumulator. Refer to "Pneumatic Systems" in the *AMM*.

6. Turn the brake accumulator bleed-valve clockwise for four or five seconds again to make sure that all of the hydraulic fluid is removed from the accumulator.

7. Close the brake accumulator service valve and disconnect the hose assembly. Refer to "Pneumatic Systems" in the *AMM*.

8. Hold a rag in front of the brake accumulator service valve and open the valve to check to see if brake hydraulic fluid comes out.

9. If hydraulic fluid comes out of the brake accumulator service valve, the accumulator is leaking and must be replaced. Refer to Chapter 32—"Motor/Pump and Accumulator."

10. If no hydraulic fluid comes out of the brake accumulator service valve, connect the adapter and hose assembly from the pressure source to the brake accumulator service valve.

11. Open the brake accumulator service valve to add nitrogen until the accumulator is charged to a pressure of 675 psi, ± 25 psi (4654 kPa, ± 172.4 kPa) as shown on the pressure gauge. Refer to "Pneumatic Systems" in the *AMM*.

12. Close the brake accumulator service valve and disconnect the hose assembly. Refer to "Pneumatic Systems" in the *AMM*.

13. Make sure that the brake accumulator service valve does not leak.

14. Install the valve cap on the brake accumulator service valve.

15. Engage the SKID CONTROL circuit breaker on the left CB panel.

16. Close the brake service door on the left, aft side of the aircraft.

TIRES

Description

Servicing the tire by maintaining correct inflation pressure is the most important job in any tire preventative maintenance program. Improper inflation pressure causes uneven tread wear.

Underinflation—Indicated by excessive wear in the shoulder area, is particularly severe. It increases the chance of bruising sidewalls and shoulders against rim flanges. In addition, it shortens tire life by permitting excessive heat buildup.

Overinflation—Is indicated by excessive wear in the center of the tire. This condition reduces traction, increases tire growth and makes treads more susceptible to cutting.

Servicing the tire(s) requires maintenance personnel to handle compressed gas. Observe safety precautions.

Servicing

Safety Precautions:

1. Allow the tire and brake to cool before attempting to service.

WARNING

Introducing relatively cooler nitrogen into a tire that is hot (or when the brakes are hot) may cause the tire to burst.

2. Stand at a 90° angle to the axle along the centerline of the tire during servicing.

WARNING

The tendency of a bursting tire is to rupture along the bead. Standing in front of either bead area could cause injury if the tire bursts.

CAUTION

Applying a tire sealant on the tire may cause wheel corrosion.

3. Follow all local safety and technical directives while servicing tires.

Procedures:

1. Check tire pressure regularly.

   a. Tire pressures must be checked with an accurate gauge on a regular basis (daily, if aircraft is operated daily). When practical, check pressures before every flight.

   b. Check only cool tires at least two to three hours after a flight. Use an accurate gauge. Inaccurate gauges are a major cause of improper inflation.

2. Use the recommended tire pressure.

CAUTION

Refer to Chapter 32—“Wheels - Troubleshooting” Table 101 of the *AMM*, when tire pressure falls below recommended limit, to determine proper corrective action.

   a. Maintain main gear tire pressure at 210 psig, +2 or –5 psig (1448 kPa, +14 or –34 kPa) unloaded, 218 psig, +2 or –5 psig (1503 kPa, +14 or –34 kPa) loaded with an ambient temperature of 70°F.

   b. Maintain nose gear tire pressure at 130 psig, ± 5 psig (896 kPa, ± 34 kPa) unloaded, 135 psig, ± 5 psig (930 kPa, ± 34 kPa) loaded with an ambient temperature of 70°F.

NOTE

The operating pressures are to be measured with the weight of the aircraft on the wheels.

3. Adjust tire pressures for climate change.

   a. Climate changes have an effect on tire pressure when flying from a hot climate to a cool climate and vice versa. When temperature change is extreme (changes in excess of 50°F). For example, a tire inflated/utilized in a warm climate drops in air pressure when the aircraft on which it is installed is flown to a cold climate. Bringing an aircraft out of a heated hangar into the cold winter weather does the same.

   b. In either circumstance, overinflate tires to compensate for the subsequent cooling and loss of pressure caused by extreme temperature changes. As a general rule, an ambient temperature change of 5°F produces a pressure change of about 1%.

Servicing Characteristics

Loss of Tire Pressure:

1. A slight amount of diffusion through the carcass in tubeless tires is normal. The sidewalls are purposely vented in the lower sidewall area to bleed off the diffused air preventing separation or blisters.

2. A tire may lose as much as 5% of the initial inflation pressure in a 24-hour period. This is considered normal. A tire with an abnormally high leak-down rate must be replaced. Applying an unapproved tire sealant to the tire may cause wheel corrosion or cause an out of balance condition.

Above Normal Brake Energies Have Been Exceeded (Rejected Takeoff or Emergency Braking):

1. Even though inspection may show no apparent damage, the tires may have sustained incipient damage that could result in premature failure.

2. Also, wheels must be checked using the applicable wheel overhaul manual.

LANDING GEAR STRUT AND OLEO

Description

Complete servicing of the landing gear shock strut and oleo assemblies is required to ensure correct operation during taxi, takeoff, and landing. The correct gas (nitrogen) pressure must be maintained in the shock strut gas chamber, to prevent bottoming out during landing. All gas bubbles must be removed from the fluid chamber to prevent irregular operation.

Required safety precautions for servicing the shock strut and oleo are as follows:

- Safety and maintenance precautions pertinent to the handling of phosphate ester base hydraulic fluids are covered in “Hydraulic Fluid Systems” in the *AMM*.
- The safety precautions required when servicing the high-pressure gas are listed in “Pneumatic Systems” in the *AMM*.
- Any additional safety and/or technical precautions stated in local directives also apply.

Servicing Procedures

Nose Gear Strut Servicing:

1. Jack the aircraft until the tires clear the ground. Refer to Chapter 7—“Lifting,” for jacking instructions.

WARNING

High-pressure gas is dangerous. Personnel must fully understand the safety precautions when they work with high-pressure gas as outlined in “Pneumatic Systems” in the *AMM*.

2. Open the gas service valve and deplete the pressure in the lower chamber. After pressure is released remove the safety wire and the gas service valve to drain any fluid in the chamber.

3. Remove the oil fill plug from top of the strut.

4. Install the union assembly in the oil fill plug opening.

5. Connect the service hose from the hand pump service unit (containing the approved phosphate ester hydraulic fluid) to the union assembly.

   a. Close the check valve on hand pump and pump fluid into the strut until fully extended and pressure is 200 psig, ± 50 psig (1379 kPa, ± 344 kPa).

   b. Open the check valve on the hand pump and slowly move the strut to the compressed position with a hydraulic jack.

   c. Repeat the bleeding steps until no gas is returned to the service unit reservoir.

   d. With the strut in the fully compressed position, disconnect the hand pump service line.

   e. Remove the union assembly from the oil fill plug opening.

   f. Install the oil fill plug with a new packing (O-ring).

   g. Install safety wire on oil fill plug. Refer to Chapter 20—"Safetying."

6. Install the gas service valve with the new packing (O-ring) and the safety wire. Refer to Chapter 20—"Safetying."

7. Connect the nitrogen source to the gas service valve with the gauge/adapter assembly and service the oleo to 130 psig (896 kPa). Refer to "Pneumatic Systems" in the *AMM,* for servicing high-pressure gases through the gas service valve.

8. Slowly bleed off pressure to 100 psig (690 kPa). Remove gauge/adapter assembly and install dust cap.

Main Gear Oleo Servicing:

1. Jack the aircraft until the tires clear the ground. Refer to Chapter 7—"Lifting," for jacking instructions.

WARNING

High-pressure gas is dangerous. Personnel must fully understand safety precautions when working with high-pressure gas as outlined in "Pneumatic Systems" in the *AMM*.

2. Open the gas service valve and decrease the pressure from the oleo. Remove the safety wire and gas service valve after the pressure is released.

3. Connect the service hose from the hand pump service unit (containing the approved phosphate ester hydraulic fluid) to the oleo.

   a. Open the check valve on the hand pump and slowly move the oleo to the fully compressed position with a hydraulic jack. Record the distance between the upper and lower barrels.

   b. Close the check valve on the hand pump and add fluid into the oleo until it is fully extended.

   c. Open the check valve on the hand pump and slowly move the oleo to 0.4 inch (10.16 mm) from the fully compressed position with a hydraulic jack.

   d. Repeat steps (3a) and (3b) again, a minimum of four times.

   e. Complete the bleeding process with the oleo 0.4 inch (10.16 mm) from the fully compressed position.

NOTE

The fluid level must be at the service valve hole with the oleo 0.4 inches (10.16 mm) from the fully compressed position.

f. With the oleo 0.4 inches (10.16 mm) from the fully compressed position, disconnect the hand pump service line.

g. Make sure the oleo is kept full of fluid and do not release it from the fully compressed position (0.4 inches/10.16 mm) until the gas service valve has been installed.

h. Install the service valve with the new packing (O-ring). Safety wire the service valve. Refer to Chapter 20—"Safetying."

4. Connect the nitrogen source to the gas service valve with the gauge/adapter assembly and service the oleo to 397 psig (2737 kPa). Refer to "Pneumatic Systems" in the *AMM*, for servicing high-pressure gases through the gas service valve.

5. Slowly bleed off the pressure to 297 psig (2047 kPa) and remove the gauge/adapter assembly and install the dust cap.

SHIMMY DAMPER

Description

Measure the depth of the makeup piston inside shimmy damper by inserting a measuring probe in the open end of the retainer cap. If the measurement is 4.50 inches or greater, service shimmy damper.

Service Shimmy Damper

Remove service port cap and connect a hydraulic service pump, serviced with Skydrol. Refer to "Hydraulic Fluid Systems" in the *AMM*, for safety precautions.

Pump hydraulic fluid into the service port until the makeup piston is 4.12 inches from the end of the retainer cap or until hydraulic fluid begins to flow out the end of the retainer cap.

AFT CARRY-OUT FLUSH TOILET

Description

The aft carry-out flush toilet utilizes a waste container for solid and liquid waste, and a liquid reservoir for flushing the bowl assembly.

Service the toilet reservoirs after each flight. However, toilets must be serviced when the liquid level is too low for proper operation or the liquid appears to have incorrect chemical balance.

To assure toilet recirculating systems operate properly during freezing conditions, an ethylene glycol base anti-freeze containing antifoam agent may be added to the flush liquid.

CAUTION

Fluid is corrosive to structure and electrical connectors. Use extreme care to prevent spillage when servicing toilet.

Servicing Toilet

NOTE

General instructions for servicing are provided on a decal applied to the front side of the removable tank.

Tank Removal:

1. Gain access to the toilet tank by opening the door on the front of the seat assembly.

2. Depress the locking ring on the quick-disconnect securing flush line.

3. Drain any residue of flush fluid in the hose by partially disengaging the plug from the quick-disconnect and manipulating the hose to assist drainage.

4. Remove the flush hose from the quick-disconnect. Place hose in the retaining clip on the underside of the toilet mounting plate.

5. Install the plug attached to the quick-disconnect to seal the coupling.

6. Close the knife valve at the bottom of the toilet bowl by pushing the actuator handle until the valve is fully closed.

7. Press the two Pres-Loc fasteners on each side of the knife valve actuator to unlock the tank.

8. Remove the tank by pulling the recessed carrying handle on the tank top.

Tank Cleaning:

1. Dispose of tank contents by holding the tank upside-down over a sewer or toilet. Pull the knife valve actuator handle, open the valve and allow the tank to drain.

2. Rinse the tank by filling one-half full with water. Close the knife valve and shake vigorously. Drain tank again; repeat procedure until tank is clean.

NOTE

Commercial detergents and disinfectants may be included in the rinse water if desired. However, do not include these materials in the tank precharge.

NOTE

Rinse and drain the tank several times to ensure that the tank is thoroughly clean.

3. Wipe the exterior surfaces of the tank using a cloth moistened with clear water and disinfectant.

Tank Precharge:

1. Charge the tank with a mixture of 2 quarts of water and 2 ounces of Monogram ChemKare chemical.

NOTE

To assure toilet recirculation system operation during freezing weather, ethylene glycol base antifreeze containing anti-foam agent may be added to the flush fluid.

Tank Installation:

1. Reinstall the tank by inserting the slides on each side of the knife valve into the slide plate assembly on the bottom of the toilet, and slide tank into place.

2. Press the two Pres-Loc fasteners to the first detent to secure the tank.

3. Remove the plug in the flush hose quick disconnect and connect the flush line to the quick disconnect. Lock the locking ring.

4. Pull the actuator handle to fully open the knife valve.

5. Lift the toilet seat and shroud assembly from the top of the toilet and wipe with cloth moistened with clear water and disinfectant. Wipe the bowl and surrounding area.

6. Check flushing operation of the toilet and check for leaks.

7. Close the access door.

EXTERNALLY SERVICEABLE FLUSH TOILET

Description

Service the flush toilet during routine ground maintenance of the aircraft after usage. It is more efficient and convenient to service the toilet on a regular basis than to wait until the tank is filled to capacity.

Servicing Toilet

Waste Removal and Recharge:

1. Open the aircraft exterior service panel door.
2. Remove the water inlet cap and open the waste drain valve.
3. Connect the water fill hose and waste drain hose from a ground service unit to the aircraft service panel connections.
4. Toggle the PUSH TO OPEN lever on the upper edge of waste drain valve to open inner-waste drain-valve flapper door.
5. To dump waste, pull the drain valve handle and turn to lock.
6. Turn inlet water on and rinse tank with drain valve open.
7. Release drain valve handle and fill tank with 3.0 gallons of water.
8. Turn water off and open drain valve to empty tank.

NOTE

Maximum capacity of toilet tank is 4.0 gallons.

9. Note overfill light.
10. Release the drain-valve handle, charge toilet tank with 1.0 gallons of chemical to precharge level.
11. Note precharge light.
12. Disconnect the water fill hose and waste drain hose from the aircraft service panel connections.
13. Replace the water filler cap and close outer waste drain valve door, which in turn closes the inner flapper door.
14. Close and secure the aircraft exterior service panel door.

NOTE

To ensure toilet recirculation during freezing weather, ethylene glycol base anti-freeze containing an antifoam agent may be added to the flush fluid.

VANITY WATER SUPPLY

Description

The aft vanities with wash basins incorporate running water. The water system is a gravity-feed system consisting of a single storage tank, pressure transducer and necessary tubing to the faucet. The water may be heated when desired.

Servicing Vanity Water Supply System

NOTE

Servicing the vanity water supply system consists of replenishing the water tanks with potable water.

Service Water System:

1. Gain access to the water tank from inside the clothes closet. Unlatch the door/panel on the closet side wall.

WARNING

Before handling a hot water tank, verify that it has cooled to prevent personnel injury.

2. Disengage the circuit breaker for the water tank heater in the closet.

NOTE

The water hose quick-disconnects prevent drainage of any water that may be left in the tank.

3. Release the tank at the hold down and remove the tank. The electrical connector for the water tank heater will disconnect as the tank is removed.
4. Remove the filler cap from the tank and empty any water remaining in the tank. Rinse the tank out thoroughly with fresh, clean potable water.
5. Fill the water tank with fresh, clean potable water. Verify filler cap vent hole is clear and install filler cap.
6. Place the water tank in the cabinet. Carefully push the hot water tank in position to connect the electrical connector.
7. Secure the tank in place with the hold down.
8. Check for water flow from the faucet.
9. Close and latch the door/panel.

Freeze Protection in Cold Weather

The vanity water system is subjected to water freeze damage when the aircraft remains in below freezing weather over night or longer. Take the following actions to prevent freeze damage:

1. Remove the water tank. Refer to Chapter 38—"Servicing Vanity Water Supply System" in the *AMM*.
2. Drain remaining water from faucet.

OXYGEN SYSTEM

Description

The oxygen filler valve is inside the right nose bay door at FS 90.20. There is a pressure indicator gauge on the right instrument panel.

Breathing oxygen that conforms to MIL-O-27210 Type 1 must be used for charging the cylinders.

Precautions

WARNING

Oxygen supports combustion. Materials that do not normally flash in the atmosphere, readily burn or explode in the presence of concentrated oxygen.

Ensure that safety precautions are adhered to at all times:

1. Do not service the oxygen bottle while the aircraft is being fueled.
2. Ensure that no uncontained flammable material is near when servicing oxygen bottle.
3. Do not direct highly compressed oxygen towards personnel.
4. Follow all local safety directives.

Charging Oxygen System

Charge Oxygen System:

1. Open the right nose bay door.

2. Remove the oxygen filler valve dust cap on the right nose compartment aft frame.

3. Connect the charging cylinder line from the oxygen service cart to the filler valve.

4. Slowly open the charging cylinder valve and charge the aircraft oxygen bottle to the correct pressure.

CAUTION

Ambient temperature has a direct effect on indicated pressure.

5. Shut off oxygen at the charging cylinder and disconnect the line.

6. Install the dust cap on the filler valve.

7. Close the right nose bay door.

ACRYLIC WINDOW

Description

The openable cockpit side windows and cabin windows are constructed of stretched acrylic. Care must be exercised to avoid scratches and gouges caused by using improper cleaners and cleaning materials.

Each cockpit side window consists of an outer pane and an inner frost pane with dry air space between the panes.

The openable cockpit side windows and cabin windows are constructed of high impact materials, which withstand a wide range of temperature and pressure fluctuations. The inner and outer surfaces are constructed of stretched acrylic with a hardness similar to brass or copper.

Cleaning Windows

Use only approved cleaners and repellents when cleaning windows. Procedures for cleaning are as follows:

CAUTION

Never use paper towels, which are highly abrasive and cause hairline scratches on the window.

1. Determine what cleaner is required.

2. Read the manufacturer's instructions and precautions.

WARNING

Cleaners/solvents and repellents are petroleum based. Do not use near open flame. Some may have an effect on the aircraft finish. Take the appropriate steps for protection.

CAUTION

Remove all rings from fingers to prevent scratching the window when scrubbing.

3. Before cleaning windows with a general purpose cleaner, clean windows with a nonabrasive soap or detergent and water. (For example, mix 4 teaspoons (19.8 ml) of Joy or Ivory liquid dishwashing detergent per gallon (3.78 liters) of water.) Use bare hands or fingertips to feel and dislodge residue adhering to the transparency. A soft cloth or chamois may be used as a means of carrying water to the transparency.

   a. Apply a general purpose cleaner to the transparency one area at a time, then wipe with a soft, nonsynthetic cloth. Genuine chamois or 100% cotton terry cloth or flannel are good choices.

b. Cleaning with a circular motion can cause glare rings. Use long up and down straight strokes. Folding the cloth to expose a clean area after each pass prevents scratching from dirt that accumulates on the cloth. Discard the cloth when it becomes soiled. Repeat as necessary until all contaminants are removed from the surface of the transparency.

4. Allow the windows to air dry. Do not dry with cloth or chamois.

5. After cleaning, apply a coat of polish and wax to protect the windows.

Polishing/Waxing Windows

Polishing:

1. After cleaning the windows, apply a moderate amount of plastic polish to the outboard surface. Polish the surface with a polishing cloth using a circular motion.

2. Polishing time depends on surface conditions, like tape residue, dirt, light scratches or paint overspray. Repeat polishing as needed to obtain a clean surface finish.

Waxing:

NOTE

Apply a wax coating after cleaning and polishing to improve the overall appearance of the windows and make any repeat cleaning easier.

1. After the outboard surface has been cleaned and dried, apply a thin coat of wax. Wax the outboard surface with a polishing cloth using a circular motion.

2. Apply and rub the wax sparingly. Excessive rubbing scratches the acrylic and charges it with static electricity, which attracts dust.

Window Rain Repellent and Surface Conditioner

A rain repellent and surface conditioner may be used to increase natural cleaning of the acrylic windows during rain. Read and adhere to the repellent/surface conditioner manufacturer's instructions and cautions. If a substitute is used, check to see what effect it will have, if any, on the aircraft finish.

ELECTRIC HEATED GLASS WINDSHIELD AND SIDE WINDOWS

Description

The electric heated windshield and heated side windows are of glass construction. Care must be exercised in cleaning these windows to avoid damage or deterioration to the Surface Seal™ rain repellent outer surface coating.

CAUTION

Do not apply unauthorized rain repellent coatings or compounds to the electric heated glass windshield or associated heated glass side windows. Surface Seal™ is the only authorized rain repellent coating. Apply only with the windshield manufacturer's authorization and instructions.

Each electric heated side window incorporates an inner frost pane. The dry air space between the frost pane and heated window assembly is maintained with heated air.

For cleaning/polishing/servicing of acrylic windshields and windows, refer to Chapter 56—"Acrylic Windows" in the *AMM*.

The electric heated windshield and electric heated forward side windows are comprised of all glass construction with bonded fiberglass edge attachments, to withstand a wide range of temperature and pressure fluctuations. Heating the windshield/windows is accomplished through an electrically conductive film applied to the inner surface of the outboard glass ply.

Cleaning Heated Glass Windshield/Forward Side Windows

Use only approved cleaners and materials when cleaning the windshield/windows. Procedures for cleaning are as follows:

NOTE

Clean the outer glass surface of the electric heated windshield and forward side windows in a manner that protects the Surface Seal™ water repellent coating.

1. Flush the outer surface of windshield/side windows with clean water to remove excessive dirt and other substances.

NOTE

Dislodge any surface particles using fingers or fingernails.

CAUTION

Do not use abrasive materials such as pumice or strong acid based cleaners. These materials damage the Surface Seal™ water repellent outer coating of the windshield and forward side windows.

2 Using materials, such as a soft cloth or clean sponge, wash the windshield/side windows with a 50/50 solution of isopropanol and water. If isopropanol is not available, the following alternate cleaning solutions may be utilized:

- A 50/50 solution of rubbing alcohol and water.
- Mild liquid dishwashing liquid (Ivory or Joy) mixed 1/4-ounce (7.1 ml) per gallon (3.8 liters) of water.
- Full strength Windex glass cleaner.

3. Flush thoroughly with clean water then dry. Wipe dry with strokes in one direction using a damp soft cloth, damp sponge, or soft paper towel (Kaydry Wipes).

NOTE

Do not apply polish or wax to the glass surface of the heated windshield or heated forward side windows.

BATTERY

Description

A new battery is normally shipped discharged and contains the proper amount of electrolyte. It does not require leveling even though the battery may appear to have insufficient electrolyte.

The electrolyte, which is 30% by weight solution of potassium hydroxide in distilled water, does not take an active part in the chemical reaction. The electrolyte is used only to provide a path for the current flow. At 70°F (21°C), the specific gravity (density) of the solution must remain within the range of 1.24 to 1.30.

Another unusual characteristic of the nickel-cadmium battery is that when completely discharged, some cells reach zero potential and charge in the reverse polarity. This action adversely affects the battery, so that it does not retain a full capacity charge. As a result, the battery becomes equivalent to a much lower-rated battery. To solve this problem discharge the battery and short-circuit each cell to obtain a cell balance at zero potential. This process is known as equalization.

Never service a nickel-cadmium (NiCad) battery inside the aircraft. The battery electrolyte has a high affinity for carbon dioxide. Any amount of electrolyte expelled reacts with carbon dioxide to form white crystals of potassium carbonate. This substance is noncorrosive, nontoxic, and nonirritating. It can be wiped away with a clean damp cloth.

Operation

Two basic requirements must be met to avoid battery failures and/or damage:

- The electrolyte level must be maintained as defined below in "Servicing Battery" for the type battery used.
- The battery must be maintained in a fully operational state of charge condition so that an engine start is not attempted with a low battery.

The more frequently an engine is started with the battery, the more frequently the battery requires servicing. Refer to the *FAA Approved AFM* for engine starts with battery limitations.

Safety Precautions:

- Make electrolyte adjustment with distilled, deionized, or demineralized water only.
- Do not overfill.
- Do not add water when the battery is in a discharged state unless an abnormally high cell voltage reading (greater than 1.5 volts) is encountered immediately after placing the battery on charge.
- Do not add electrolyte.
- Do not use acid or tools which have any acid on them. Personal injury and/or equipment damage may result.

Servicing Battery

WARNING

The electrolyte used in nickel-cadmium batteries is a caustic solution of potassium hydroxide. Serious burns result if contact is made with any part of the body. If electrolyte gets on the skin, wash the affected areas with large quantities of water, neutralize with 3% acetic acid, vinegar, or lemon juice. If electrolyte gets into the eyes, flush with water and get immediate medical attention.

Marathon Battery Electrolyte Level Check:

1. Remove battery from the aircraft. Refer to Chapter 24—"Battery."
2. Remove battery cover.
3. Liquid level is determined by looking down into the vent well after removing the cap.

NOTE

Remove the cap using a nylon filler-cap vent plug wrench.

4. If it is not possible to determine the liquid level in the manner above, use a clear polystyrene tube, open at both ends, about six inches long with approximately 1/8-inch inside diameter.
5. Insert the tube into the filler opening for enough to touch the top of the plates or plastic insert.
6. Hold the tube between the thumb and the fingers. Place your index finger over the top end of the tube and remove the tube from the filler well.
7. The electrolyte level should be 1/8-inch above the visible insert after allowing the battery to stand 2 to 4 hours following a charge. If time does not permit the 2 to 4 hour waiting period, an approximate level will be about 1/4-inch above the visible insert immediately after charge.
8. If no liquid is withdrawn, add distilled or demineralized water until the proper level is reached in the polystyrene tube. Use a syringe to add the liquid.
9. Reinstall the filler cap using a nylon wrench.
10. Install the battery cover.
11. Install the battery in the aircraft. Refer to Chapter 24—"Battery."

Saft Battery Electrolyte Level Check:

CAUTION

Addition of water by any method other than the following procedure causes spewing and loss of electrolyte during overcharge.

1. Remove the battery from the aircraft. Refer to Chapter 24—"Battery."

2. Remove the battery cover.

NOTE

The electrolyte is at its maximum level and is most uniform from cell to cell near the end of the constant current charge, with the charging current still flowing. Check the electrolyte level during the last half hour charge.

3. Remove the relief valves (vented caps) with a special plastic tool provided in the Saft tool kit.

NOTE

The Saft tool kit contains the special plastic tool and syringe. The syringe has a nozzle that is cut to a specific length for the Saft battery.

4. Insert the syringe into the cell opening until the shoulder of the nozzle rests on the valve seat.

5. Withdraw the syringe plunger and check for any liquid in the syringe. If the level is too low, the syringe remains empty.

6. If liquid is drawn into the syringe, draw out excess liquid until the level of the electrolyte is at the nozzle end level. This is the correct electrolyte level.

7. If the electrolyte level is too low, perform the following procedure using only distilled or demineralized water.

   a. Draw a measured amount of distilled/demineralized water (i.e., 5 cc) into the syringe and inject it into the cell.

   b. With the syringe nozzle inserted in the cell (with the shoulder of nozzle resting on valve seat), fully withdraw the syringe plunger.

   c. If the syringe remains empty, repeat (a) and (b), counting the number of 5 cc injections used to achieve the correct electrolyte level.

   d. At the point (in above item (b) above when excess liquid is drawn into the syringe, the correct level for that cell has been reached.

   e. Expel the excess liquid into a container for proper disposal.

8. It is important to check that the quantity of water added per cell does not exceed 25 cc. If water consumption is too high, check the setting of the charging system or regulator. If the setting is correct, shorten the time period between servicing.

SEALED LEAD ACID BATTERY

Description

The lead acid battery in the aircraft is rated at 44 ampere-hours and is maintenance free. For removal and installation of the lead acid battery, refer to Chapter 24—"Sealed Lead Acid Battery" in the *AMM*.

Storage Requirements

The lead acid battery used in the aircraft must be serviced and charged upon receipt of new battery and must be recharged (when in storage) every 90 days.

Place new batteries into service from storage within 2 years of the manufacture date.

Batteries not recharged every 90 days when in storage must be conditioned by discharging at the test rate of 35.2 amperes for one hour. Charging after conditioning must be at 28.2 VDC, ± 0.5 VDC, and 3.5 amperes (constant current) for approximately 18 hours (or until the voltage reaches 30 volts and remains 30 volts for one hour).

Battery Checks

Recharge the battery when its open circuit voltage drops below 2.08 volts per cell or the open circuit voltage drops below 25.0 VDC.

CAUTION

Never deep cycle the lead acid battery. Whether in storage or in operation, do not allow the lead acid battery voltage to drop below 18 VDC. Even if subsequent recharging restores the battery voltage to an acceptable level (25 VDC minimum), the battery life cycle could be severely degraded.

CAUTION

If the lead acid battery open circuit voltage is above 18 VDC but below 22 VDC, the battery must be removed and serviced.

NOTE

The following tests may be performed if the capacity of the battery is in question.

Reserve or Emergency Capacity Test

1. Make sure the battery is fully charged.

2. With the battery temperature above 59°F (15°C), discharge the battery at the rate of 35.2 amperes for one hour.

3. Using a voltmeter, check the open circuit voltage. Voltage must be 18 VDC or greater.

4. If the battery fails the voltage check, it is no longer considered serviceable and must be replaced.

A visual check of battery compartments must include observing the exterior case for evidence of deformities or burned areas. Ensure the vent tubes are not pinched or deformed. When batteries are disconnected, visually check the battery terminals for evidence of burns or arcing.

Battery Charging

The battery must be charged using a constant potential or constant voltage charger regulated at 28.2 VDC (± 0.5 VDC).

The battery must contain a reserve or emergency capacity. The aircraft electrical system can charge the battery by placing the battery switch to on with generators operating or with external power applied, provided the battery voltage is above 22 VDC.

CAUTION

If the battery appears to be "dead", do not attempt to charge using the aircraft generators or external power.

Always make sure that the battery is either disconnected or ensure that the battery switch is off during long periods of maintenance with external power applied.

R134A AIR CONDITIONING SYSTEM

Description

The R134a vapor cycle air conditioning system uses a refrigerant that alternately evaporates and condenses, to move heat from one location to another. Heat is removed from the cabin through the evaporators and is expelled to the outside air through the condenser.

Remote servicing ports for the air conditioning system are inside the tail cone maintenance door outboard and aft of the compressor and condenser assembly. The inboard port is high side and the outboard port is low side.

Servicing the R134a air conditioning system consists of discharging and charging the system.

Precautions

Observe safety precautions when handling refrigerant or servicing and performing maintenance on the air conditioning system. Care must be taken to minimize the release of refrigerant into the atmosphere. The Environmental Protection Agency (EPA) requires recycling/recovery of R134a as of November 15, 1995. All reclamation and recovery equipment must be EPA and UL listed. Use the R134a reclamation system per manufacturer instructions whenever evacuating the system.

Liquid refrigerants at normal atmospheric pressure and temperature expand and absorb heat. As a result, the refrigerant freezes anything it contacts. Use of protective clothing, gloves, and goggles. Protect the skin and eyes.

Never weld, use a blow torch, or use excessive amounts of heat on (or in the immediate area of) any part, or the air conditioning system, or a refrigerant supply tank, while they are closed to the atmosphere (charged or not).

Connection of low-pressure equipment (gauges, refrigerant bottles) to the high side of the compressor can result in personal injury or equipment damage. Always ensure valves on gauges are closed when connecting gauges. Ensure that hoses are properly connected.

Federal law prohibits the servicing of liquid refrigerants by non-certified personnel.

A mercury thermometer cannot be used in aircraft due to the hazard of possible mercury reaction with aluminum.

Servicing

Discharging System:

NOTE

To allow for pressure equalization in the high and low sides, let the system stabilize for 10 minutes after shutdown before discharging.

1. Disengage AC circuit breaker on left CB panel.

2. Remove the service port caps and connect the servicing cart (per manufacturer instructions).

3. Slowly open the high and low side valves enough to allow pressure to discharge.

NOTE

The system must be bled down very slowly to avoid losing compressor oil.

4. Close the high and low side valves after all system pressure has discharged.

5. Check and record the oil level in the catch bottle (on the reclaimer).

Charging System:

1. Disengage AC circuit breaker on left CB panel.

2. Remove the service port caps and connect the servicing cart (per manufacturer instructions).

3. Open high and low side valves and pull a vacuum of 25–27 inches Hg.

NOTE

If a vacuum of 25–27 inches Hg is not obtained in 10 to 15 minutes, a leak is indicated.

4. Shutoff the vacuum pump and allow system to stabilize.

5. Note gauge reading and check 30 minutes later.

NOTE

Evacuate the system for 30 minutes minimum at a minimum of 25 in. Hg.

6. If pressure in the system changes within the 30 minute time period, a leak is indicated.

7. Following the servicing cart instructions, replace any oil vented while discharging system.

NOTE

Care must be taken to not add more oil than was vented. Too much oil in the system can deteriorate the cooling performance of the evaporators. If the compressor was drained, 5 oz. (147.87 ml) of clean, new oil should be injected into the system.

8. Following the service cart instructions, open the high and low side valves and allow approximately 0.5 pounds (0.23 kg) of refrigerant to enter the system until a pressure of 50 psig (335 kPa) minimum is shown on the gauges for approximately five minutes.

NOTE

If internal pressure is not 50 psig (335 kPa) or above, the binary switch does not close and the drive motor does not operate. To ensure an internal pressure of 50 psig (335 kPa), the container can be heated following service cart instructions.

9. Close high and low side valves and verify that the system internal pressure remains at 50 psig (335 kPa) or above.

10. Connect an external power source to the aircraft.

11. Engage the AC circuit breaker on the left CB panel and fan circuit breakers (HT032 and HC034) in the aft junction box.

12. Set the BATT switch to ON.

13. Select AC HIGH on AC-FANS rotary switch.

14. Verify airflow across both the evaporator and condenser.

CAUTION

Do not overcharge the system or component damage may occur.

15. Open the low side service valve only and add approximately 2.5 pounds (1.13 kg) of R134a refrigerant until the sight glass, (on the receiver dryer), has cleared. Do not exceed maximum system pressures. Approximate capacity of system is 3.5 pounds (1.59 kg) of R134a.

NOTE

As refrigerant enters compressor, the compressor speed reduces along with a slight increase in discharge pressure (2 to 5 psig or 13.4 to 33.5 kPa).

16. Close the low side valve and allow the system to stabilize for 10 minutes. Recheck the sight glass and high side (discharge pressure) gauge.

17. If the sight glass is not clear, open the low side valve and add a small quantity (0.10 pounds or 0.5 kg) of refrigerant until the majority of bubbles disappear or high side pressure reaches its limits.

ENVIRONMENTAL AND PRESSURIZATION

Description

The environmental control unit (ECU) and the pressurization system use components that need to be serviced periodically. This section is a consolidation of those components and the procedures to service them. For scheduled periodic time servicing, refer to Chapter 5—"Inspection Time Limits" in the *AMM*.

ECU components that need serviced are the water separator and ozone converters.

Safety and Maintenance Precautions

Safety Precautions:

When work is done on or around the ECUs after engine shutdown, let the ECU cool, or wear protective clothing. The bleed-air components/ducting becomes extremely hot during and immediately after the engine has operated.

Maintenance Precautions:

Never use a screwdriver or similar tool to pull apart a part or assembly. A pull with a slight twisting motion is normally enough to separate assemblies.

Water Separator Cleaning

The water separator on the ECU needs condenser cleaning and replacement at regular intervals.

Tools and Special Equipment:

- Mild Detergent—Commercially available
- PD-680 Type III—Commercially available
- Epoxy-polyamide primer—Commercially available

Clean the Water Separator:

1. Remove the water separator from the aircraft. Refer to Chapter 21—"Cool Air Distribution System" in the *AMM*.

2. Remove the coupling clamp. Pull the inlet shell assembly, outlet duct assembly, and condenser assembly apart.

3. Remove the screws and gasket from the condenser assembly.

4. Attach a string to the spring on the large end of the condenser.

5. Disconnect the spring, then pull the chain assembly and attached spring through the hem of the condenser, leaving the string in the hem for reassembly.

6. Remove the spring at the small end of condenser.

7. Remove the condenser from the support assembly by pulling the condenser toward the narrow end of the support assembly. Use care to avoid damage to condenser.

8. Remove the safety wire, screws, washers and valve assembly from the support assembly.

CAUTION

Air dry the condenser without wringing or scrubbing.

9. Put the condenser in a solution of mild detergent and lukewarm water, and rinse until the water runs clear.
10. Let the condenser air dry thoroughly.
11. Clean the remaining parts with PD-680 Type III and dry thoroughly with compressed air.
12. Apply epoxy-polyamide primer (MIL-P-23377) to the mating surfaces of the washers and screws.
13. Put the valve assembly on the support assembly and attach with washers and screws while the primer is wet.
14. Safety the screws with wire. Refer to Chapter 20—"Safetying" in the *AMM*.
15. Attach the end of the spring to the string (inside the hem of the condenser).
16. Pull the string to install the spring and attached chain assembly in the hem, then remove the string from the spring.

NOTE

New condensers come with a string in the hem.

17. Connect the free end of the spring through the fastener at the end of the chain assembly and move the assembly into the recess at the base of the support assembly.
18. Pull the condenser toward the small end of the support assembly so that the condenser is tight.
19. Join the ends of the spring and attach the condenser to the groove at the small end of the support assembly.
20. Apply epoxy-polyamide primer (MIL-P-23377) to the mating surface of the screws.
21. Put the gasket on the condenser assembly and attach with screws while the primer is wet.
22. Assemble the condenser assembly, outlet duct assembly and the inlet shell assembly.
    a. Attach with coupling clamp.
23. Install the water separator in the aircraft. Refer to Chapter 21—"Cool Air Distribution System" in the *AMM*.

Ozone Converters Cleaning

The ozone converters, in the tail cone maintenance access between FS 462.50 and FS 479.50 (left and right), need to be cleaned at various intervals.

Ozone converter cleaning:

1. Remove the ozone converters from the aircraft. Refer to Chapter 21—"Ozone Converter" in the *AMM*.
2. Blow no more than 60 psig of compressed air into the converter, opposite the direction of normal flow, to loosen dirt and debris.
3. Flush the converter with a solution of phosphate-free detergent and warm water in the direction opposite of normal flow.
4. Flush the ozone converter with clean, warm water (or steam) in the direction opposite of normal flow, immediately after flushing it with detergent solution to remove all soap residue.
5. Remove as much water from the unit as possible by blowing compressed air through the inlet and/or outlet flanges.
6. Dry the ozone converters.

Methods for drying:

a. Put the ozone converter in an oven at a temperature of 250° F (121°C) and bake for two hours, or until completely dry.

b. Dry the ozone converters on a rack in a dry room.

c. Use a hair dryer to dry the ozone converters.

d. Use shop air to dry the ozone converter.

NOTE

Once the ozone converters are installed in the aircraft, the hot bleed-air completely dries the ozone converters.

7. Install ozone converters in aircraft. Refer to Chapter 21—"Ozone Converter" in the *AMM*.

SCHEDULED SERVICING

DESCRIPTION

This section provides necessary information to perform scheduled lubrication and cleaning of the aircraft. This section does not include lubrication procedures necessary for the accomplishment of maintenance practices.

Contents in this section are subdivided to give personnel a table separate from text and illustrations, to prevent confusion.

PRECAUTIONS

Lubricants are flammable and must be handled accordingly. Remove all spilled or excess lubricants from or around the aircraft.

Cleaning solvents can be harmful to personnel and aircraft components. Be aware of all manufacturer warnings/precautions before use.

APPLICATION

General mechanical knowledge, common sense, and cleanliness are necessary to correctly lubricate the aircraft. The information listed below assists in the selection and application of lubricants.

Lubricants and dispensing equipment must be kept clean.

Use only one lubricant in a grease gun or oil can.

Store lubricants in tightly closed containers in a protected area.

Before lubricating, wipe grease fittings and areas to be lubricated with a clean dry cloth.

When lubricating bearings which are vented, force grease into fittings until old grease is extruded.

After lubrication, clean excess lubricant from all but actual working parts.

When flush-type grease fittings are specified, use a special grease gun adapter.

All sealed or prepacked antifriction bearings are lubricated with MIL-PRF-23827 grease by the manufacturer (unless otherwise specified).

NOTE

Unless specifically forbidden, MIL-PRF-81322 grease may be used in all applications in which MIL-PRF-23827 is called out in this manual. Do not mix the two types of grease. MIL-PRF-23827 may not be substituted when MILPRF-81322 is called out.

Do not oil antifriction bearings or expose them to spray from steam or chemical cleaning.

Lubricate unsealed pulley bearings, rod ends, pivot end hinge points and all other friction points that obviously need lubrication with general purpose oil (MIL-PRF-7870).

FLIGHT CONTROLS

Description

The aircraft must be serviced in a contamination-free area, free from sand, dust, or other environmental conditions that contribute to unsatisfactory lubrication practices.

Lubrication Notes

Control Cables and Chain Assemblies:

During preventive maintenance as outlined in Chapter 5 (or more often as conditions make it necessary) lubricate the control cables, pressurization seals and chains as follows:

1. To clean the cables, moisten a clean cloth with MIL-PRF-680 Type III solvent. Do not soak the cloth or cables with solvent, because solvent that gets to the cable core washes out the lubricant and results in rapid wear and corrosion.
2. Wipe the cables dry.
3. Lubricate the cables (near the pressurization seals) the full length of the cable travel, using silicone grease, Cessna Part Number 5565450-28 or Dow Corning Molykote 33.
4. Fill the pressurization seals with silicone grease, Cessna Part Number 5565450-28 or Dow Corning Molykote 33.
5. Chain assemblies will be lubricated with silicone grease, Cessna Part Number 5565450-28.
6. Wipe off unwanted lubricant as needed.

Trim Tab Actuators:

The trim tab actuators must be lubricated with silicone grease, Cessna Part Number 5565450-28. Do not substitute lubricant on the trim tab actuators.

Trim Tab Actuators Lubrication

Special Tools and Equipment:

- Grease gun
- Silicone Grease (5565450-28)

Lubricate the Trim Tab Actuator:

1. Remove access panels 351AB and 352AB from the lower surface of the horizontal stabilizer to get access to the elevator trim tab actuators. Refer to Chapter 6—"Access Plates and Panels Identification" in the *AMM*.
2. Remove access panel 340DR on the right side of the vertical stabilizer and the forward rudder closeout panel, to access the rudder trim tab actuator. Refer to Chapter 6—"Access Plates and Panels Identification" in the *AMM*.
3. Operate the trim control and position the trim tabs to fully retract the actuator screw assemblies as follows:
   a. Put the aileron trim tab to the full-down position.
   b. Put the elevator trim tabs to the full-down position.
   c. Put the rudder trim tab to the full-right position.

4. Put grease into the grease fitting with a gun until the grease can be seen coming out of the actuator.

5. Operate the trim control and cycle the actuator through full travel three to four times, then wipe the grease from actuator.

6. Install the access panels on the lower surface of the horizontal stabilizer.

7. Install the access the panel on the right side of the vertical stabilizer and forward rudder closeout panel.

Horizontal Stabilizer Actuator Reservoir Oil Leakage Acceptance Criteria and Oil Level Check

Description

The actuator uses a hydraulically-powered motor, which extends and retracts two jackscrews. Each jackscrew is lubricated by an oil bath (contained in a reservoir). Actuating the jackscrews results in some of the lubricating oil being released from the reservoirs. This oil leakage is permitted. If the oil loss per jackscrew is more than 50 ml per 1200 flight hours (or three years). Replace the actuator.

The capacity of each reservoir is approximately 200 ml.

CAUTION

Do not overfill. If the actuator is overfilled, it forces the oil past the jackscrew seals at the top of the actuator.

Do the oil level check with the actuator fully retracted (stabilizer leading edge down).

NOTE

Use (MIL–PRF–7808) oil or equivalent when servicing the hydraulic actuator to prevent internal damage to the jackscrew assembly. Refer to “Scheduled Servicing” in the *AMM.*

Horizontal stabilizer actuator oil level check for actuators incorporating oil level sight glass:

1. Remove lower vertical stabilizer access panels 340AL and 340BR to get access to the actuator. Refer to Chapter 6—“Access Plates and Panels Identification” in the *AMM*.

NOTE

Actuator part number 41011680-103 must have 96 hours since the retraction cycle for an accurate oil level reading.

CAUTION

Before an oil level check is performed, make sure that the horizontal stabilizer is in the –2° position (take-off) with flaps extended. The oil level check must be done with the actuator fully retracted (stabilizer leading edge down). Adding oil to the actuator in any other position results in an overfull condition damaging the actuator and/or aircraft. The oil level decreases as the jackscrew extends.

2. Check the oil level at both sight glasses. The sight glass on the left actuator faces aft, while the sight glass on the right actuator faces forward.

3. If you cannot see the oil level in the sight glass, remove the safety wire and fill plug. Add oil (MIL-PRF-7808) as needed to raise the level to the center of the sight glass. Refer to "Scheduled Servicing" in the *AMM*. Careful records must be kept of oil fill amounts. Refer to Chapter 27—"Two Position Horizontal Stabilizer System" in the *AMM*.

NOTE

Signs of oil leakage are found by examining of the actuator, actuator compartment, and accurately kept maintenance records.

4. Install the fill plug and safety wire it. Refer to Chapter 20—"Safetying" in the *AMM*.
5. Install the lower vertical stabilizer access panels.

Horizontal stabilizer actuator oil level check for actuators without oil level sight glasses:

1. Remove lower vertical stabilizer access panels 340AL and 340BR to access the actuator. Refer to Chapter 6—"Access Plates and Panels Identification" in the *AMM*.

CAUTION

Before an oil level check is performed, make sure that the horizontal stabilizer is in the –2° position (take-off) with flaps extended. The oil level check must be done with the actuator fully retracted (stabilizer leading edge down). Adding oil to the actuator in any other position results in an overfull condition, damaging the actuator and/or aircraft. The oil level drops as the jackscrew extends.

2. Remove the safety wire and fill plug. Add oil (MIL-PRF-7808) as needed to raise the level to the bottom of the fill plug port. Refer to "Scheduled Servicing" in the *AMM*. Careful records must be kept of oil fill amounts. Refer to Refer to Chapter 27—"Two Position Horizontal Stabilizer System"in the *AMM*.

NOTE

Signs of oil leaks are found by examining the actuator, actuator compartment, and accurately kept maintenance records.

3. Install the fill plug and safety wire it. Refer to Chapter 20—"Safetying" in the *AMM*.
4. Install the lower vertical stabilizer access panels.

Lubricate Horizontal Stabilizer Scissor Fitting Bolts and Aft Pivot Bolts

Special Tools and Equipment:

- Grease Cessna Part Number U197005 (MIL–PRF–83261), commercially available.
- Grease Gun to match Zerk fitting.

Lubricate Horizontal Stabilizer Scissor Fitting Bolts and Aft Pivot Bolts:

- Remove the access Panels 340CR, 340GR, 340HR, 340BL, 340DL, 340EL and horizontal stabilizer upper and lower wiper panels. Refer to Chapter 6—"Access Panels and Plates" in the *AMM*.

NOTE

On aircraft 5574 and on, record the location of the access panel 340BL screws. This data is used to ensure the screws are in the correct locations.

Lubricate the horizontal stabilizer scissor fitting bolts (aircraft 5085 and on and aircraft 5001 thru 5084 incorporating SB560XL–27–05:

NOTE

The bolts do not need to be torqued again if the cotter pin remains installed during the lubricating procedure.

1. Using a grease gun and MIL–PRF–83261, lubricate the bolts.
2. Turn the bolt 1/3 turn and lubricate.
3. Turn the bolt an additional 1/3 turn and lubricate.

Lubricate the horizontal stabilizer scissor fitting bolts aircraft 5001 thru 5084 not incorporating SB560XL–27–05:

1. Remove the upper and lower scissor fitting. Refer to Chapter 55—"Two Position Horizontal Stabilizer" in the *AMM*.
2. Examine the bolts for corrosion, nicks, gouges, and other damage. Replace the bolts if they are damaged.

NOTE

It is recommended that SB560XL–27–05 be done at this time with the bolts removed.

3. Examine the scissor fittings for corrosion, bushing deformation, nicks, gouges, and other damage.
   a. Make sure that the upper to lower fitting attachment bushing inside diameter is 0.5629 inches, ± 0.0004 inches (14.2 mm ± 0.01 mm).
   b. Make sure that the upper scissor fitting outer lug to spar attachment bushing inside diameter is 0.3759 inches, ± 0.0004 inches (9.5 mm ± 0.01 mm).
   c. Make sure the upper scissor fitting inner lug to spar attachment bushing inside diameter measures 0.5629 inches, ± 0.0004 inches (14.2 mm ± 0.01 mm).
   d. Make sure the lower scissor fitting outer lug to horizontal spar attachment bushing inside diameter is 0.5629 inches, ± 0.0004 inches (14.2 mm ± 0.01 mm).
   e. Make sure that the scissor fitting inner lug to the horizontal spar attachment bushing inside diameter is 0.5009 inches, ± 0.0004 inches (12.7 mm ± 0.01 mm).
4. Lubricate the outside of the bolt shank with MIL–PRF–83261 grease, and also inside of the scissor link bushing.

5. Install the scissor fittings. Refer to Chapter 55—"Two Position Horizontal Stabilizer."

Lubricate the horizontal stabilizer aft pivot bolts aircraft 5031 and on and aircraft 5001 thru 5030 incorporating SB560XL–55–01:

NOTE

Bolts do not need to be torqued if the cotter pin remains installed during the lubricating procedure.

1. Remove the safety wire.
2. With a grease gun adapter, lubricate the aft pivot bolt with MIL–PRF–83261 grease.
3. Turn the bolt 1/3 turn and apply grease.
4. Safety the bolt with wire. Refer to Chapter 20—"Safetying" in the *AMM*.

Lubricate the horizontal stabilizer aft pivot bolts aircraft 5001 thru 5030 not incorporating SB560XL–55–01:

1. Remove the aft pivot bolts. Refer to Chapter 55—"Two Position Horizontal Stabilizer" in the *AMM*.
2. Examine the bolts for corrosion, nicks, gouges, and other damage. Replace the bolts if damage is found.

NOTE

It is recommended SB560XL–55–01 is accomplished at this time if more time is necessary.

3. Examine the aft pivot bolt fittings assembly for corrosion, bushing deformation, nicks gouges, and other damage.
4. Make sure the bushing inside diameters measure 0.7503 inches, ± .0010 inches (19.0576 mm, ± 0.0254 mm).
5. Safety the bolt with wire. Refer to Chapter 20—"Safetying" in the *AMM*.
6. Install access panels 340CR, 340GR, 340HR, 340DL and 340EL. Refer to Chapter 6—"Access Plates and Panels Identification" in the *AMM*.
7. Install the access panel 340BL as follows:

   a. On aircraft 5001 thru 5573, install the access panel 340BL.

   CAUTION

   On aircraft 5574 and on, use the correct screw length on the access panel 340BL. A screw that is too long damages the rudder assembly.

   b. On aircraft 5574 and on, install the access panel 340BL. Use data from removing the panel to install the screws in the correct location.

8. Install the horizontal stabilizer upper and lower wiper panels. Refer to Chapter 55—"Two Position Horizontal Stabilizer" in the *AMM*.

Flap Rollers Lubrication

Description

This task must be done in conjunction with other lubrication tasks for the flight control system. The flaps must be fully extended for access to all of the flap rollers.

Special Tools and Equipment:

- Lubrication Kit, Portable: CJMD132-003 (or equivalent)
- Grease Gun Coupling: MS24203 (or equivalent)
- Grease: MIL–PRF–23827

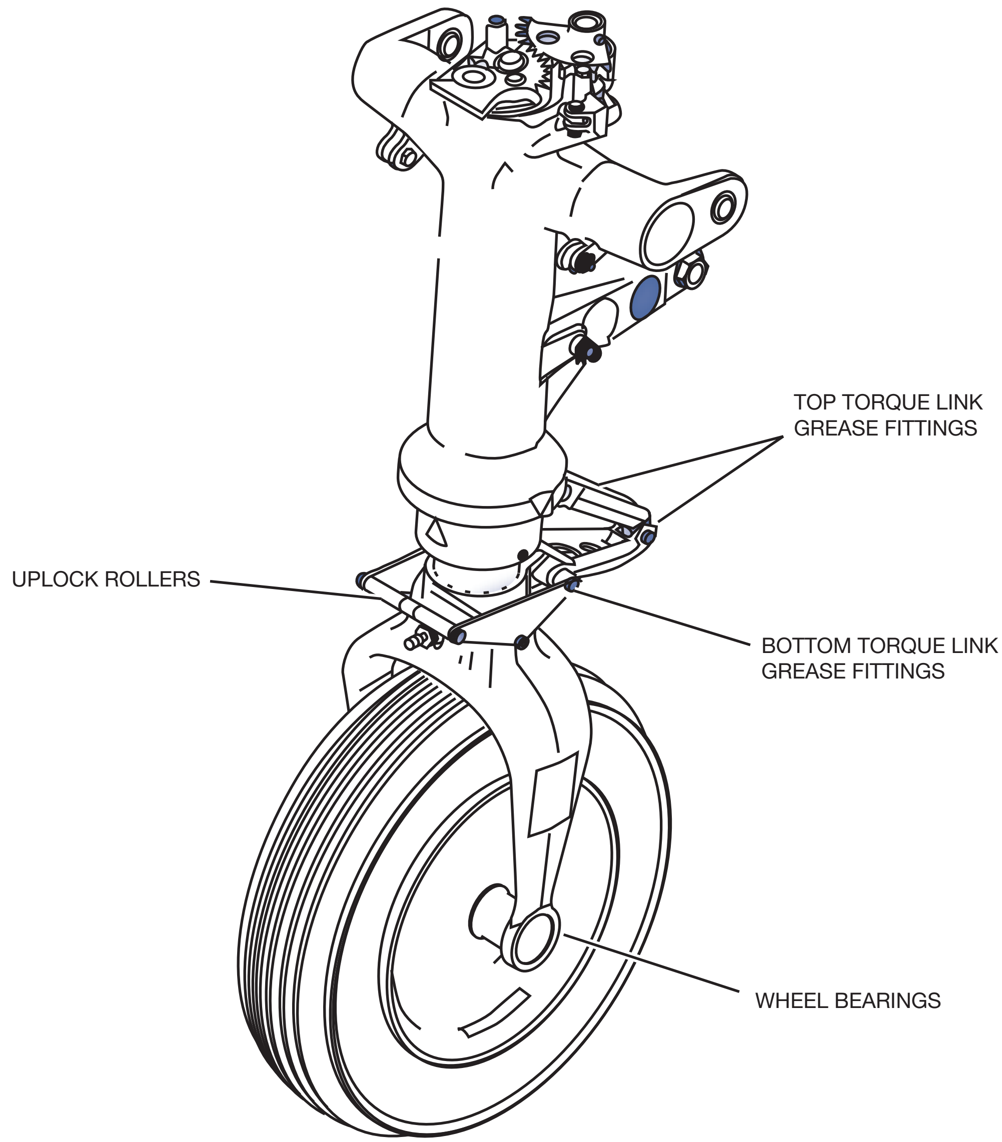


Figure 12-1. Nose Landing Gear Lubrication

Flap Rollers Lubrication:

1. Fully extend the flaps.
2. Remove the electrical and hydraulic power from the aircraft
3. Refer to the *AMM* to find the flap rollers.
4. Use clean cloths to carefully clean the roller grease fitting and the adjacent area.
5. Apply MIL–PRF–23827 grease to the fittings found in the bolt head of all of the flap rollers.

CAUTION

Do not apply grease to any part of the flap tracks. Clean as necessary.

6. Clean the flap tracks, if necessary.
7. Make sure you see grease on the edge of the roller after you lubricate it.
8. Use clean cloths to remove all excess grease.

LANDING GEAR

Description

The aircraft must be serviced in an area that does not have contamination from sand, dust, or other environmental conditions that may contribute to poor lubrication practices.

Obey nose and main gear maintenance, and safety precautions before and during lubrication servicing.

Servicing equipment should include:

- Grease guns
- Oil cans
- Brushes
- Clean cloths
- Other equipment necessary for proper lubrication servicing

WARNING

Lubricants are flammable. Do not expose to any ignition source.

Nose Landing Gear Lubrication

Clean all grease fittings with a clean dry cloth before lubricating.

Lubricate the Nose Gear Torque Links. Refer to Figure 12-1.

NOTE

Mobil Aviation Grease SHC 100 is the recommended grease for Goodrich wheel bearings, axle threads and axle nut threads. Do not mix Mobil Aviation Grease SHC 100 with any other grease. MIL–PRF–81322 grease can be used as an alternate, or on other manufacturer's wheels.

1. Lubricate the top torque link at three places with a grease gun and MIL-PRF-23827.
2. Lubricate the bottom torque link at two places with a grease gun and MIL-PRF-23827.

Lubricate the nose landing gear door control rods:

Use a grease gun and MIL–PRF–23827 to lubricate each of the nose landing gear door control rod bearings that have an integral grease fitting.

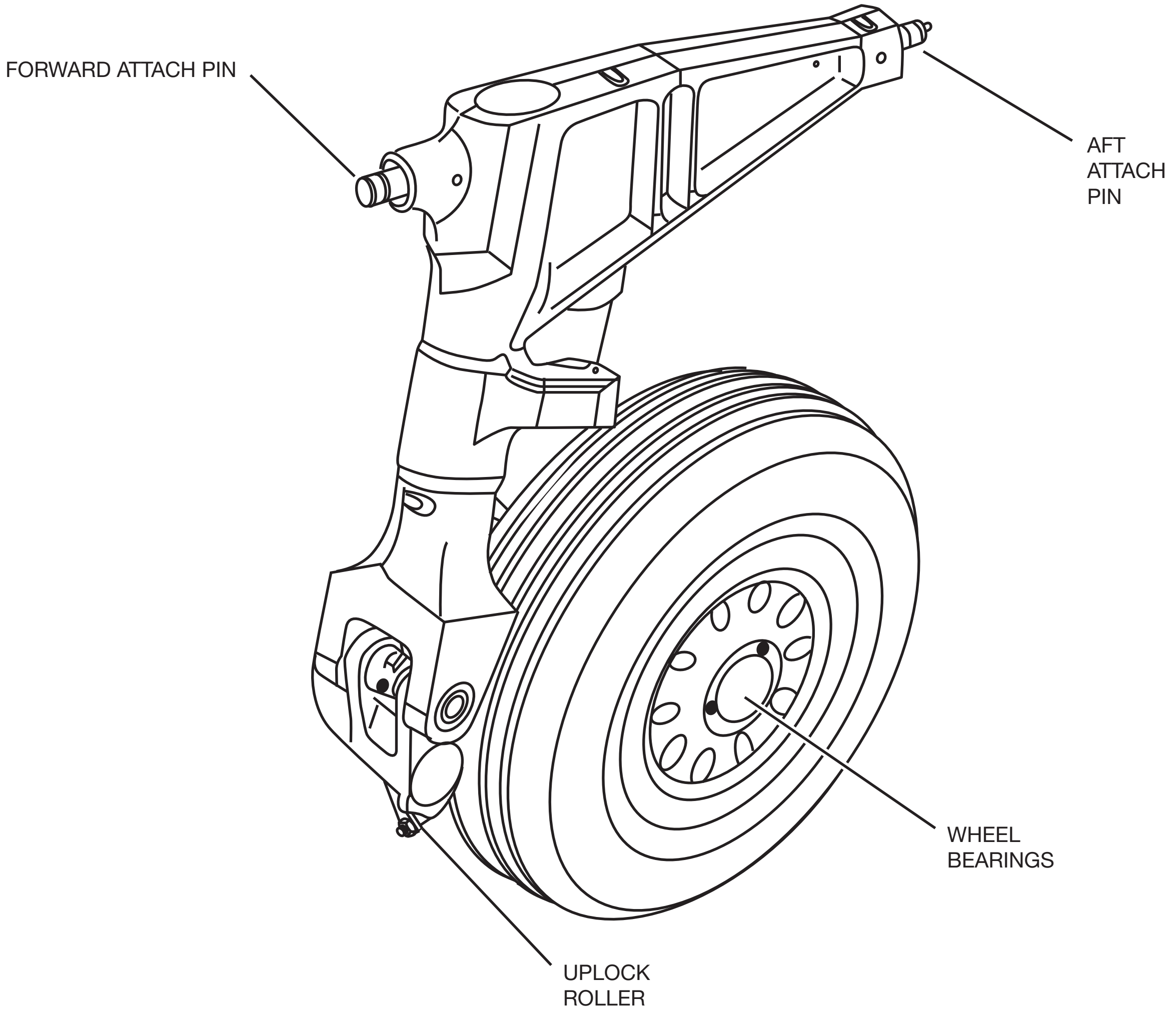


Figure 12-2. Main Landing Gear Lubrication

Lubricate the nose landing gear uplock rollers:

1. Remove the safety wire from the uplock roller bolts.
2. Remove the bolts, washers, bushings, bearing, and spacer from the bracket assembly.
3. Apply grease by hand to the spacer.
4. Install the spacer, bearing, bushings, washers, and bolts to the bracket assembly.
5. Make sure that the uplock rollers turn freely.
6. Install safety wire on the uplock roller bolts. Refer to Chapter 20—"Safetying" in the *AMM*.

Inspect and Pack the Nose Gear Wheel Bearings

Inspect and Pack the Nose Gear Wheel Bearings:

1. Remove the nose wheel and tire assembly from the axle. Refer to Chapter 32—"Nose Landing Gear Wheel" in the *AMM*.

CAUTION

Touch the bearings carefully and avoid contact with dirt, dust, moisture, or other contaminants.

2. Thoroughly clean the bearings with solvent. Blow out excess solvent with clean, dry compressed air.

CAUTION

Do not turn the bearings with compressed air when they are dried.

3. Wash the bearings again with clean solvent. Turn each of the bearing cages by hand after immersing fully in solvent.
4. Remove the bearings from the solvent and thoroughly rinse in toluol, isopropyl, or butyl alcohol.
5. Let the bearings air dry for five minutes.
6. Visually inspect the bearings for nicks, corrosion, signs of overheating, or other types of damage.
7. Fill the bearings with grease and wrap in waxed or greased paper to protect it against corrosion or dirt until it is installed in the wheel.
8. Install the nose wheel and tire assembly. Refer to Chapter 32—"Nose Landing Gear Wheel" in the *AMM*.

Main Landing Gear Lubrication

Clean all of the grease fittings with a clean dry cloth before lubrication.

Lubricate the main gear attach pins. Refer to main landing gear lubrication (Figure 12-2).

NOTE

Mobil Aviation Grease SHC 100 is the recommended grease for Goodrich wheel bearings, axle threads, and axle nut threads. Do not mix Mobil Aviation Grease SHC 100 with any other grease. MIL–PRF– 81322 grease can be used as an alternate, or on other manufacturer wheels.

1. Lubricate the forward attach pin using a grease gun and MIL–PRF–23827.

NOTE

The forward attach pin grease fitting is accessible through an opening in the top of the trunnion.

2. Lubricate the aft attach pin with a grease gun and MIL–PRF–23827.

NOTE

The aft attach pin grease fitting is accessible from the inboard flap well.

Lubricate the main landing gear uplock roller:

1. Remove the cotter pin, nut, washers, uplock roller, and bolt from the trailing link assembly.
2. Apply grease by hand to the bolt.
3. Install the bolt, uplock roller, washers, nut, and cotter pin in the trailing link assembly.
4. Make sure that the uplock roller turns freely.

Inspect and Fill Main Gear Wheel Bearings

Inspect and fill the main gear wheel bearings with grease:

1. Remove the main wheel and tire assembly from the axle. Refer to Chapter 32—"Main Landing Gear Wheels" in the *AMM*.

CAUTION

Touch the bearings carefully and avoid contact with dirt, dust, moisture, or other contaminants.

2. Remove the bearings from the wheel assembly.
3. Thoroughly clean the bearings with solvent. Blow out the remaining solvent with clean, dry, compressed air.

CAUTION

Do not turn bearings with compressed air when drying.

4. Wash the bearings again with clean solvent. Turn the bearing cage by hand after fully immersing it in the solvent.
5. Remove the bearings from the solvent and thoroughly rinse in toluol, isopropyl or butyl alcohol.
6. Let the bearings air dry for five minutes.
7. Visually inspect the bearings for nicks, corrosion, signs of overheating, or other types of damage.
8. Fill the bearings with grease and wrap in waxed or greased paper to protect against corrosion or dirt until installed in the wheel.
9. Install the main wheel and tire assembly. Refer to Chapter 32—"Main Landing Gear Wheels" in the *AMM*.

ENTRANCE DOOR

Description

It is recommended that the aircraft be serviced in an area free of contamination from sand, dust, or other environmental conditions, which may contribute to improper lubrication practices.

Obey all warnings and cautions related to the handling of lubricants.

Servicing equipment should include the following for correct lubrication and servicing:

- Oil can
- Brushes
- Clean cloth
- Other related equipment

Main Entrance Door Lubrication

Lubricate all areas shown in "Lubrication Details" in the *AMM* with the correct lubricant.

WARNING

Lubricants are flammable. Do not expose to any ignition source.

NOTE

Clean the lubrication point with a clean cloth before lubrication.

Main entrance door hinge bolts lubrication:

NOTE

Lubricate the parts in accordance with schedule provided in Chapter 5— "Inspection Time Limits" in the *AMM*, and when high pressure washing is performed in and around the door hinge area.

Lubricate with Molykote G–N. No disassembly is necessary.

DOOR LOCKS

Description

The Citation XL/XLS has seven exterior doors that can be locked with a key. To ensure that lock assemblies are maintained and in good working order, it is necessary to provide lubrication.

For lubrication location on a typical key lock assembly, refer to "Lubrication Details" in the *AMM*. The lockable exterior doors are the:

- Cabin entrance door
- Single point refueling door
- Left and right nose avionics compartment door
- Battery door
- Tail cone baggage door
- Tail cone maintenance access door

Door Lock Lubrication

Door Lock Lubrication:

1. Using Medeco Key Lube (PXP), refer to the list of lubricants in "Scheduled Lubricating/Cleaning" in the *AMM*, spray a small amount in each key lock opening.
2. Operate the key lock several times.
3. Repeat as needed to ensure each key lock assembly operates properly.

THRUST REVERSER

Description

The following instructions identify the lubricant and the lubrication procedures used on the thrust reverser system pivot points and sliding surfaces.

Lubrication Procedure

NOTE

Refer to the *Nordam Group Thrust Reverser Component Maintenance Manual* for lubrication intervals and procedures.

NOTE

It is recommended that lubrication be applied during other thrust reverser maintenance. Aircraft that operate in high humidity, coastal areas, or that are usually parked on the ramp must have lubricant applied at more frequent intervals.

Preliminary:

1. Deploy the thrust reversers. Refer to Chapter 78—"Thrust Reverser" for operation.
2. With the thrust reversers in deploy position, stop the supply of electrical and hydraulic power to the aircraft. If electrical power on the aircraft is necessary, disable the thrust reverser system by disconnecting the electrical connectors from the thrust reverser control valves. Refer to Chapter 78—"Thrust Reverser" for disable procedures.

WARNING

Failure to remove electrical and hydraulic power could result in serious injury to personnel and/or damage to the aircraft.

3. Remove the stang fairings. Refer to Chapter 78—"Thrust Reverser" for removal/installation:

Application:

Refer to the *Nordam Group Thrust Reverser Component Maintenance Manual* for lubrication procedures.

1. Install the stang fairings. Refer to Chapter 78—"Thrust Reverser" for installation.
2. Connect the electrical connectors to the thrust reverser control valves. Refer to Chapter 78—"Thrust Reverser" for installation.
3. Retract the thrust reversers. Refer to Refer to Chapter 78—"Thrust Reverser" for operation.

EXTERIOR

Description

Wash the aircraft frequently in order to keep its appearance and minimize corrosion. Polish the painted area of the aircraft at periodic intervals to remove chalking paint and restore the gloss.

Water/detergent cleaning is the recommended method to clean the exterior surface of the aircraft. For recommended water-detergent application on the aircraft, refer to the list of exterior cleaners in the *AMM*.

Precautions

Read and obey all of the manufacturer's instructions, warnings, and cautions on the cleaning/solvent compounds used.

WARNING

When the wing leading edges are cleaned and polished, take extreme care not to radius or break sharp corners of the boundary layer energizers. Boundary layer energizer edge sharpness must be maintained within a maximum allowable 0.01 inch radius. If the radius exceeds that limit, the boundary layer energizers must be replaced.

WARNING

Use all solvents in a well ventilated area and obey all normal safety precautions during use.

Clean the Exterior

The following information assists personnel in selecting the correct cleaning compounds and steps for cleaning the aircraft.

1. Close all doors, hatches, windows, and any other openings.

   CAUTION

   Do not high pressure wash any antenna or its base. Hand washing is recommended.

   CAUTION

   Do not force water into any contamination vents or drain holes in the antennas or fairings.

2. On the exterior surface of the aircraft:

   a. Make sure all inlet and outlet openings are covered to prevent wash spray entry. For protective covers used on the exterior surface of the aircraft, refer to Chapter 10—"Parking" in the *AMM*.

   b. Make sure generator dust covers are on both engine cowls to prevent possible damage to generator bearings (from wash spray or caustic soap).

3. Cover all of the lubricated parts which could be affected by cleaning solvents.

   CAUTION

   Do not let the tires stay in pools of cleaning solution any longer than necessary to clean the aircraft to prevent possible tire damage.

4. Cover all of the tires with a suitable cover.

   CAUTION

   Do not pressure wash the wheels or brakes. The carbon disks in the brake assemblies must be kept dry. Degradation of the carbon properties and possible freezing of the brake assemblies may occur if carbon is exposed to water.

5. Protect the components in the landing gear wheel wells.

6. Use presoftened bristle fiber brushes to scrub the aircraft. Let the cleaner soak for a few minutes, especially on heavily soiled, stained areas.

   CAUTION

   Do not brush the windows.

7. Use nonatomizing spray equipment.

   CAUTION

   Do not direct high-pressure water directly on bearings, electrical, electronic equipment, or antennas.

   CAUTION

   Spray off the cleaning solution before it dries. Cleaning solution left to dry causes spotting and streaking of the finish.

   NOTE

   A low-pressure sprayer helps to localize cleaning solution in tight areas.

8. Hand wash the antennas.

CAUTION

Avoid buffing the global positioning system (GPS) antennas. Negative effects could result due to static build-up.

9. Thoroughly pressure wash all of the surfaces contacted by the cleaning solution with water, preferably warm water 120°F to 140°F (49°C to 60°C).

10. Clean the windshield/windows. Refer to Chapter 56—“Acrylic Window” and “Electric Heated Glass Windshield and Side Windows” in the *AMM*.

WARNING

Use extreme care not to radius or break sharp corners of boundary layer energizers during cleaning/polishing operations on wing leading edges. Boundary layer energizer edge sharpness must be maintained within a maximum allowable 0.080 inch radii. If radii exceed that limit, boundary layer energizers must be replaced.

11. Unpainted, polished (mirror finish) aluminum, or stainless steel surfaces must be kept clean and bright by frequent hand polishing with a clean cloth to remove any stains or dirt. If the surface has deterioration that cannot be cleaned by hand, a muslin buffing wheel with T-41 Tripoli compound may be used to restore a mirror finish. Do not polish anodized aluminum.

INTERIOR

Description

Clean the aircraft frequently to maintain its appearance and minimize corrosion.

Interior Cleaning Products

The information listed in “Interior Cleaners” in the *AMM* assists with selecting the appropriate cleaning agents, and in steps to clean the aircraft.

WARNING

Use solvents only in well-ventilated areas. Use normal safety precautions.

Clean the Interior

Clean the interior decorative materials:

1. Clean with Yosemite Y–999 (or equivalent) as follows:
   a. Spray or wipe the soiled surface.
   b. Wipe off the solution with a clean cloth dampened with water.
2. Clean with aliphatic naphtha as follows:
   a. Wipe the spot with a clean cloth dampened with naphtha then wipe dry with a clean cloth.
   b. Remove as much tar, asphalt or chewing gum as possible with a knife. Apply naphtha to the residue and then wipe dry with a clean cloth. This method has a buffing effect that eliminates the possibility of stain from the solution.

Clean rugs, drapes, curtains and upholstery fabrics:

Use a dry-cleaning compound, vacuum cleaner, whisk broom, or any other general cleaning utensil to assist in cleaning the interior.

WARNING

Do not use a vacuum cleaner or any object which may generate sparks while aircraft is being fueled.

1. Remove as much tar, asphalt, or chewing gum as possible with a knife. Apply naphtha PD–680 Type III to the area and wipe with a clean cloth. This method has a buffing effect that prevents the possibility of stain.

2. Host dry-cleaning compound.
   a. Sprinkle compound liberally on the soiled area.
   b. Rub the compound into the soiled area.
   c. Remove the compound with a vacuum cleaner.

NOTE

This compound is nonflammable and may be used on fueled aircraft.

3. Wet shampoo.
   a. Remove the carpet or upholstery from the aircraft. If at all possible, use the spot-cleaning method.
   b. Vacuum the carpet and upholstery, removing as much dirt and dust as possible.
   c. Place a tablespoon of shampoo in a pail and direct a jet of water into the shampoo to produce sufficient foam.
   d. Apply the foam uniformly over the surface to be cleaned.
   e. Remove the suds by wiping with a brush or clean cotton cloth. Because there is very little moisture in the foam, the fabric is not wet and does not retain moisture.

CAUTION

Do not use a mechanical shampooer. It will distort the carpet.

4. Spot cleaning.

If at all possible, spot-clean tufted carpet in the aircraft, rather than completely removing the carpet for shampooing.

   a. Soak a clean white cloth with perchloroethylene solution.

CAUTION

Do not pour perchloroethylene solution directly on the carpet.

   b. Rub the perchloroethylene cloth in a circular motion on the soiled spot.

CAUTION

Do not use a mechanical shampooer. It will distort the carpet.

   c. An upholstery hand shampooer may be used on difficult-to-clean areas.

6. To clean acrylic plastic, refer to Chapter 56—“Acrylic Window” in the *AMM*.

Cleaning Vanity and Toilet Area:

Clean, deodorize and disinfect the toilet area. Chapter 38—“Toilet and Relief Tube” in the *AMM*.

Leather:

CAUTION

Never use abrasives, solvents, saddle soap, dish detergent, household cleaners, or any other soap to clean leather. Always have severe soiling cleaned by a trained professional.

Clean leather immediately after soiling has occurred. Always remove soil with an appropriate absorbent material with a non-rubbing motion.

Vacuum and wipe all leather clean with a damp cloth for optimum life and beauty.

Use approved cleaning products to clean heavily soiled finished leather.

Clean Nubuck and Suede leather:

CAUTION

Always dry Nubuck and Suede slowly and away from direct heat.

1. Brush with a suede brush and vacuum to remove loose soil.
2. Use a very fine abrasive pad with just enough pressure to remove stubborn stains.
3. Apply cornstarch to oil/grease spots and cover with a damp cloth for four to six hours then remove the cornstarch. Repeat as necessary.
4. Clean food and beverages by pressing a clean, dry cloth onto the spot.
5. Flush blood or urine with clean water and then remove with a soft cloth.

NOTE

Use a professional care service to clean ink marks.

UNSCHEDULED SERVICING

DESCRIPTION

This section outlines procedures and recommendations to carry out normally unscheduled service. Instructions for ice and snow removal from the parked aircraft are provided.

Deicing procedures are included in the section to assist personnel in removing ice from the aircraft.

Ensure that all chemical suppliers instructions including bulletins, warnings, and cautions are adhered to.

DEICING/ANTI-ICING

Description

This servicing section is intended to provide maintenance personnel with the necessary information to deice and anti-ice aircraft when conditions of snow, ice or frost exist (or are anticipated to exist) on the aircraft. Deicing/anti-icing procedures must be followed in coordination with the flight crew. The final decision, regarding whether an aircraft components are free of frozen contaminants s made by the pilot in command.

The effectiveness of any freezing point depressant (FPD) deicing or anti-icing treatment can only be estimated due to many conditions that can influence holdover time.

These conditions are:

- Ambient temperature
- Aircraft surface temperature
- FPD fluid application procedure
- FPD solution strength
- FPD film thickness
- FPD fluid temperature
- FPD fluid type

- Close proximity to other aircraft, equipment and structures
- Operation on snow, slush, wet ramps, taxiways, and runway
- Precipitation type and rate
- Residual moisture on aircraft surface
- Relative humidity
- Solar radiation
- Wind velocity and direction

Deicing and anti-icing are two different procedures. They may be done separately or together. The one-step method is for deicing only. The two-step method for deicing followed immediately by anti-icing procedures. It is also possible to anti-ice a dry aircraft as a precaution against anticipated icing.

Approved Products

For a list of approved Type I deice fluids and Type II or Type IV anti-ice fluids, refer to "Deicing/Anti-Icing." (and Tables) in the *AMM.*

Deicing/Anti-Icing Precautions

Before Type I deicing procedures begin, maintenance personnel must familiarize themselves with areas to be sprayed and areas to avoid a direct spraying with fluid.

CAUTION

Type I deicing fluids must never be used full strength (undiluted). Undiluted glycol fluid is quite viscous below 14°F (–10°C) and can actually produce lift restrictions of about 20%. Additionally, undiluted glycol has a higher freezing point than glycol/water mixture.

If deicing/anti-icing procedures are performed with engines running, all cabin air intakes and bleed-air valves must be turned off.

Before Type II or Type IV anti-icing procedures begin, maintenance personnel must familiarize themselves with areas to be sprayed and areas to avoid spraying. Type II or Type IV anti-icing is applied primarily to protect:

- Wings
- Control surfaces
- Fuselage areas ahead of engine inlets to protect engines from possible ice ingestion

CAUTION

Although irritation from freezing point depressant fumes is classified as negligible, maintenance personnel must wear protective clothing during deicing/anti-icing procedures. Pure glycol, if swallowed in amounts of three ounces or more, may be fatal. Maintenance personnel must familiarize themselves with manufacturer Material Safety Data Sheets (MSDS) before deicing/anti-icing procedures begin.

Type I Deicing Preparations

Before deicing procedures begin, maintenance personnel need to know the lowest anticipated outside air temperature (OAT). Based on this information, the glycol/water mixture must then be adjusted to lower the freezing point of the Type I solution to at least 18°F (10°C) below this OAT. The difference between anticipated OAT and the freezing point of the solution is known as the "buffer".

Each manufacturer has specific instructions for mixing glycol/water and the freezing point that any given mixture will provide. Refer to these instructions when preparing Type I solutions.

Most manufacturers give a refractive index of their products. This index is required to ascertain the freezing point of any given solution. Tools used for glycol testing are listed in Chapter 12 of the *AMM*.

WARNING

It is the responsibility of deicing personnel to know the freezing point of any solution applied. A refractive index coupled with specific manufacturer data is the only definite method for identifying the freezing point of a previously mixed Type I solution when the glycol/water ratio is unknown.

CAUTION

Do not intermix brands of Type I deicing fluid. Manufacturers add specific dyes to their products for visual evidence of contamination. A fluid which does not meet the color criteria set forth by its manufacturer must be considered contaminated and must not be used.

Make sure that Type I deicing fluid is between 160°F and 180°F (71°C and 82°C) before application.

Type II or Type IV Anti-Icing Preparations

Type II or Type IV anti-icing fluids must be applied undiluted and at ambient temperature (unless otherwise specified by the manufacturer).

NOTE

Type II or Type IV anti-icing fluid has thickening agents added, which remain on the wings of an aircraft during ground operations or short term storage; thereby providing some anti-icing protection. This fluid flows off readily during takeoff at speeds of approximately 85 knots. Type II or Type IV anti-icing procedures provide longer holdover times than Type I deicing procedures.

CAUTION

Type II or Type IV anti-icing fluid must never be mixed with Type I deicing fluid. Type II or Type IV anti-icing fluid requires dedicated equipment and must not be dispersed with equipment used for Type I deicing fluid. Do not intermix brands of Type II or Type IV anti-icing fluids.

WARNING

Refer to the manufacturer's instructions for low temperature limits. If a Type II or Type IV anti-icing fluid is applied at temperatures lower than those approved by the manufacturer, the fluid remains on the aircraft and severely inhibits lift characteristics.

Make sure that dedicated Type II or Type IV equipment is set to apply low-to-moderate pressure fluid. Because Type II or Type IV anti-icing fluid is applied immediately after Type I deicing procedure, the Type II or Type IV equipment must be fully serviced and operational before any deicing begins.

Deicing Procedures

Preliminary removal of heavy snow is performed with brooms (or similar methods). Use caution when brushing around antennas, windows, flight controls, deice boots, probes, vanes, and similar obstructions.

If anti-icing is to follow deicing, anti-icing must begin immediately after completing the deicing procedure.

NOTE

The heat of the deicing fluid melts ice and snow. The only function of glycol in the deicing solution is to lower the freezing point of the fluid remaining on the aircraft.

Spraying hints for Type I fluid:

1. To reduce fluid heat loss, spray the fluid in a solid cone pattern in large, coarse droplets.

2. Spray the fluid as closely as possible to aircraft surfaces, but no closer than approximately 10 feet (if a high pressure nozzle is used).

3. If there is a thick layer of frozen snow or ice on the aircraft surface, it is better to concentrate a directed spray of heated fluid on one area until that section of the aircraft is cleaned. The hot fluid heats the aircraft surface, which loosens the frozen bond of ice and snow around the clean area.

4. Spray from the tip inboard, and from the leading to the trailing edge when spraying the wing and tail areas. This procedure takes advantage of dihedral to aid in fluid dispersion.

5. Make sure that the upper fuselage is clear to prevent chunks of ice and snow from being ingested into engine(s) during or after takeoff.

6. Do not spray windshields and windows directly.

7. Do not spray directly toward pitot heads and static ports.

Deice the aircraft:

NOTE

Record the time that deicing procedures begin. The length of time that deicing fluids remain effective is known as "holdover time" and is highly dependent on a number of variables. Refer to *FAA Tables* for Type I deice fluid approximate holdover times.

1. Deice the pilot side nose section and upper fuselage.

2. Deice the cabin fuselage behind the pilot side.

3. Deice the left wing.

4. Deice the left fuselage behind the wing.

5. Deice the tail section (left side).

6. Deice tail section (right side).

7. Deice the right fuselage behind the wing.

8. Deice the right wing.

9. Deice the cabin fuselage in front of the wing.

10. Deice the copilot side nose section and the upper fuselage.

11. If the anti-icing fluid is to be applied, skip steps (12) and (13) and proceed to the "anti-icing procedure" in this chapter. If no anti-icing fluid is to be applied, see steps (12) and (13).

12. Complete post-deice checks. Refer to "Post-Application Checks" in this chapter.

13. Convey deicing information to flight crew using the following statement: "This aircraft has been deiced using Type I deicing fluid with a freezing point of ________°F. Holdover time began at __________."

Anti-Icing Procedure

Anti-ice the aircraft:

WARNING

Never apply Type II or Type IV anti-icing fluids in diluted form. In addition, Type II or Type IV anti-icing fluids must never be applied to pitot heads, angle-of-attack vanes, control surfaces, windows and windshield, fuselage nose, lower side of radome, static ports, air inlets, or engines.

NOTE

Type II or Type IV anti-icing fluid must be applied within three minutes after deicing is completed, due to limited holdover times of Type I deicing fluid. If Type II or Type IV anti-icing fluid has been applied and the aircraft has not been dispatched before new ice forms, the aircraft must be completely deiced again; and a second Type II or Type IV anti-icing treatment must be applied immediately.

NOTE

Record the time anti-icing procedures begin. The length of time an anti-icing fluid remains effective is known as "holdover time" and is dependent on a number of variables. Refer to appropriate manufacturer's information for approximate holdover times of Type II or Type IV anti-ice fluid in undiluted form.

NOTE

Anti-icing fluid is applied to the aircraft surface at low pressure. forming a thin film on surfaces. Ideally, Type II or Type IV anti-icing fluids should just cover the aircraft surfaces without runoff. Type II or Type IV anti-icing fluids are applied only from the wing section aft, and on upper fuselage surfaces ahead of the engine inlets.

1. Apply anti-ice fluid to the left wing.
2. Apply anti-ice fluid to the left tail section and empennage.
3. Apply anti-ice fluid to the right tail section and empennage.
4. Apply anti-ice fluid to the right wing.
5. Complete the post-application check. Refer to the "Post-Application Checks" listed below.
6. Convey anti-icing information to flight crew with the following statement: "This aircraft has been anti-iced using Type II or Type IV anti-icing fluid. Holdover time began at ________".

Post-application checks:

After the aircraft has been deiced or anti-iced, maintenance personnel must perform a post application check to make sure that all critical areas are free of ice, snow, or slush.

These critical areas are as follows:

- Wing leading edges, upper surfaces, and lower surfaces
- Horizontal and vertical stabilizers
- All control surfaces and control surface gaps
- Speedbrakes and thrust attenuators
- Windshields for clear visibility
- Engine inlets
- All fuselage surfaces ahead of engine inlets
- Antennas
- Angle-of-attack vanes, pitot heads, and static ports
- Fuel tank and fuel cap vents
- Air inlet scoops
- Landing gear, wheel wells and associated cables, pulleys, and miscellaneous hardware

Post-Flight Clean Up

It is highly recommended that aircraft subjected to either deicing or anti-icing procedures be thoroughly cleaned after flight operations are completed. Refer to "External Cleaning" in this chapter for procedures.

Wheel Brake And Main Gear Wheel Well Deicing procedure

Wheel Brake Deicing:

In the event brake freeze-up is encountered from ice forming after the aircraft has been parked on the ramp (and when full deicing procedures are not required) the following must be performed to remove the ice from the brake area.

1. Utilize a ground heater if available.

CAUTION

Exercise care when using a ground heater to deice the brakes if aircraft is resting on ice or is in close proximity to other parked aircraft.

2. Spray or pour isopropyl alcohol on the brakes.
3. Cycle the brakes asymmetrically while applying engine power.
4. In known slush conditions, apply alcohol to the brakes (in spray form), before taxi/takeoff. This helps prevent brake freeze-up in flight.

Main gear wheel well deicing:

NOTE

Follow the manufacturer's instructions for best results and economy.

In known slush and ice forming conditions, apply ICEX or similar product in the wheel well area to prevent ice build-up during taxi. The main gear door hinge line is an area of primary importance. Ice build-up in this area does not allow the main gear to lock in the up position.

Servicing Deice Boots

The deice boots have a special, electrically conductive coating to bleed off static charges, which cause radio interference and may perforate the tail boots. Care must be exercised when working around the boots to avoid damaging this conductive coating and to avoid tearing the boots.

To prolong the life of surface deice boots, they must be washed and serviced on a regular basis. Keep the boots clean and free from oil, grease and other solvents that cause rubber to swell and deteriorate.

Recommended cleaning and servicing procedures are outlined below:

NOTE

Deicing and anti-icing fluids produce no adverse effects on the deice boots. Type I, Type II, or Type IV applications, however, require a more frequent application of ICEX and AGE MASTER Number I on the deice boots. The following procedure is approved by BFGoodrich for their deice boots.

CAUTION

To prevent damage to deice boot material, do not clean with petroleum-based liquids (such as Methyl n-Propyl Ketone, unleaded gasoline, etc.).

Clean the boots with mild soap and water, then rinse thoroughly with clean water.

NOTE

The temperature of water for cleaning deice boots shall not exceed 140°F (60°C).

NOTE

Isopropyl alcohol is be used to remove grime, which cannot be removed using soap. If isopropyl alcohol is used for cleaning, wash the area with mild soap and water, then rinse thoroughly with clean water.

To improve the service life of deice boots and to reduce the ice adhesion, it is recommended that the deice boots are treated with AGE MASTER No. 1 and ICEX.

AGE MASTER No. 1 is used to protect the rubber against deterioration from ozone, sunlight, weathering oxidation and pollution. ICEX is used to help retard ice adhesion and to keep the deice boots looking new longer. Both are recommended by BFGoodrich Company.

Both the AGE MASTER No. 1 and ICEX must be applied according to the manufacturer's recommended directions (outlined on the containers).

CAUTION

Protect nearby areas and clothing, and use plastic or rubber gloves during applications. AGE MASTER Number 1 stains and ICEX contains silicone, which makes paint touch up almost impossible.

CAUTION

Be sure to obey all manufacturer warnings and cautions,when using AGE MASTER No. 1 and ICEX.

SHINE MASTER and SHINE MASTER PREP are applied to the deicer boots to give the best smooth, shiny surface.

SHINE MASTER and SHINE MASTER PREP must be applied in accordance with the manufacturer's recommended directions (outlined on the containers).

Small tears and abrasions in surface deice boots can be repaired temporarily without removing the boots, and the conductive coating can be renewed. Citation Service Centers have the proper materials and procedures to perform these repairs.

NOTES

CHAPTER 21
AIR CONDITIONING

CONTENTS

ILLUSTRATIONS

TABLE

CHAPTER 21
AIR CONDITIONING

INTRODUCTION

This Chapter describes the air distribution, air conditioning, and pressurization systems on the model 560 XL/XLS/XLS+ aircraft. These three separate but interrelated systems are presented in three sections. Information is provided regarding air distribution within the cabin and how it is controlled. The components and their operation for the air conditioning (vapor cycle cooling) system and pressurization system are also discussed. References for this chapter and further specific information can be found in Chapter\ 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," Chapter 21—"Air Conditioning," and Chapter 36—"Pneumatics" of the *Aircraft Maintenance Manual* (*AMM*).

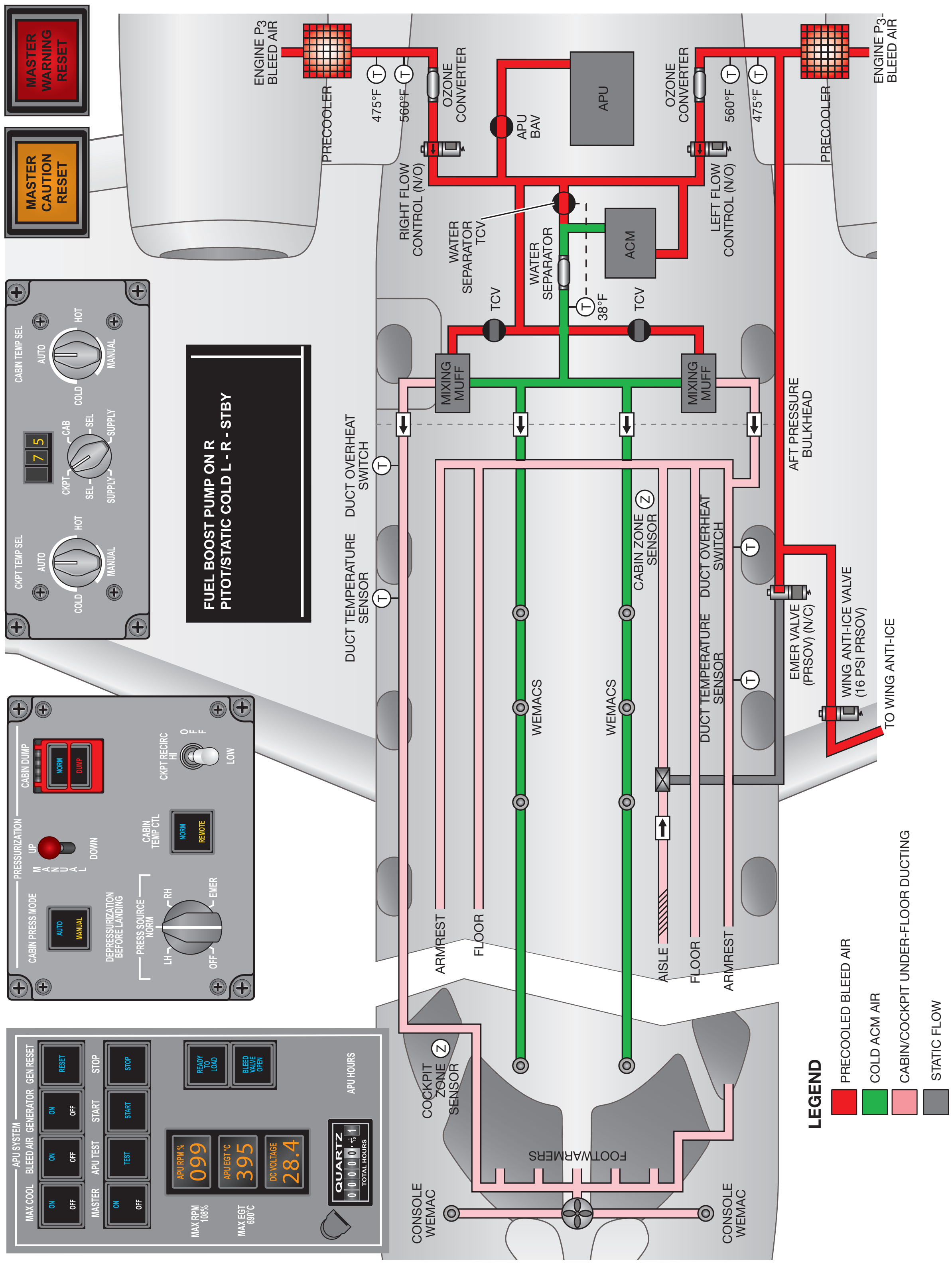


Figure 21-1. Air Distribution System

AIR DISTRIBUTION GENERAL

This section describes the devices and components used to create cool air/warm air, the methods of distributing this air to each area of the aircraft, and pressurization. The 560 XL/XLS/XLS+ uses a single allied signal environmental control unit (ECU) to transform hot engine [or auxiliary power unit (APU)] bleed air to cool conditioned air (Figure 21-1). This cooled conditioned air is available for use in the cool air distribution system or mixed with hot bleed air for the warm air distribution system.

NOTES

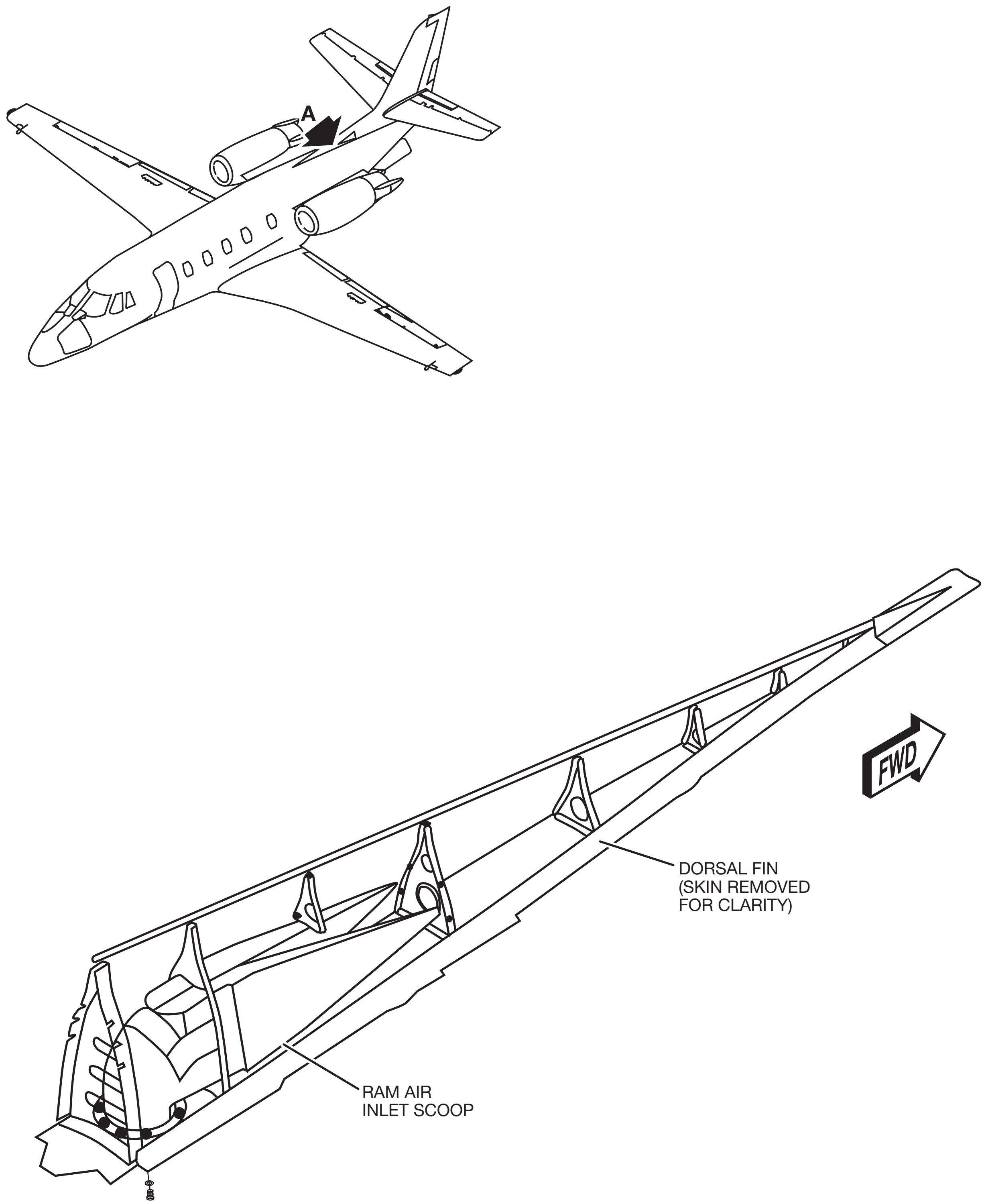


Figure 21-2. Ram Air Inlet

RAM AIR/FRESH AIR

DESCRIPTION

Ram air is used as a source of cooling air for the ECU, for tail cone pressurization, and for fresh ventilation air to the cockpit and cabin when the ECU is not operating during unpressurized flight. Ram air enters the dorsal scoop and is ducted into the tail cone area where it is drawn into the ECU heat exchanger (Figure 21-2).

After passing through the heat exchanger, air vents overboard through louvers in the lower right tail cone skin. If the ECU is not operating during unpressurized flight, this air also passes through a ventilation check valve in the tail cone, supplying air to the cockpit and cabin duct systems.

When the ECU is operating, conditioned airflow holds the ventilation check valve in the closed position.

NOTES

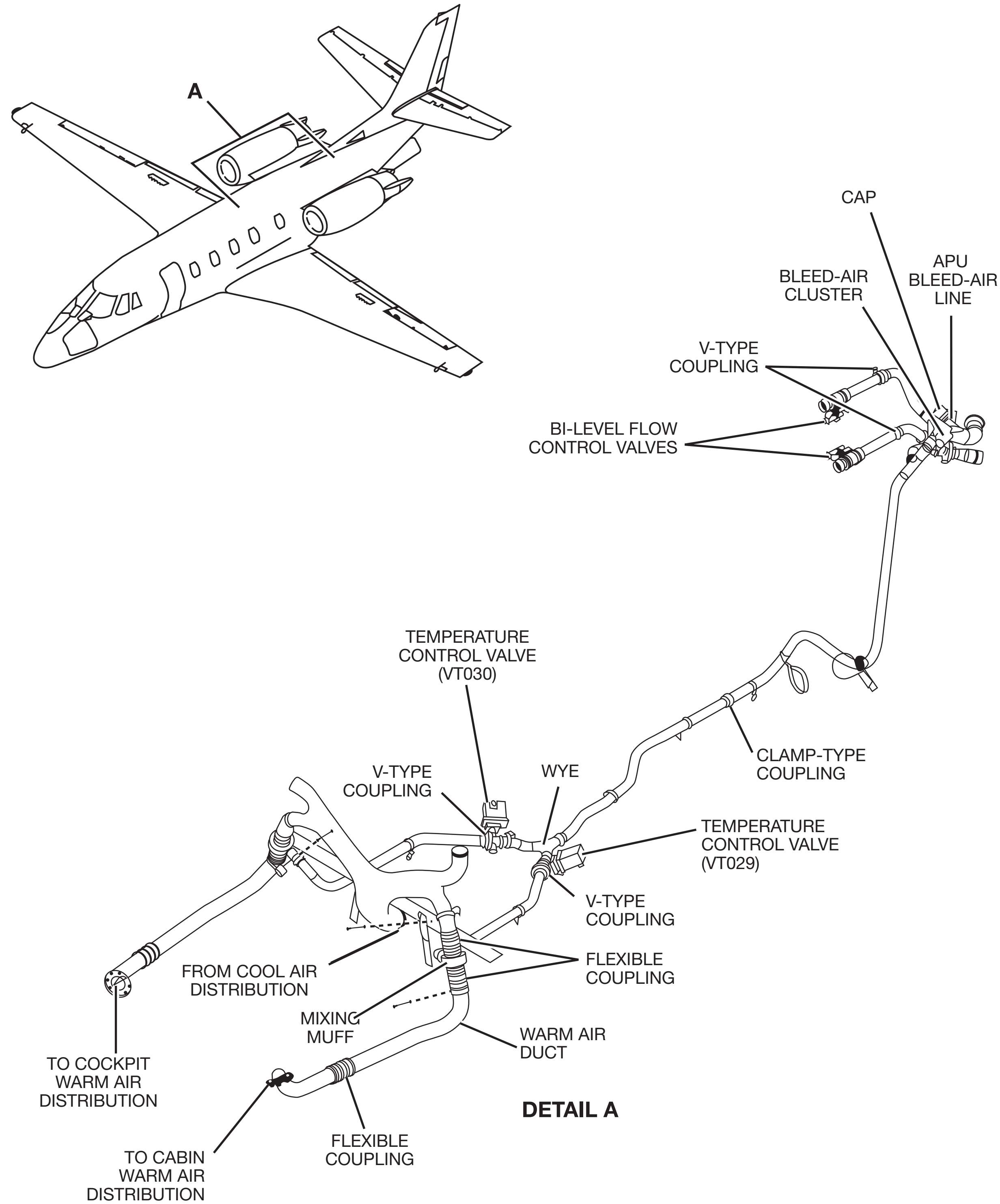


Figure 21-3. Tail Cone Hot Bleed Air

HOT BLEED AIR

DESCRIPTION

Anytime the engines are operating and the flow control valves are open, hot bleed air is distributed to the ECU and the temperature control valve (Figure 21-3). The bleed air is used for ECU operation and it is also routed through a single line under the baggage compartment floor where it branches off and goes to two of the temperature control valves. From the temperature control valves, the bleed air is routed to the mixing muffs, where it is mixed with cooled air from the ECU to obtain the desired temperature.

COMPONENTS

Flow Control Valves

The flow control valves include:

- A pressure regulation section
- A flow control nozzle
- A reverse check valve.

Regulated pressure is referenced to ambient to allow the flow to decrease with altitude. There is a shut off feature on the flow control valves that normally open in the absence of electrical power. The valve closes when electrical power is applied to the solenoid.

The source select switch allows the crew to select:

- Left engine only
- Right engine only
- Or NORM for both engines

In the EMER position, both valves are powered closed.

Mixing Muffs

Two mixing muffs in the tail cone are connected to the end of their respective temperature control valves.

The mixing muffs are devices that mix hot and cold air streams. The mixing muff surrounds the conditioned air duct and injects hot bleed air into the conditioned air duct. The quantity of bleed air to be mixed is controlled upstream by the temperature-control valve. The mixed air becomes temperature controlled conditioned air and is routed to the cabin and the cockpit.

NOTES

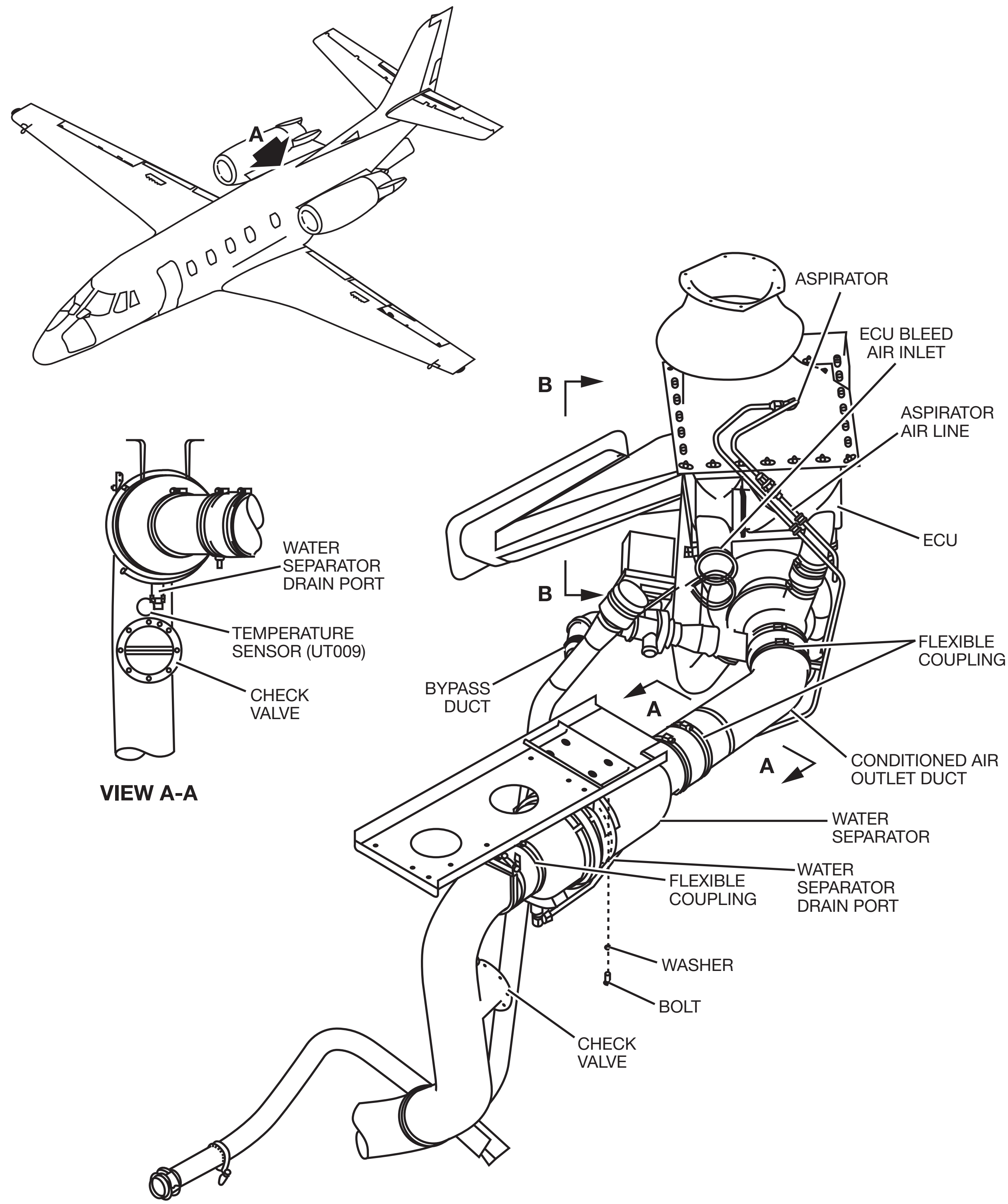


Figure 21-4. ECU Installation

ENVIRONMENTAL CONTROL UNIT

DESCRIPTION

The environmental control unit (ECU) utilizes bleed air from the engines for operation and provides cooling and pressurization for the cockpit and cabin (Figure 21-4).

The ECU consists of:

- Primary heat exchanger
- Secondary heat exchanger
- Air cycle machine (ACM) consists of compressor, turbine, and fan
- Water separator
- Water aspirator
- 37°F (2°C) temperature control system
- Over temperature switch (Figure 21-5)

COMPONENTS

Primary and Secondary Heat Exchangers

The primary and secondary heat exchangers are joined as a unit and arranged in parallel with the ram-air flow. The NACA scoops on the dorsal of the aircraft supply air for the heat exchangers. When the aircraft is on the ground and the NACA scoops are ineffective. A fan connected to the ACM turbomachinery draws air through the NACA scoops,and through the heat exchangers. Air is pumped overboard through a louvered duct on the right side of the tail cone. Fan inlet pressure is boosted by ram air in flight.

Engine bleed air passes through the primary heat exchanger and into the ACM compressor. The air is compressed and released through the secondary heat exchanger into the turbine. In the air cycle machine turbine, bleed air turns the turbine and expands rapidly to produce cold air. The cold air exits the ACM, passing through the water separator.

NOTES

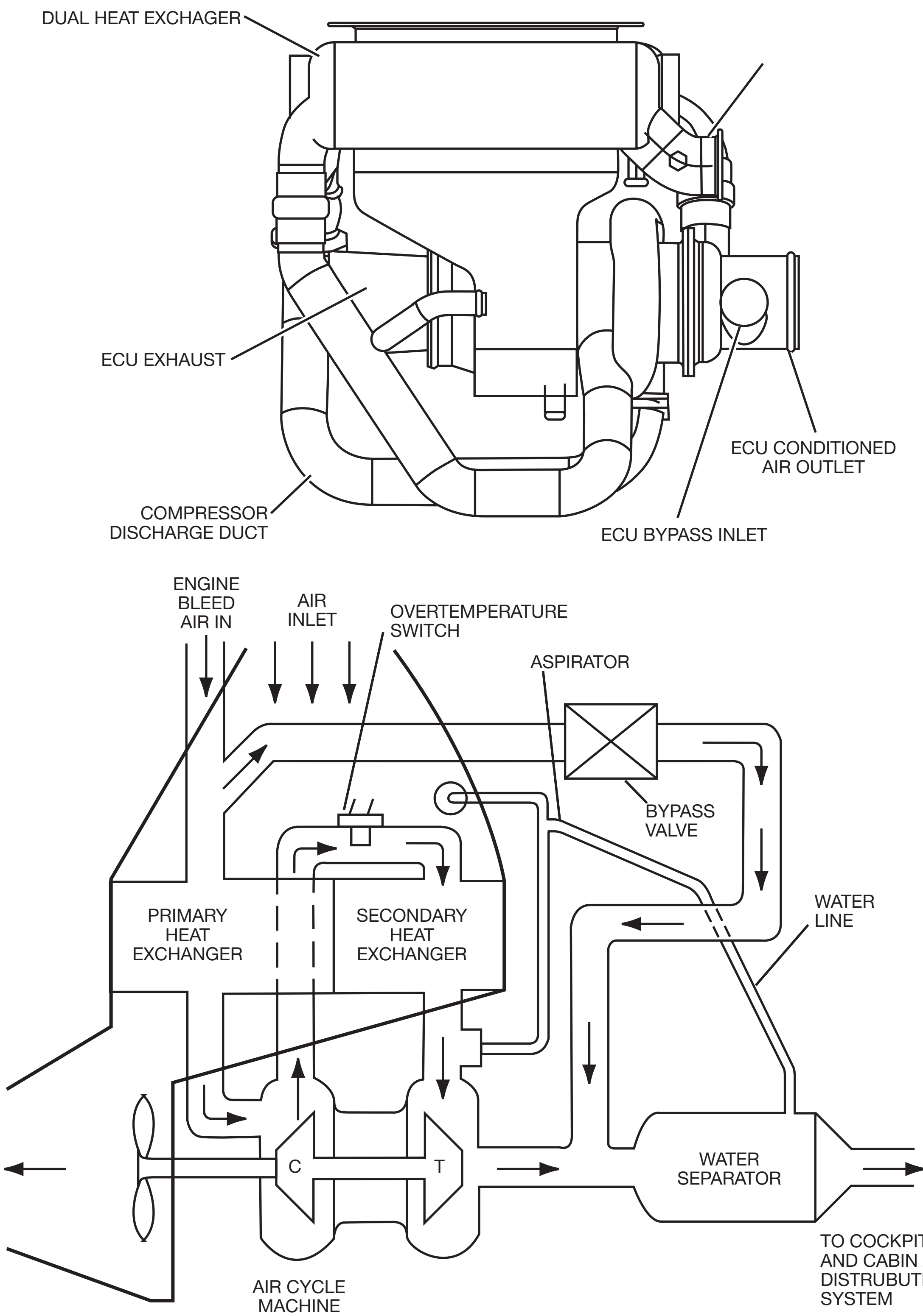


Figure 21-5. ECU Components

Air Cycle Machine

The ACM consists of a compressor, a turbine, and a fan, which are on a common shaft supported by air bearings, which do not require any oil as they ride on a film of compressed air (Figure 21-5). During the startup process, the shaft is supported by thin overlapping, spring-loaded foils, which keep the shaft centered. These foil segments grip the shaft with a preload so that the turbomachinery does not rotate freely by hand. The ACM requires approximately 5 psig to spool up the rotating equipment and bring the air bearings into effect. After the rotating equipment is spooled up, the shaft does not contact the foils. It stays supported by the air bearings down to approximately 1.5 psi.

Overtemperature Protection

The overtemperature switch consists of a temperature switch used in conjunction with a logic module and flow control valve.

The overtemperature switch is in the ECU compressor discharge outlet duct. It is a normally closed (NC) switch. Switch actuation is at 420 ± 10°F (216 ± 5.5°C). Deactuation is at 380°F (193°C). This switch senses the bleed-air temperature leaving the compressor portion of the ACM and protects the ECU from excessively high temperature.

In the event of an overtemperature indication, an electrical signal is sent to the pressurization logic module. This logic module closes the flow control valves, and stops all bleed-air flow to the ECU. If in flight, it opens the emergency pressurization valve and illuminates both the EMER PRESS and ACM O'HEAT annunciators. When the switch senses an overheat condition on the ground, it closes the flow control valves and illuminates the EMER PRESS and ACM O'HEAT annunciator, but the emergency pressurization valve does not open due to the left squat switch. When the temperature drops to an acceptable level, the logic module reopens the flow control valves.

NOTE

A wire modification on aircraft that have complied with SB560XL-21-19 and SNs 5603 and subsequent prevent emergency pressurization at cabin altitudes that are less than 14,500 feet.

ACM O'HEAT	**ACM OVERHEAT** Annunciator flashes to indicate the ACM has overheated and automatically shut down. EMER PRESS automatically activates (AD configured aircraft only). Activates MASTER CAUTION lights.
EMER PRESS	**EMERGENCY PRESSURIZATION** Annunciator flashes to indicate pressurization is active. The system can be manually or automatically activated.

XL/XLS ANNUNCATORS

ACM OVERTEMP			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the ACM has overheated. When the ACM is too hot, a 28V signal is sent to the EICAS, which posts the message. When the ACM is normal temperature, an open signal is sent to the EICAS, which removes the message.			

EMERGENCY PRESSURIZATION			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when emergency pressurization is active. When emergency pressurization is active, 28V is provided to the emergency pressurization valve to provide additional inflow into the cabin. This 28V signal is also sent to the EICAS system. When the input is 28V, the message is displayed. When the input is open, the message is not displayed. The EICAS system also provides a ground/open output which is used by the audio attentuation PC board. When the emergency pressurization input is 28V, the output is ground. When the input is open, the output is open.			

XLS+ CAS MESSAGES

Figure 21-6. Overtemperature Indications

21 AIR CONDITIONING

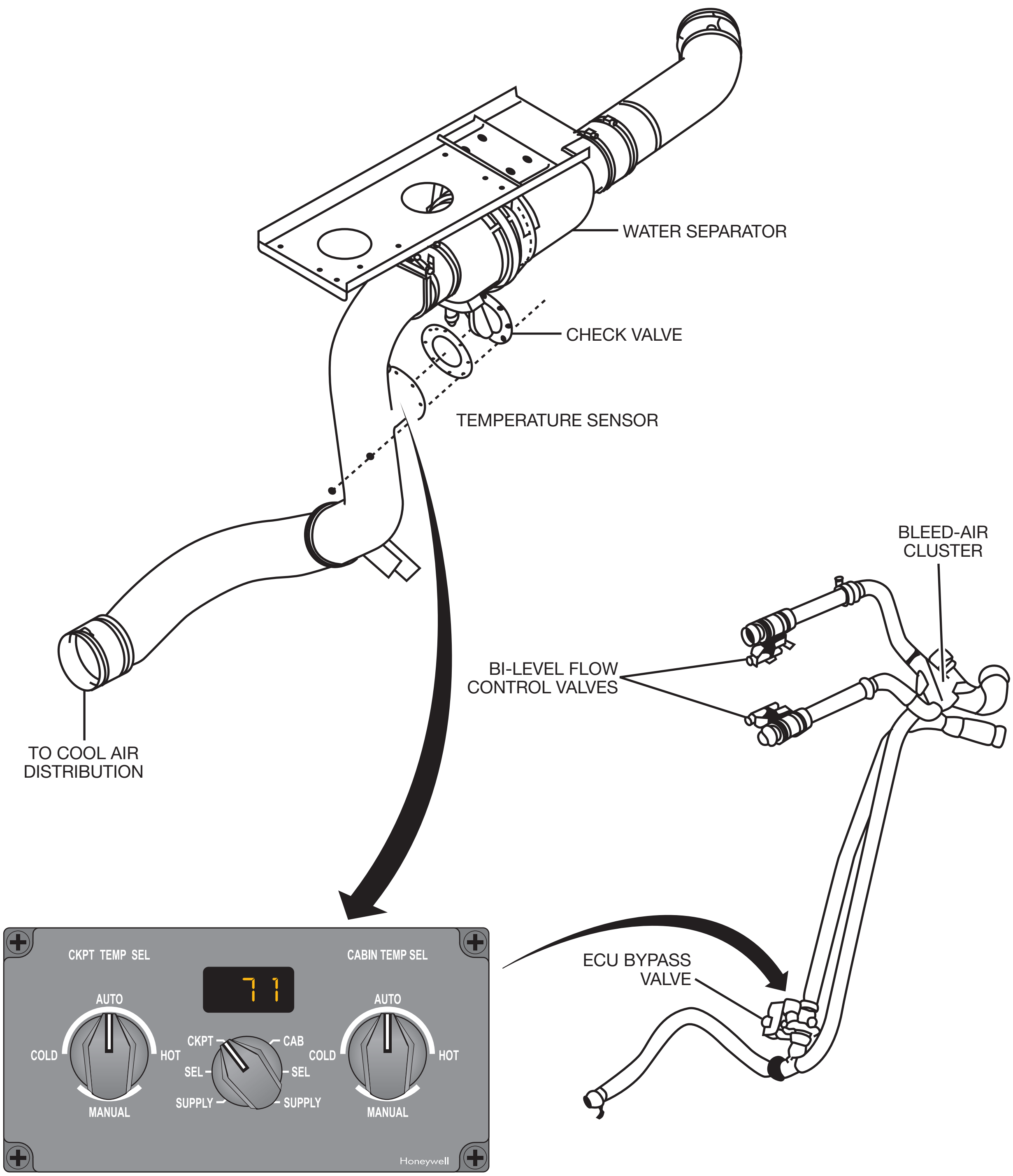


Figure 21-7. Low Temperature Control System

NOTE

The EMERGENCY PRESSURIZATION CAS message on the XLS+ does not illuminate unless power is applied to the emergency pressurization flow control valve.

Water Separator

The water separator is comprised of a coalescer to collect moisture, and a bypass relief valve to permit air to bypass the water separator if the coalescer becomes clogged or frozen over.

Cool air from the ACM turbine is ducted to the water separator. The moisture is collected on the coalescer and routed via drain tubes to the water aspirator on the ACM inlet duct. Dehumidified cool air from the water separator is distributed to the four-way transition duct.

Water Aspirator

The water aspirator on the ACM inlet duct provides a vacuum for removing water from the water separator, using pressurized air from the ACM turbine inlet. The water is ejected onto the secondary heat exchanger.

CONTROLS AND INDICATIONS

Low Temperature Control

The 37°F (2°C) temperature control system consists of:

- Controller
- Temperature sensor
- Bypass valve

The controller on the instrument panel (Figure 21-7) provides an interface between temperature switch and bypass valve.

The temperature sensor is directly downstream of the water separator (near the ram-air/fresh-air check valve). It monitors ECU cool air temperature. If the temperature falls below 37°F (2°C), an electrical signal is sent to the temperature controller. The controller then actuates the bypass valve.

The bypass valve between the bleed-air duct and the ECU outlet consists of an inline valve and motor. The bypass valve operates in conjunction with the ECU controller and inline temperature sensor. When the inline temperature sensor detects temperature below 37°F (2°C), the motor actuated bypass valve opens, allowing a variable amount of hot bleed air to mix with ECU cool air. This mixing ensures cool air temperature does not fall below 37°F (2°C). By doing so, it prevents water from freezing in the water separator.

NOTES

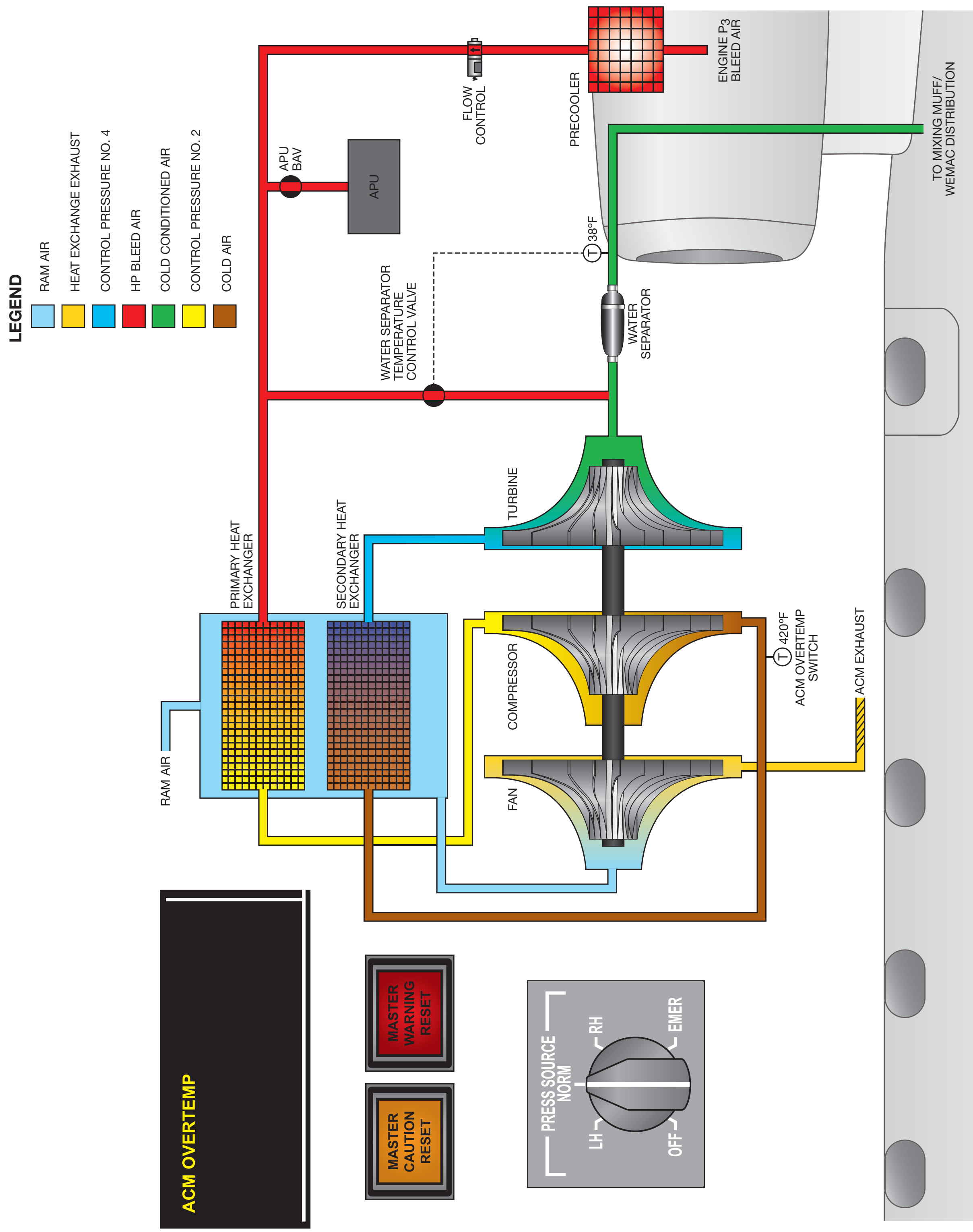


Figure 21-8. ECU Operation

OPERATION

Precooled (475 ± 25°F) engine bleed air enters the ECU from two wyed together flow control valves (Figure 21-8). The flow control valves drop the pressure of the bleed air and control system. The system flows approximately 20 lbm/min at sea level and 12 lbm/min at FL 450. Bleed air enters the ECU at the primary heat exchanger and is cooled by the primary heat exchanger to approximately 200–300°F, before entering the ACM. The compressor is the first stage of the ACM. The compression process raises the temperature of the bleed air to approximately 300–400°F.

The bleed air then exits the ACM and enters the secondary heat exchanger. The air is cooled in the secondary heat exchanger to approximately 100–150°F. The bleed air then re-enters the ACM and is expanded across a nozzle onto a turbine wheel. The expansion process extracts energy from the air, which is used to drive the compressor as well as the fan that draws the ambient air through the primary and secondary heat exchangers. This expansion process cools the air to approximately 40-50°F on a hot day.

On moderately cold days, the turbine outlet temperatures drop well below freezing. When the outlet temperature of the turbine drops below the dewpoint of the ambient air, the entrained water vapor is condensed out of the air in liquid form. When the outlet temperatures are below freezing, these water droplets freeze and create ice particles. To prevent these ice particles from freezing over the water separator and blocking airflow, the cold turbine outlet air is mixed with hot bleed air. The hot bleed air is modulated by the 37°F low-limit temperature control valve to obtain a temperature between 32°F and 37°F downstream of the water separator.

After the air exits the water separator, it is routed forward toward the temperature control ducting. The air out of the ECU is always controlled to obtain 32°F to 37°F at the outlet of the water separator. To obtain heat for the cabin, bleed air is bypassed around the ECU and mixed with the cooled air to obtain the proper temperature.

NOTES

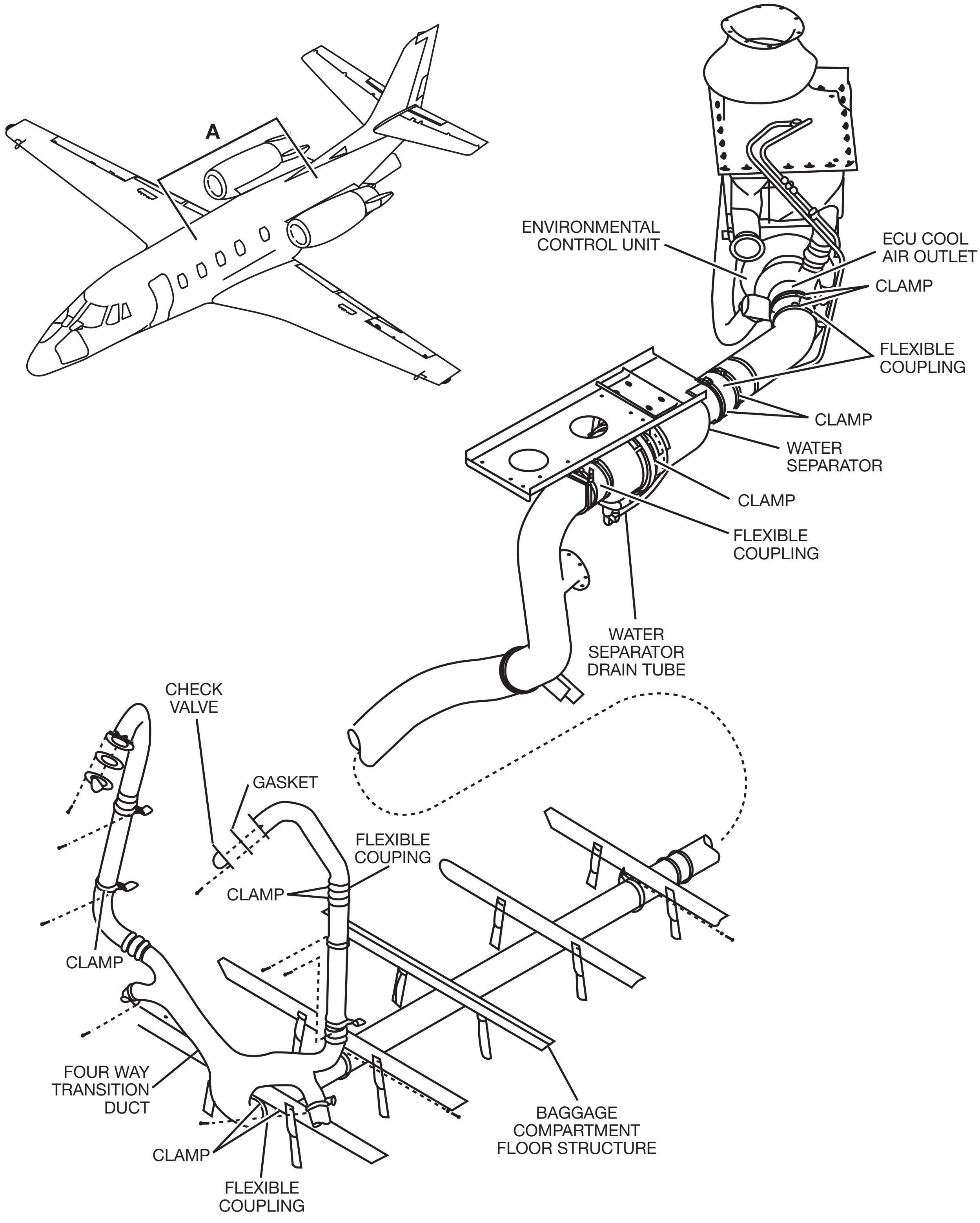


Figure 21-9. Tail Cone Cool Air Distribution

TAIL CONE COOL AIR DISTRIBUTION

DESCRIPTION

This section describes the distribution of cool air from the point where it leaves the ECU until termination in the overhead cockpit vents (Figure 21-9). The aircraft uses one ECU to provide a system of cool air distribution, which runs overhead from the tail cone to the cockpit area.

Cool air leaving the ECU water separator is plumbed through a single line to just behind the aft pressure bulkhead. At this point it connects to a four-way transition duct in the tail cone. Half of the cool air is routed downward and mixed with hot, modulated bleed air to provide warm air for the:

- Cabin armrests
- Cabin floor system
- Cockpit armrests
- Foot warmers
- Ventilation outlets

The other half of the cool air is divided into left and right ducts. The left and right ducts are routed upward and penetrate the aft pressure bulkhead. From there the cool air is distributed into the cabin and cockpit compartments.

COMPONENTS

Bulkhead Check Valves

There are four check valves in the cabin/cockpit environmental system. Two are at the top of the aft pressure bulkhead and two at the lower portion of the aft cabin. The check valve is a dual-flapper spring-loaded closed valve. The check valve permits conditioned and warm air to flow into the cabin air distribution system without losing cabin pressurization in the event of a duct failure.

NOTES

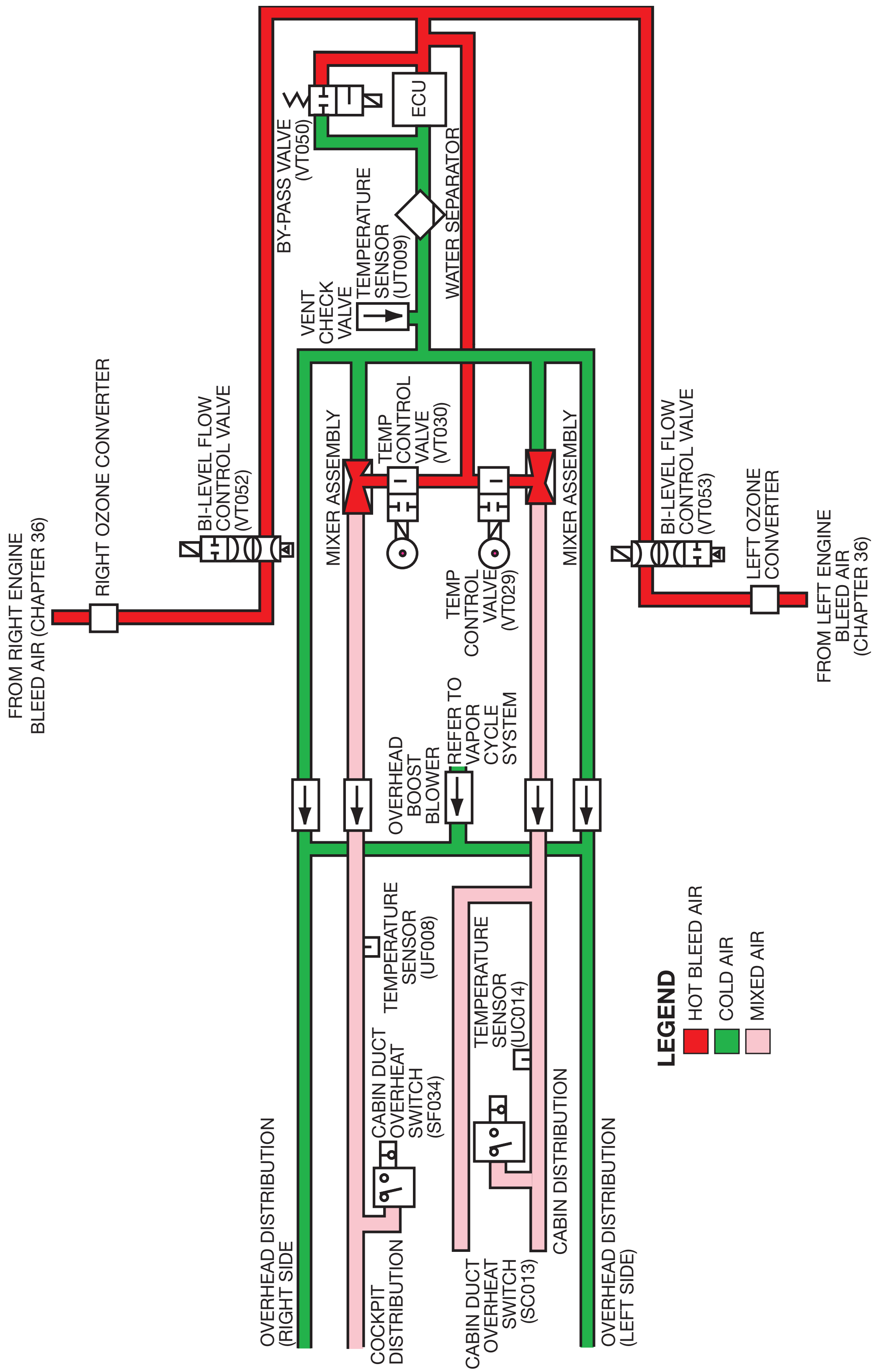


Figure 21-10. Cabin/Cockpit Air Distribution Diagram

CABIN/COCKPIT AIR DISTRIBUTION

DESCRIPTION

The cabin and cockpit air distribution systems direct the flow of fresh/temperature conditioned air to provide a comfortable and adequately ventilated cabin and cockpit (Figure 21-10).

There are three air distribution networks:

- Overhead cold air distribution
- Lower cabin air distribution
- Cockpit air distribution

Half of the ECU air (approximately 37°F (2°C)) is distributed directly overhead throughout the cabin and cockpit areas. The other half mixes with hot bleed air via a mixer assembly (mixing muff) to produce warm air. The warm air is distributed under the cabin floor to either the cabin or cockpit distribution system.

The temperature control panel assembly is on the copilot tilt panel.

NOTES

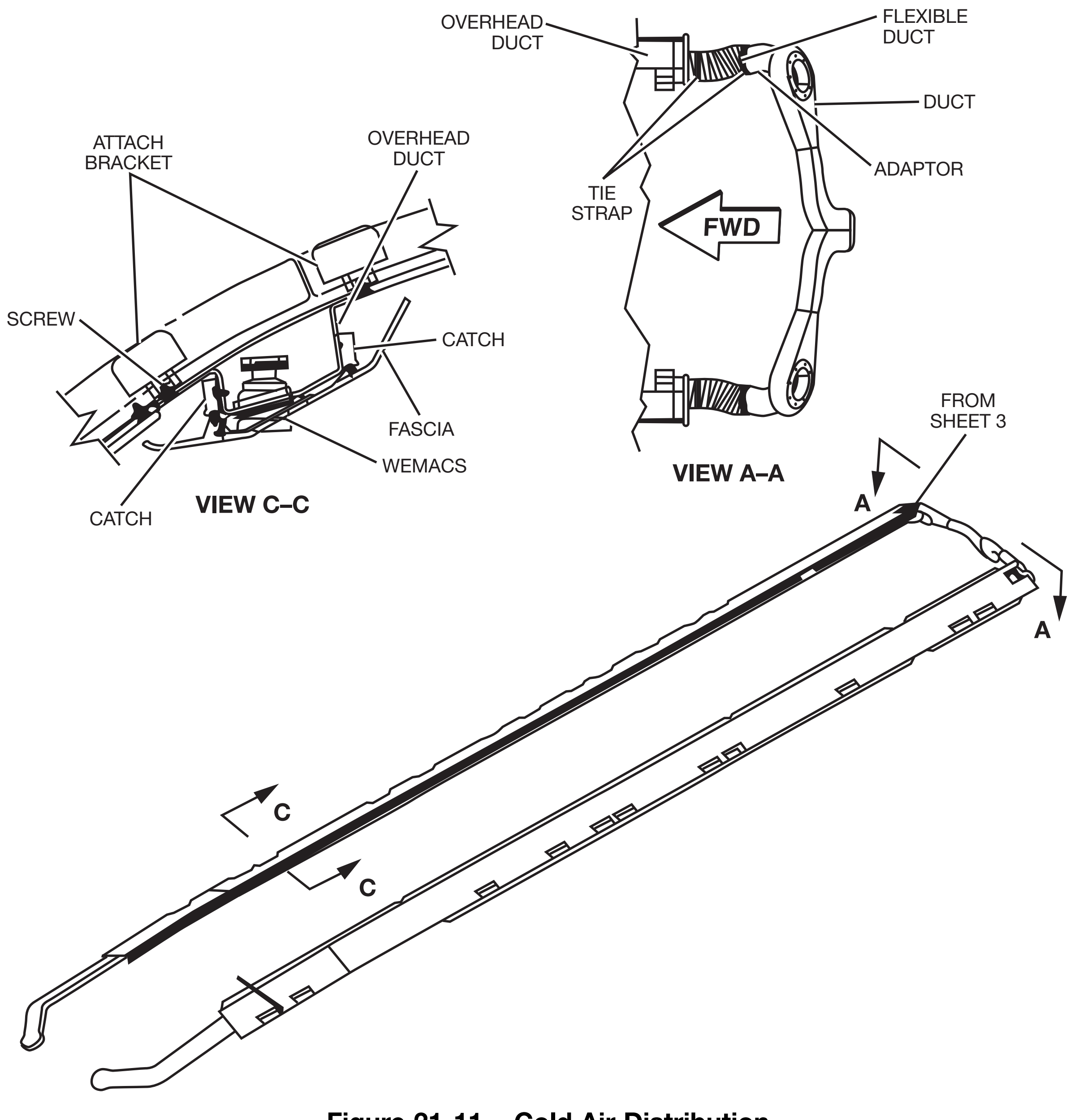


Figure 21-11. Cold Air Distribution

Overhead Cold Air Distribution

The overhead cold air distribution originates from the ECU (Figure 21-11). Left and right cold air distribution supplies the cockpit and cabin through overhead Wemac air outlets. An outlet is provided for each passenger seat, pilot and copilot positions. The outlets are individually operated from full open to a full closed position. These outlets are installed along two fore and aft overhead ducts that extend from the cockpit area to the aft pressure bulkhead. These overhead ducts are continuously pressurized with cold air when the engines are running and bleed air is supplied.

At the aft pressure bulkhead, each cool air distribution duct contains a check valve, which prevents reverse flow in the system. From the check valve forward, cool air routes overhead through the cabin using both flexible and formed ducting. Wemac air outlets positioned overhead distribute the cool air. Formed cabin ducting terminates on both left and right sides near the cabin entry door.

Flexible ducting connects the cabin portion and the cockpit portion of the system together. Each distribution line (left and right) terminates in an overhead Wemac outlet above the flight crew.

NOTES

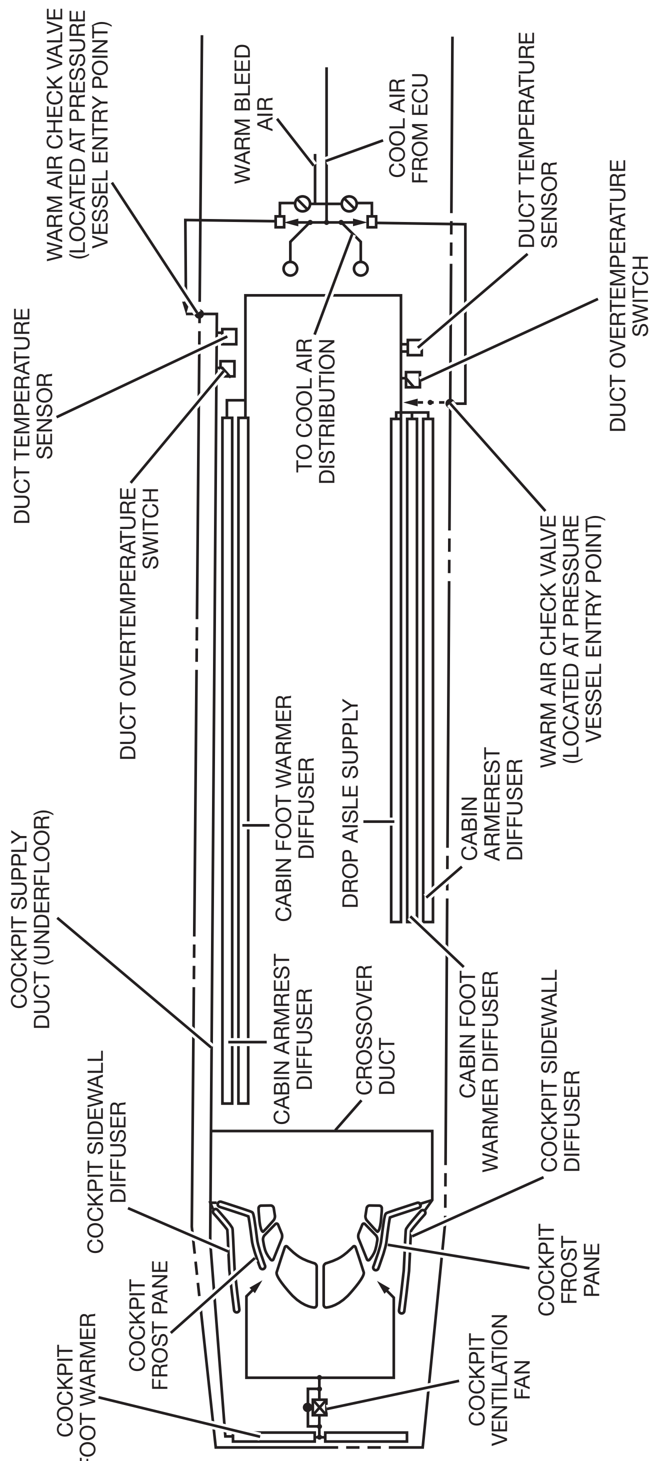


Figure 21-12. Warm Air Distribution Diagram

Warm Air Distribution

This section describes the distribution of warm air from the tail cone to the cockpit. Warm air is created when conditioned, cool air from the ECU is mixed with an amount of hot bleed air to modulate air temperature (Figure 21-12). This warm air is then distributed to the cabin and cockpit areas via a series of ducts, hoses, and valves.

From the mixing muffs, warm air for each system (cabin and cockpit) exits the tail cone (approximately FS 381.72) and is routed through fuselage fairings. The cabin system is routed through left fuselage fairings, and the cockpit system is routed through right fuselage fairings. Both systems enter the pressure vessel at approximately FS 339.01 to the left and right of BL 0.00. Two check valves (one per side) are at these entry points.

NOTES

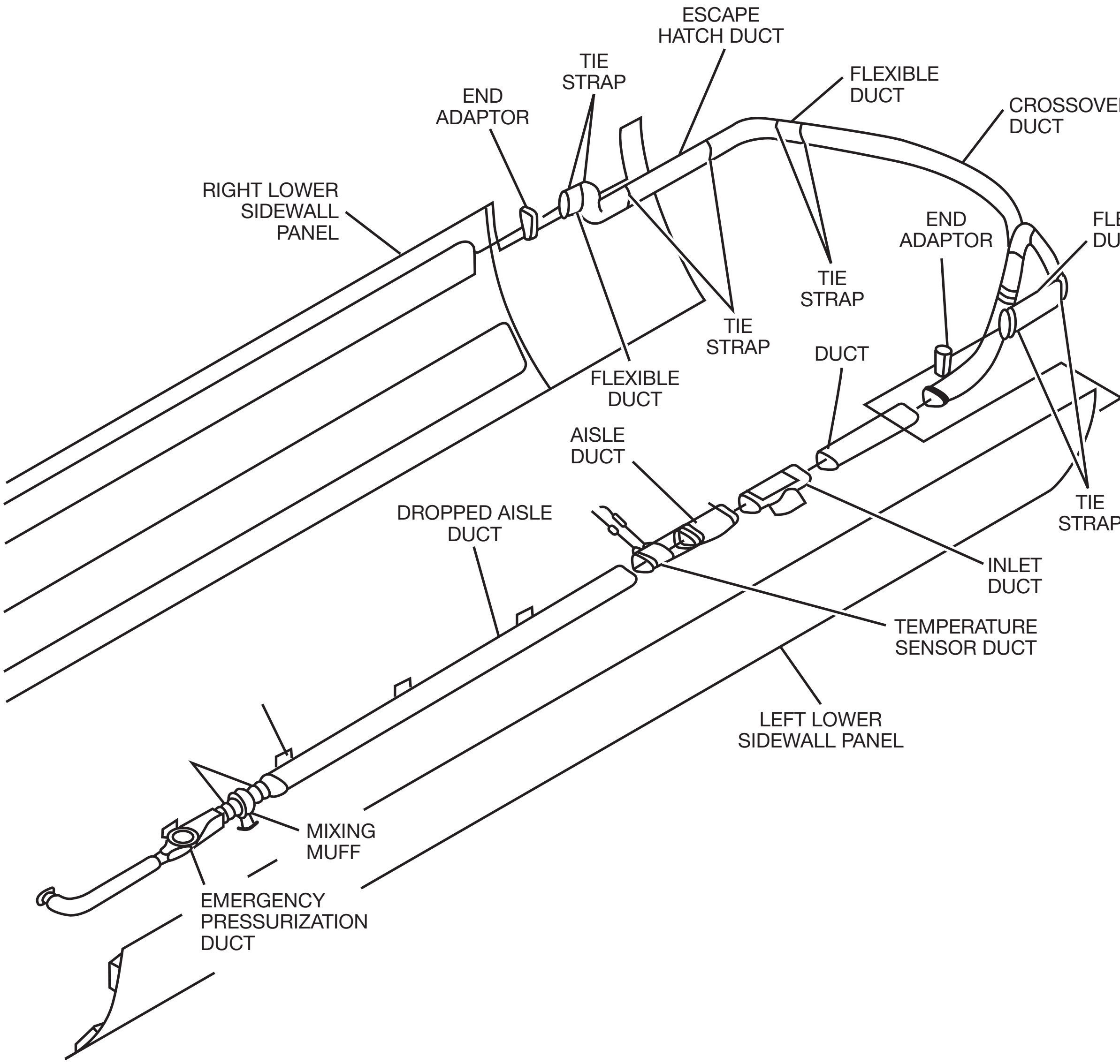


Figure 21-13. Cabin Air Distribution System

CABIN AIR

DESCRIPTION

From the check valve forward, cabin warm air is routed to both the left and right sides of the cabin(Figure 21-13).

On the left side of the cabin, warm air is routed underneath the floorboards for dropped aisle heating. It is routed through interior shell assemblies (side wall ducting) to integral foot warmer and armrest diffuser outlets.

On the right side of the cabin, warm air is routed through interior shell assemblies (side wall ducting) to integral foot warmer and arm rest diffuser outlets.

Lower Cabin Air Distribution

The left lower supply duct supplies air for the lower cabin air distribution system.

When the air enters the cabin it splits into several paths directing air towards the main paths:

- Left armrest and footwarmer ducts
- Dropped aisle ducts on the left side of the dropped aisle
- Right armrest and footwarmer ducts

The footwarmer and armrest ducts are a piccolo tube design that allows air to flow evenly over the length of the cabin.

Cabin Air Ducting

The cabin sidewall is a bond assembly with two ducts (referred to as footwarmer and armrest ducts). The ducting incorporates spray holes to discharge cabin environmental air into the cabin. The left side ducting extends from the aft cabin forward to the main entrance door. The right side ducting extends from the aft cabin forward to just aft of the cabin/cockpit divider. A wyed crossover duct on the forward side of the aft pressure bulkhead connects the left and right sidewall ducts to each other and to the duct under the floor.

NOTES

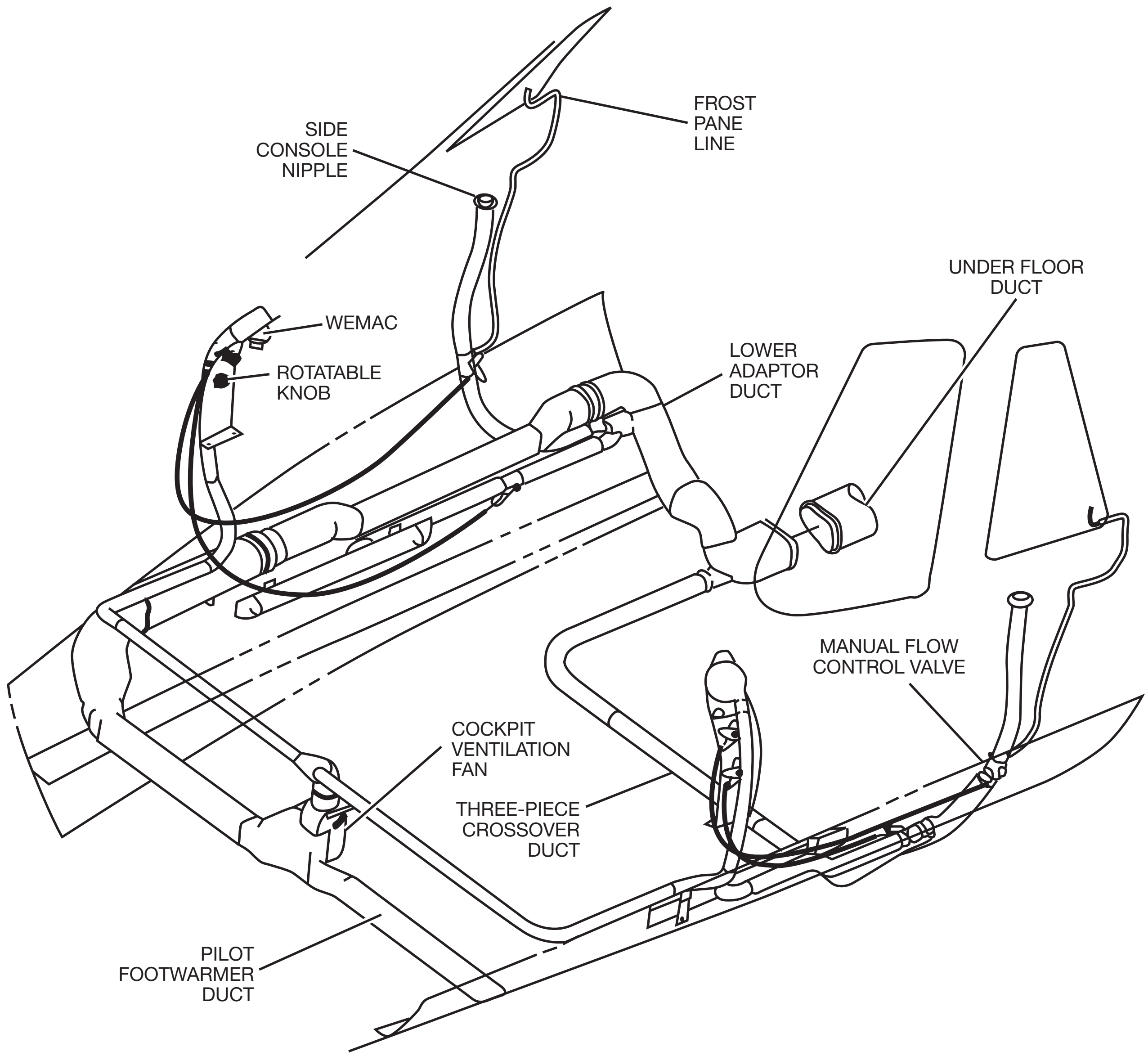


Figure 21-14. Cockpit Air Distribution System

COCKPIT AIR

DESCRIPTION

The cockpit air distribution system is supplied with conditioned air through the right lower supply duct (Figure 21-14). Air enters the aft cabin and is ducted underneath the right seats toward the cockpit. After reaching the cockpit, the air splits off into:

- Sidewall diffusers
- Side window
- Defog diffusers
- Forward bulkhead diffusers

The cockpit vent system consists of side console air outlets footwarmers and console WEMACs which are supplied by a fan in the footwarmer ducting (Figures 12-15, 12-16, and 12-17). The side console air outlets are on the top surface of the pilot and copilot side consoles. The right console air outlet connects to the cockpit air duct (inside the console). A crossover duct, extending from the copilot console, supplies air to the left console, and follows the lower fuselage contour. The side console air outlets are opened and closed by rotating the nozzle. Left and right footwarmer outlets are in the forward crossover ducting.

The side console WEMACS are supplied with air pulled from the cockpit supply and mixed with recirculated air, pulled in from the footwarmer ducts. Air is supplied through flex ducts connected to a center outlet, between the left and right footwarmers on the forward ducting. The console WEMAC fan is in the center of the forward footwarmer duct. The cockpit recirculation fan is controlled electrically by the CKPT RECIRC fan switch on the copilot lower right instrument panel.

Condensation on the cockpit side windows is prevented by using frost panels to prevent moist cockpit air from coming in contact with cold outer window surface. Conditioned air from the cockpit supply is fed between the panels from the bottom of the window. There is a small vent hole placed in the upper corner of the frost pane to allow the air to flow over the pane and into the cockpit.

Figure 21-15. Side Console Vent

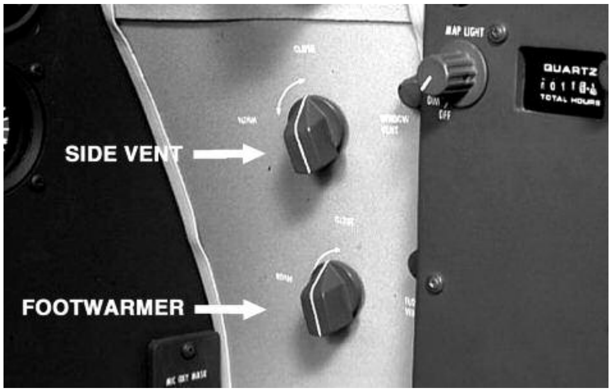


Figure 21-16. Side Console Vent Knobs (XL Only)

Figure 21-17. Forward Cockpit Wemac Vents

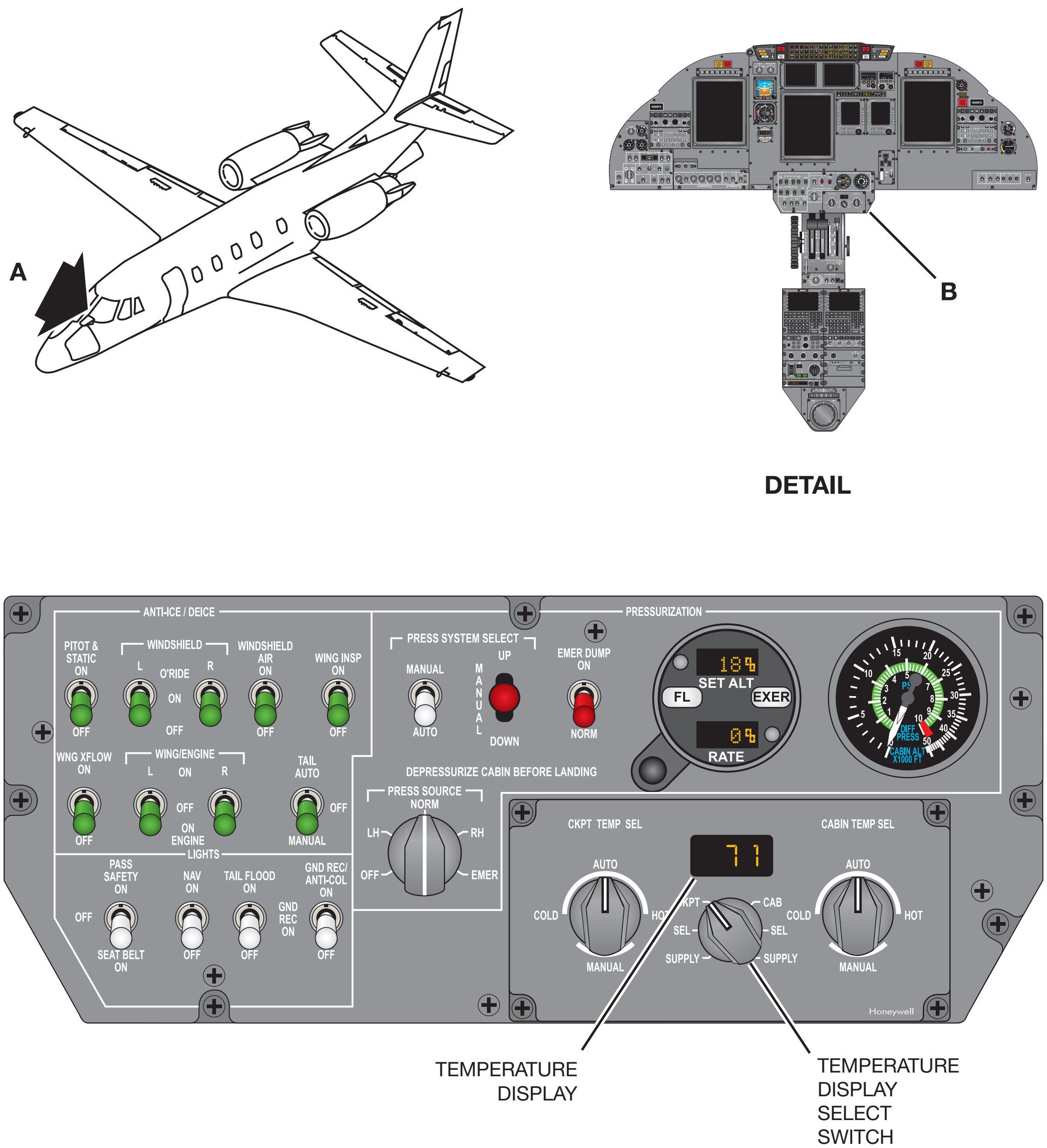


Figure 21-18. Cockpit Tilt Panel

TEMPERATURE CONTROL

DESCRIPTION

This section describes with temperature control systems for cabin and cockpit conditioned air. The temperature controls for cabin and cockpit conditioned air are separate and individually controlled. The controls are on the cockpit tilt panel (Figure 21-18).

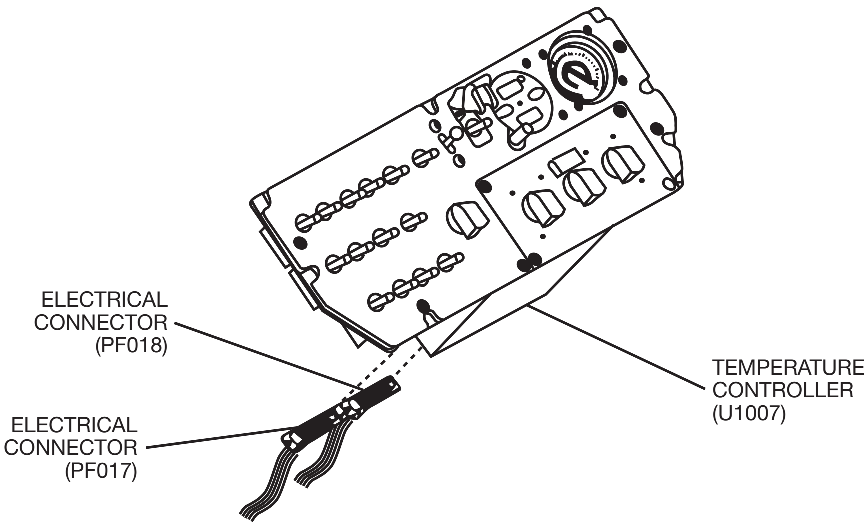
Temperature is controlled in the cabin and cockpit by mixing constant-temperature cool air (approximately 37°F (2°C) as it leaves the ECU with variable temperature warm air. Controls and valves vary temperature in the warm air distribution system to alter overall cabin/cockpit temperature as it mixes with the cool air distribution system.

Components common to both the cabin and cockpit temperature control system include:

- Temperature control valves
- Temperature controller
- Mixing muffs
- Temperature sensors
- Zone sensors
- Duct overheat switch
- Tail cone ducting

A temperature controller is part of the cockpit environmental control panel, and provides flight crew with temperature control of cabin/cockpit systems.

NOTES



DETAIL B

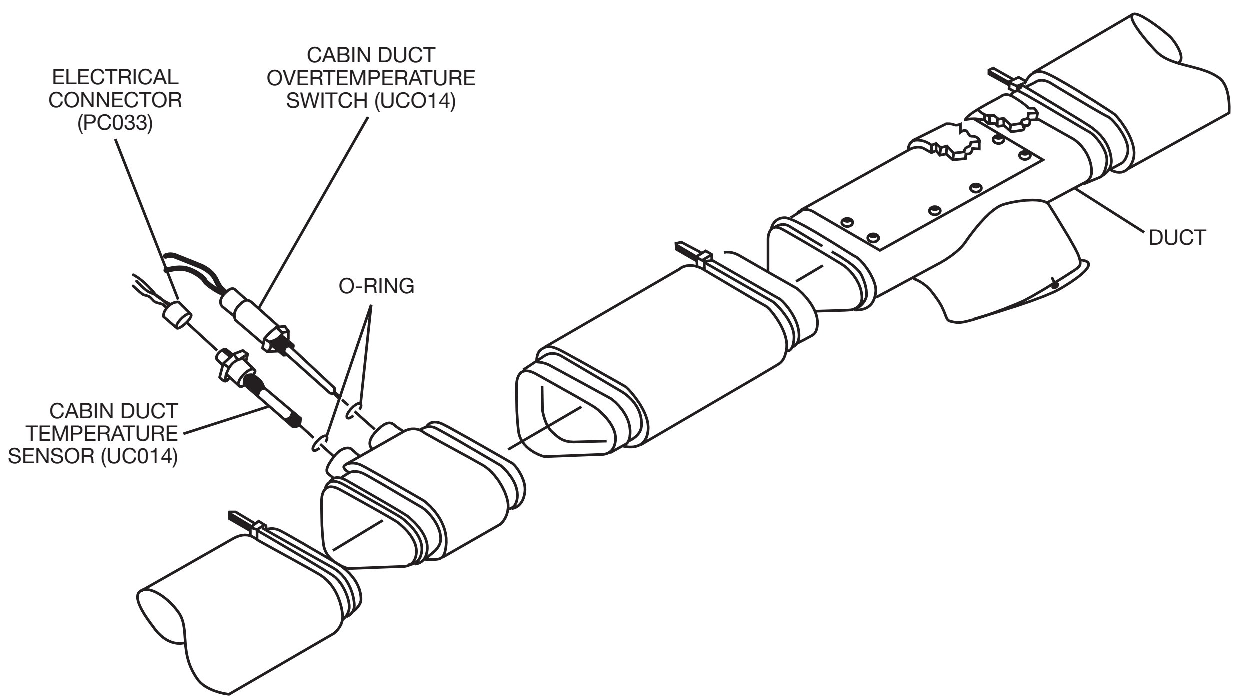


Figure 21-19. Temp Control Sensors and Switches

COMPONENTS

Duct Temperature Sensors

There are two duct temperature sensors under the cabin floorboards at approximately FS 339.01 (just downstream of the pressure vessel check valves) (Figure 21-19). Each sensor monitors the temperature of warm air as it enters the respective cabin or cockpit air distribution system. Each sensor sends this information to the temperature controller.

The left duct temperature sensor is connected to the cabin warm air distribution system and is under the left side cabin floorboards.

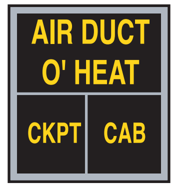
The right duct temperature sensor is connected to the cockpit warm air distribution system and is under the right side cabin floorboards.

Duct Overheat Switches

There are two duct overheat switches under the cabin floorboards (one next to each of the duct temperature sensors). These switches are connected to the annunciator panel and the MASTER CAUTION light to give the flight crew a visual indication of an overheat condition.

The left overheat switch is on the left side of the cabin and indicates cabin air overheat conditions when temperature exceeds 300°F (149°C) with the illumination of the AIR DUCT O'HEAT CAB annunciator (XL/XLS) or CABIN AIR DUCT OVERTEMP CAS message (XLS+) (Figure 21-20).

The right overheat switch is on the right side of the cabin and indicates cockpit air overheat conditions when temperature exceeds 300°F (149°C) with the illumination of the AIR DUCT O'HEAT CKPT annunciator (XL/XLS) or COCKPIT AIR DUCT OVERTEMP CAS message (XLS+) (Figure 21-20).



AIR DUCT O'HEAT
Annunciator flashes if air in the cockpit duct and/or cabin duct has exceeded temperature limits, activates MASTER CAUTION lights.

XL/XLS ANNUNCATORS

CABIN AIR DUCT OVERTEMP			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the supply air in the cabin air duct is too hot. A temperature switch in the supply duct provides a ground signal to the EICAS, which posts the message. When the supply temperature is normal, the switch provides an open to the EICAS, which removes the message.			

COCKPIT AIR DUCT OVERTEMP			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the supply air in the cockpit air duct is too hot. A temperature switch in the supply duct provides a ground signal to the EICAS, which posts the message. When the supply temperature is normal, the switch provides an open to the EICAS, which removes the message.			

XLS+ CAS MESSAGES

Figure 21-20. Duct Overheat Indications

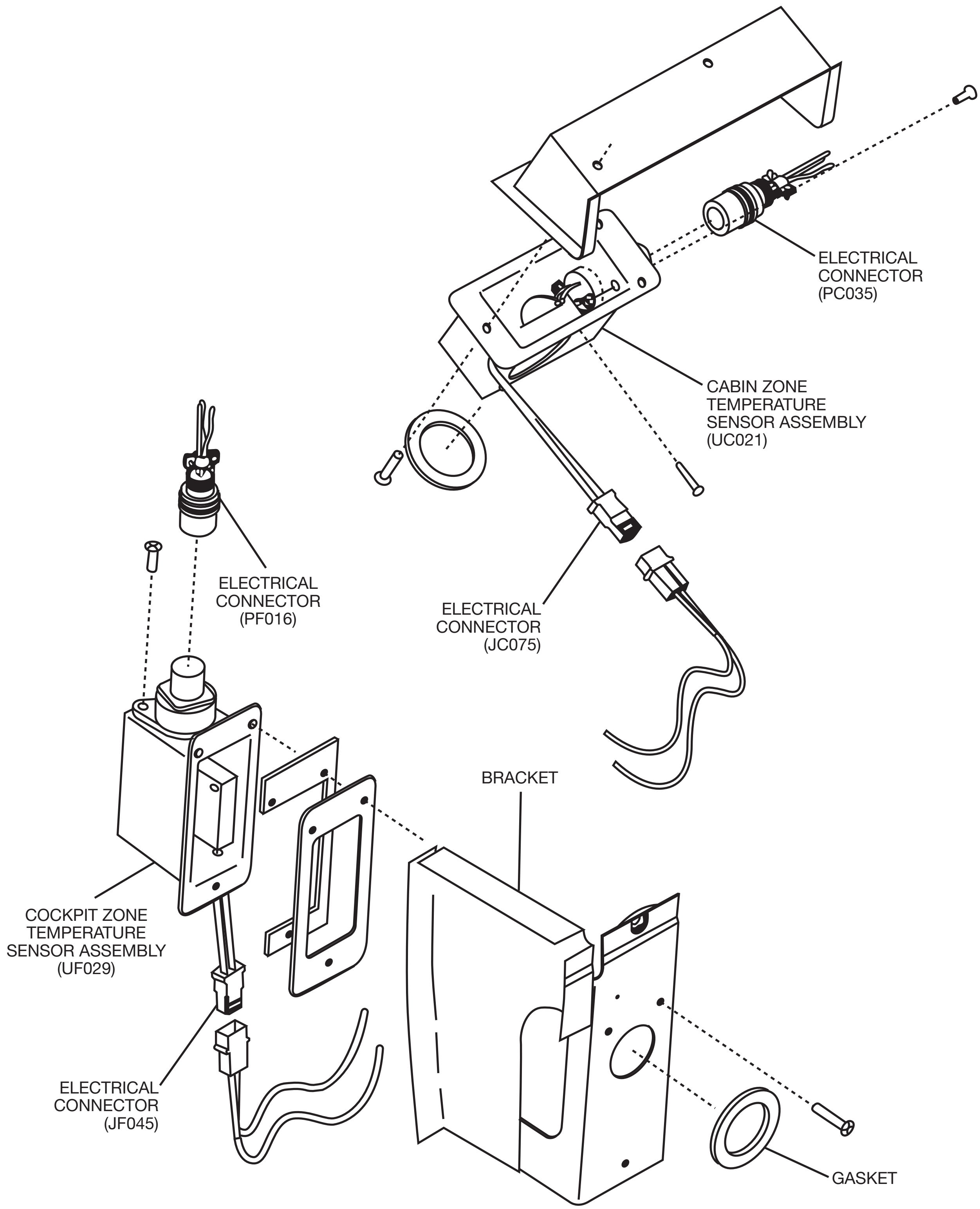


Figure 21-21. Zone Temperature Sensors

Zone Temperature Sensors

There are two zone temperature sensors in the temperature system: one in the cockpit and the other in the cabin. The cockpit zone temperature sensor is in the right side console. The cabin zone temperature sensor is in the aft, left side of the passenger service unit behind the fascia panel (XL) or on the lower sidewall between the aft passenger seat on the left side of the cabin (XLS/XLS+) (Figure 21-21). The sensor assembly is comprised of a fan and sensor (in a single box). The fan draws cabin/cockpit air across the sensor to provide a more representative zone temperature. The sensors monitor the temperature of the air in the cockpit and cabin and provide a reference temperature to the temperature controller.

Temperature Control Valves

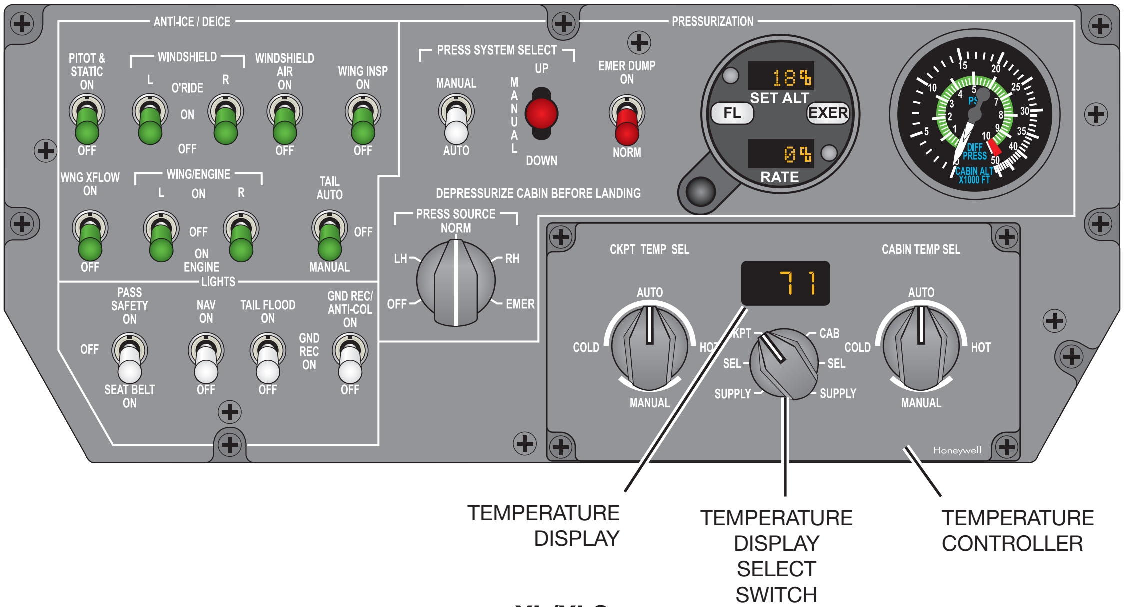
Three temperature control valves regulate the amount of hot bleed air, mixed in with cold air out of the ECU (see Figure 21-18). There are two temperature control valves are just aft of the aft pressure bulkhead and a bypass valve by the ECU. The valves are butterfly type valves, controlled by brushless DC motors. The DC motor of each valve receives pulses of power from the controller to position the butterfly, modulating the flow of bleed air to the mixing muffs. By varying the amount of hot bleed air mixed with ECU cool air [approximately 37°F (2°C)], cabin/cockpit temperature is controlled as it enters respective warm air distribution lines.

The low temperature control valve (bypass valve) by the ECU regulates the temperature of the air to the water separator [37°F (2°C)].

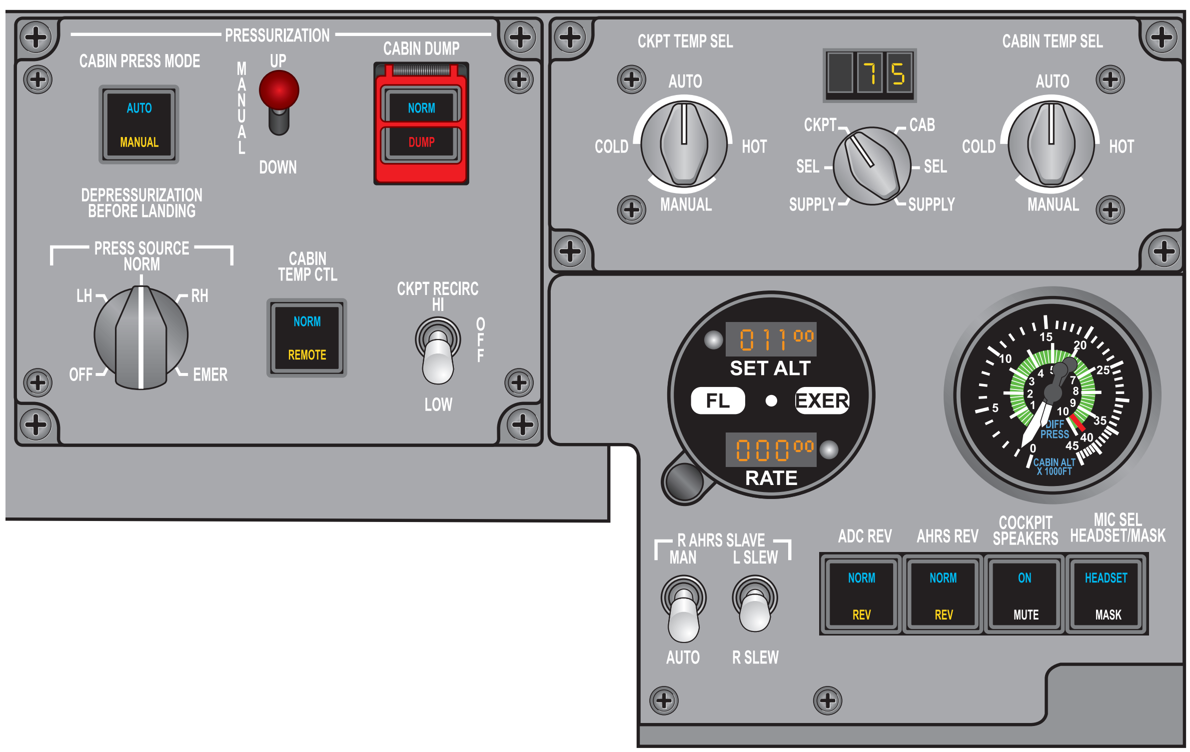
The left temperature control valve connects to the cabin warm air distribution system.

The right temperature control valve connects to the cockpit warm air distribution system.

NOTES



XL/XLS



XLS+

Figure 21-22. Temperature Control Panel

CONTROLS AND INDICATIONS

Temperature Controller

A single temperature controller is on the tilt panel in the cockpit (Figure 21-22). All logic for the temperature control system is contained in the temperature controller and selector-indicator portion of the environmental control panel. The cabin and cockpit logic receives input signals from their respective zone sensors and supply sensors. The temperature controller sends a DC current signal to the temperature control valves to modulate the control valves toward the open or closed position as required to attain the selected temperature.

Temperature Control Selector-Indicator

A single temperature control selector-indicator is integrated into the temperature controller. The temperature selector-indicator consists of:

- Cabin temperature selector
- Cockpit temperature selector
- Digital temperature indicator
- Display selector

The cabin temperature selector (CABIN TEMP SEL) and the cockpit temperature selector (CKPT TEMP SEL), rotary switches, incorporate both automatic and manual mode controls. The temperature indicator provides a digital temperature readout of the selected switch position. Switch positions provide temperature readouts of cabin zone temperature (CAB) and cockpit zone temperature (CKPT). The SUPPLY positions display the cabin supply duct temperature and the cockpit supply duct temperature. The SEL positions display the selected temperature to which the cabin and cockpit are being controlled. The temperature controller performs system diagnostics each time power is applied to the controller. The diagnostics identify and report nine potential error conditions.

NOTES

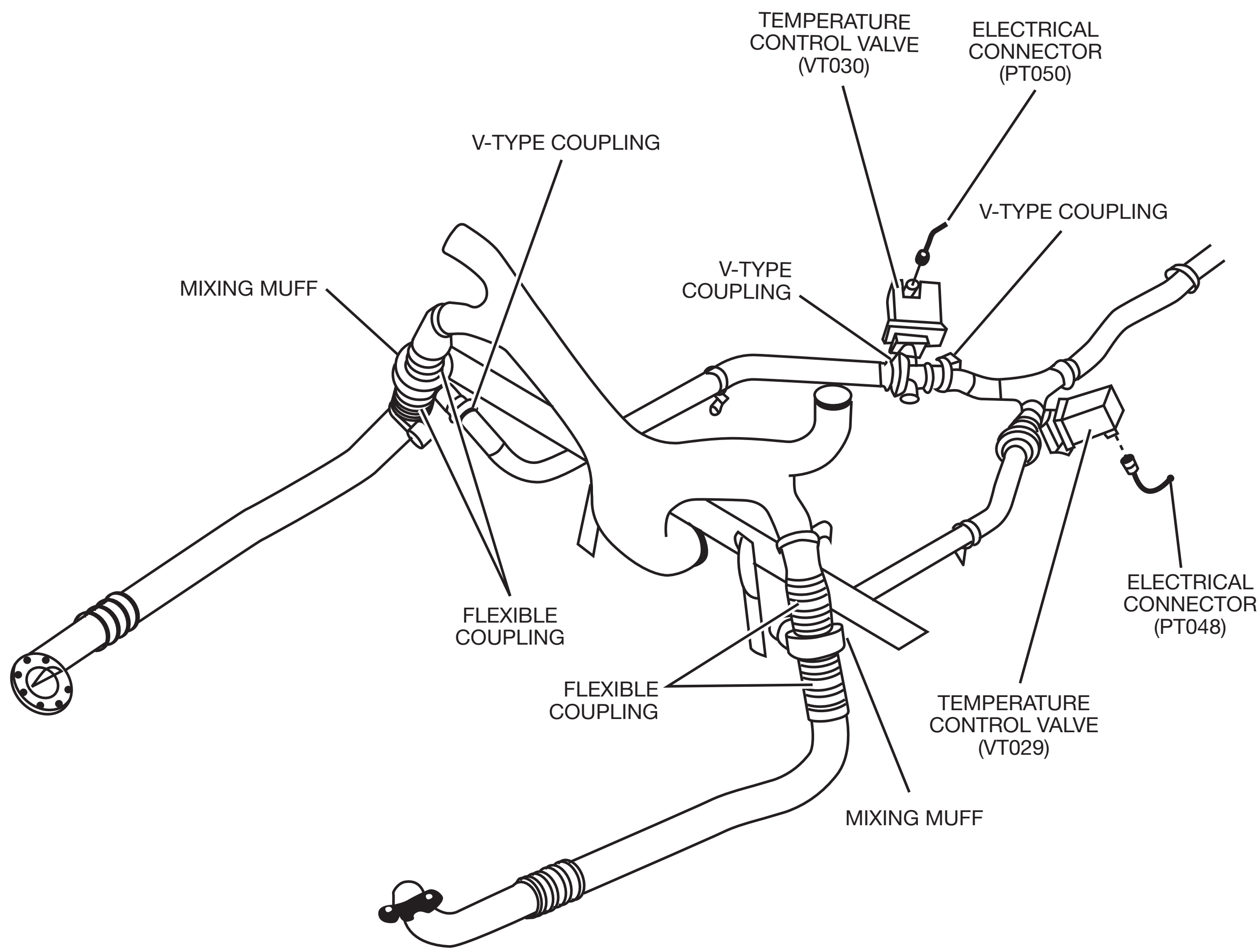


Figure 21-23. Temperature Control Valves

TEMPERATURE CONTROL SYSTEM

OPERATION

The steady state temperature control system (Figure 21-23) automatically maintains average cabin and cockpit temperatures independently to within ±1°F (±0.6°C) of the temperature selected. After selecting a new temperature, the temperature control system stabilizes the cabin/cockpit average temperature ±3°F (±1.7°C) within six minutes. The zone sensors and supply sensors input signals that are compared to the respective temperatures selected on the selector-indicator. If any corrections are required to maintain the selected temperature, the logic portion of the temperature and selector-indicator sends a DC current signal to the respective cabin/cockpit temperature control valve. The control valve modulates toward the opened or closed position to provide warmer or cooler air as required to bring the respective compartment to the selected temperature. The automatic temperature control of the temperature control valve is independent of the manual (backup mode) control of the valve. The automatic temperature selector allows selection of cockpit and cabin temperatures in the range of 65° to 85°F (18.3° to 29.4°C). Selecting manual operation of the cabin/cockpit temperature control system directly controls modulation of its respective temperature control valve. The valve opens or closes in relation to the position of the rotary selector.

NOTES

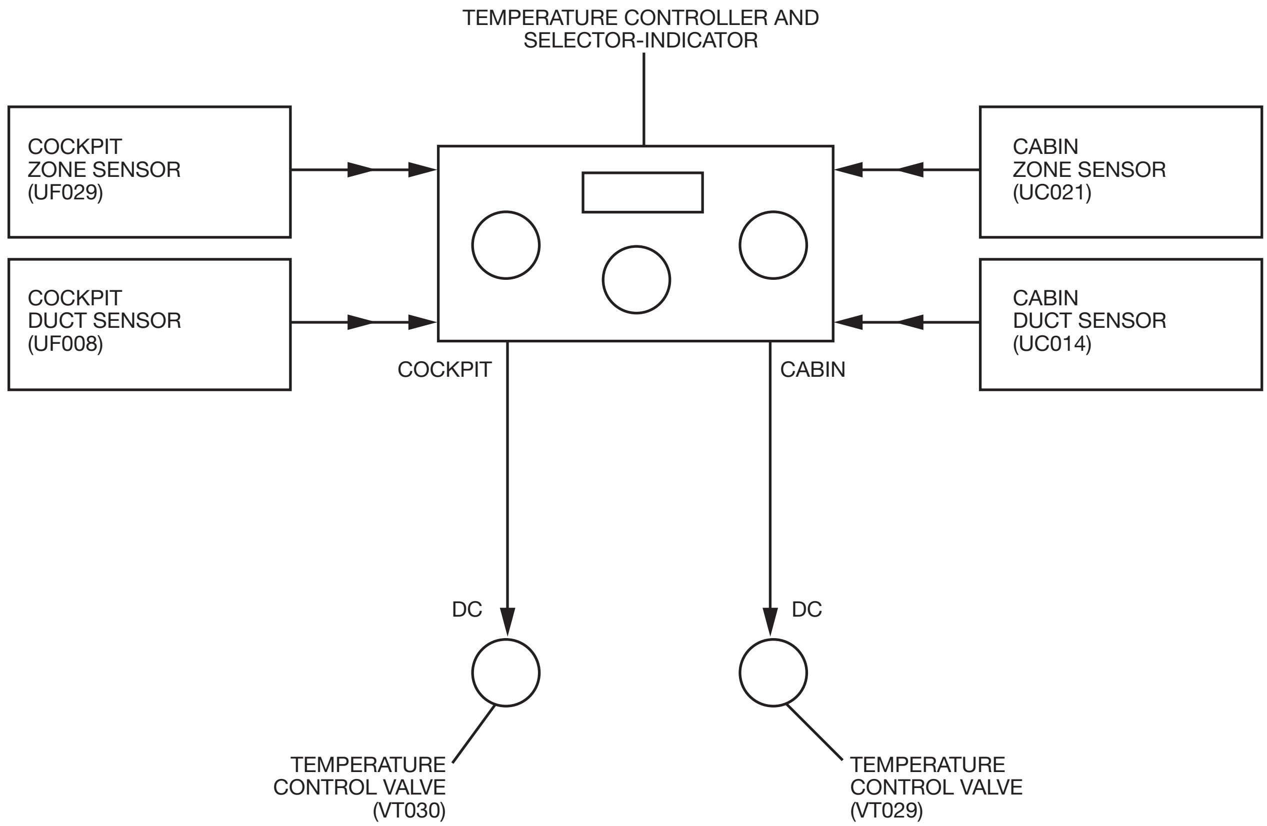


Figure 21-24. Temperature Control System

Temperature displays on the digital display unit of the environmental control panel (Figure 21-24). Place the display selector knob to the desired area temperature to be monitored (CKPT or SUPPLY, CAB or SUPPLY). The temperature for the selected compartment/supply is indicated on the digital display. The cockpit area temperature is taken from the cockpit zone sensor and the cockpit supply air temperature is taken from the cockpit air supply duct sensor. The cabin area temperature is taken from the cabin zone sensor. The cabin supply air temperature is taken from the cabin air supply duct sensor. When the selector is positioned to cockpit SEL/cabin SEL, the display indicates the temperature selected or being selected. The display updates the temperature sensed two times a second and only the digit being updated changes.

If cabin/cockpit duct temperature exceed 300°F (126.7°C) in the cabin/cockpit temperature sensor ducts, the duct overheat switch closes and the AIR DUCT O'HEAT CKPT–CAB annunciator illuminates (XL/XLS) or CABIN AIR DUCT OVERTEMP CAS message (XLS+). If a duct overheat occurs, select a cooler temperature on the environmental control panel and allow the system to cool. When the duct overheat switch cools to below 260°F (126.7°C), the annunciator extinguishes. Positioning the temperature control selector-indicator on the environmental control panel to SUPPLY, monitors the duct temperature.

If the cabin/cockpit duct temperature continues to increase and exceeds 410°F (210°C), a ground is provided from the logic module, enabling the cabin or cockpit temperature control relay to close. Close the temperature control relay disconnect the duct sensor from the cabin or cockpit temperature controller. When the temperature falls below 410°F (210°C) the logic module ground (for the temperature control relay) is lost, allowing the duct sensors to reconnect and become operational.

NOTES

DIAGNOSTICS

Cabin/Cockpit Temperature Control Adjustment/Test

This section explains the built-in tests (BITs) and error codes that appears at the selector indicator when a problem exists.

The BITs and error codes are presented in this section to aid in troubleshooting and testing the system.

Controller Built-In Test

The temperature controller is equipped with BIT capabilities. This test initiates each time power is applied to the temperature controller.

To perform Built-in Test

1. Make sure the electrical power is off and engage circuit breakers AUTO TEMP and MANUAL TEMP.
2. On the temperature controller, set the temperature control selector-indicator to CKPT and the CKPT TEMP SEL switch to AUTO.
3. While observing the digital temperature indicator, apply electrical power to the aircraft. Place the DC POWER BATT switch ON.
4. Verify that the digital temperature indicator indicates the letters EO. This indicates the controller is performing a self test.

   The letters EO display for a short time then change to display the selected temperature.

   At this time any detected errors are indicated on the digital temperature indicator.
5. Place the DC POWER BATT switch to OFF and remove electrical power from the aircraft.
6. On the temperature controller, set the temperature control selector-indicator to CAB and the CABIN TEMP SEL switch to AUTO.
7. While observing the digital temperature indicator, apply electrical power to the aircraft and place DC POWER BATT switch to ON.The letters EO displays for a short time then change to display the selected temperature.

If any errors are detected, the code alternately displays selected temperature and the code.

NOTE

A false E5 and E6 code is reported if either the CKPT TEMP SEL or the CABIN TEMP SEL switch is in MANUAL position during the respective test.

NOTES

Error Codes

The following list explains each test code and the meaning of each:

888 Test of display elements is being conducted.

E0 BIT is enabled and a test of the associated system is being conducted.

El Zone temperature sensor resistance is too low.

E2 Zone temperature sensor resistance is too high.

E3 Duct temperature sensor resistance is too low.

E4 Duct temperature sensor resistance is too high.

E5 Temperature control valve closing current is too low.

E6 Temperature control valve opening current is too low.

E7 Temperature control valve closing current is too high.

E8 Temperature control valve opening current is too high.

E9 Idle current is too high.

An unusually hot or cold sensor may cause El - E4 to display. The error codes clear themselves when the sensors enter the normal operating range. Error codes E5 and E6 is displayed if the indicator selector is in manual operation mode, when a BIT is initiated. Normal operation of the BIT occurs with the CKPT TEMP SEL or CABIN TEMP SEL switch set to AUTO. Error codes E5 through E9 are latched, and continue to be displayed while the BIT is enabled, until a new BIT sequence is initiated. A BIT sequence is initiated by disabling, then enabling the built in test by turning the system power off and back on.

NOTE

A display check is conducted on system power-up, regardless of the status of the BIT. A display check and a test of the associated system is conducted immediately when the BIT is enabled, and on each subsequent temperature controller power-up. If a system fault is discovered, one or more of the error codes (at left) is displayed. Error codes are displayed for the specific compartment selected.

NOTES

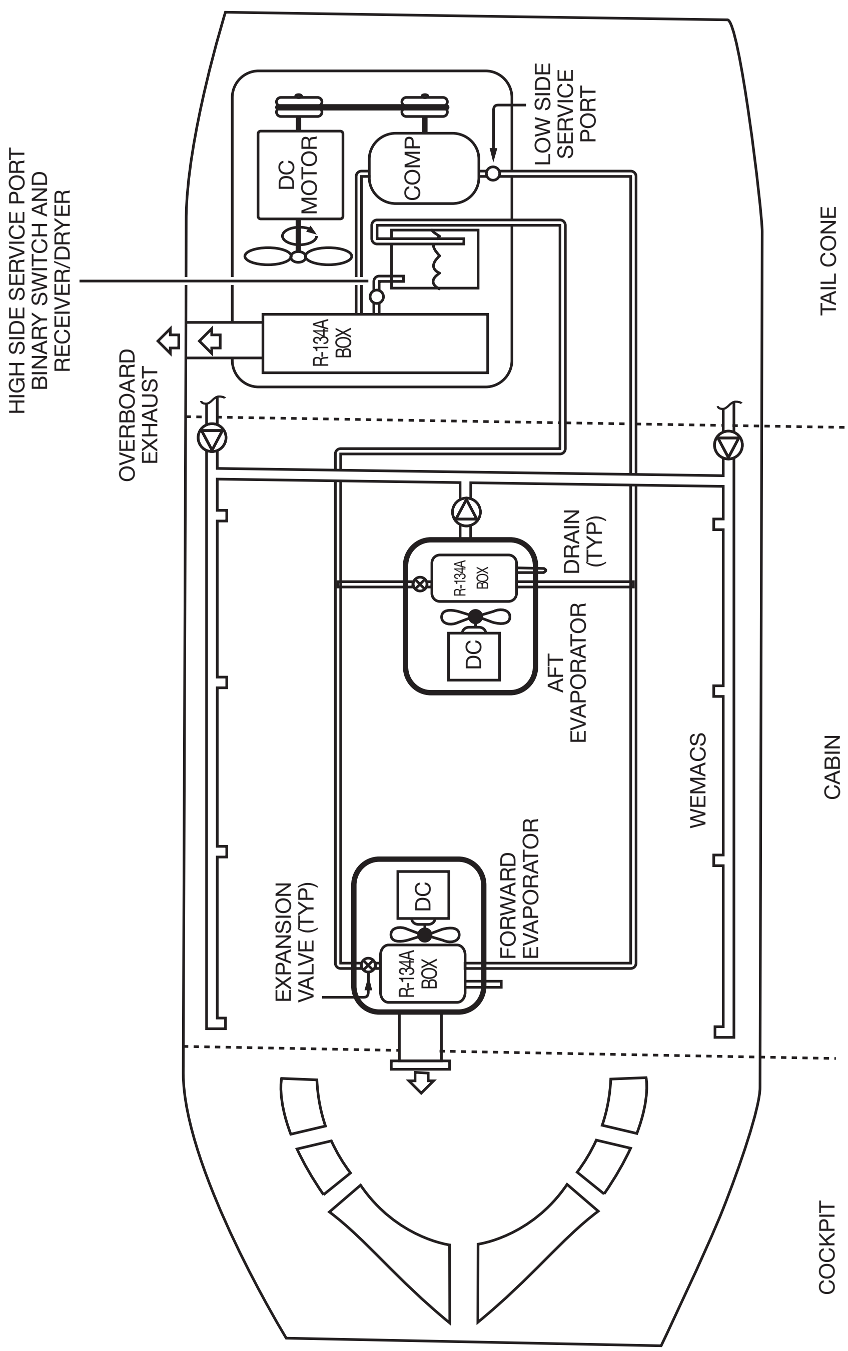


Figure 21-25. Vapor Cycle System Schematic

VAPOR CYCLE COOLING SYSTEM OPTIONAL (XL ONLY)

DESCRIPTION

The vapor cycle cooling system provides cooling and air circulation during ground operations and in flight at low altitude (Figure 21-25). It operates independently or in conjunction with the ECU.

The system includes:

- Two cabin evaporators (one forward and one aft)
- Tail cone condenser
- Compressor and motor
- Associated controls, wiring and plumbing

COMPONENTS

Compressor

The compressor is a rotary piston-type unit on a pallet with the electrical compressor drive motor and condenser. The pallet is on the upper right side of the tail cone.

Condenser

The condenser is on the pallet, just aft of the compressor and drive motor. The condenser inlet and outlet duct through the tail cone right sidewall skin, with a fan driving air through the condenser and out the exhaust duct.

Receiver/Dryer

The receiver/dryer on the pallet, is a nonserviceable part that filters and removes moisture from the refrigerant. The receiver/dryer also functions as a reservoir to separate the liquid from the gaseous refrigerant, allowing only the liquid refrigerant to continue the cycle.

Pressure Switch (Binary)

The system uses a binary pressure switch on the receiver/dryer, to perform two functions for the system. The switch acts as a low-pressure safety switch to prevent system operation in the event of low refrigerant pressure or low ambient temperatures, and it acts as a high-pressure safety switch to prevent damage to the system from excessively high pressure.

NOTES

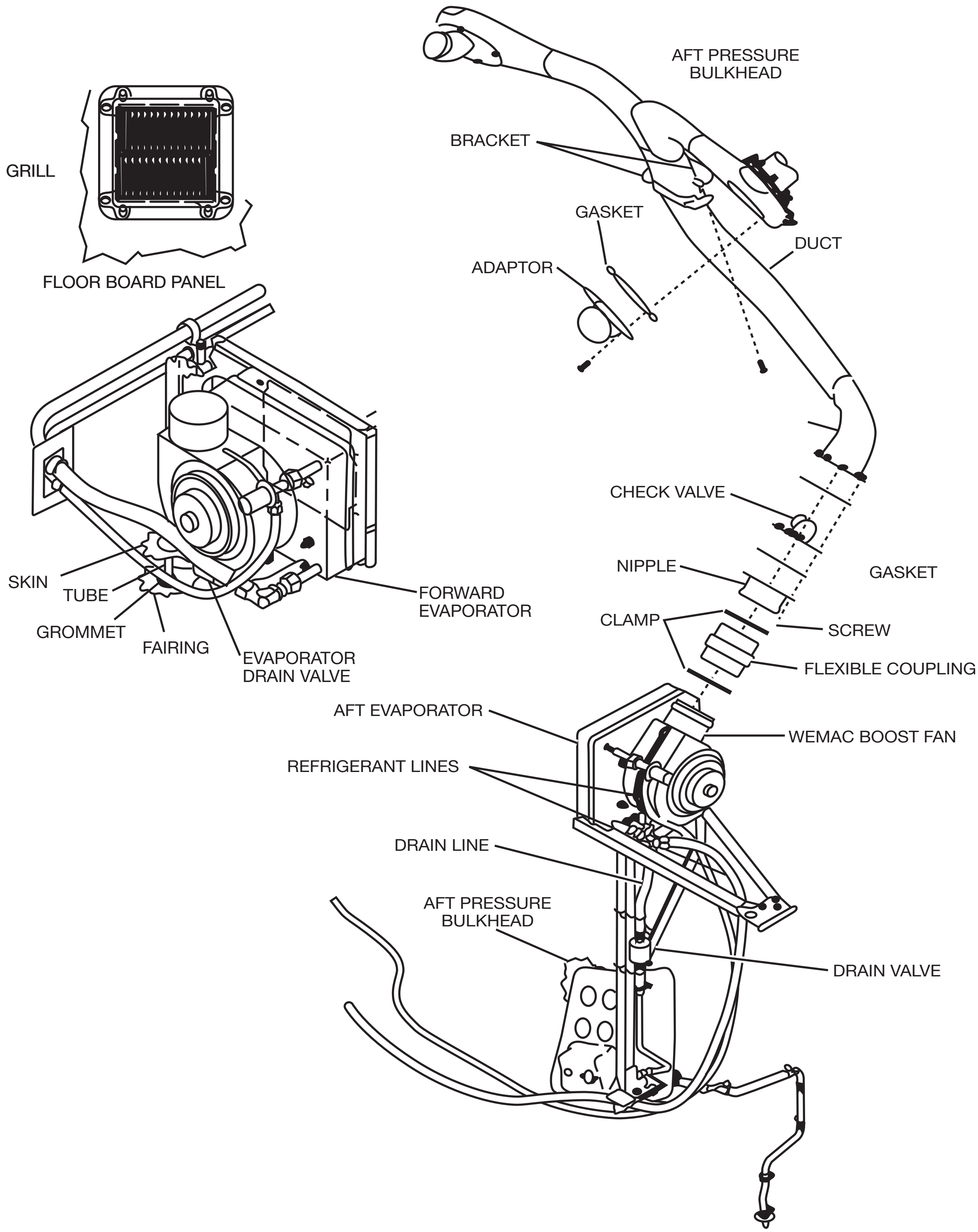


Figure 21-26. Evaporators/Wemac Boost

Evaporators

There are two evaporators in the system (Figure 21-26). The forward evaporator is in the forward end of the dropped aisle. The aft evaporator is in the left vanity area, just forward of the aft pressure bulkhead. The aft evaporator is connected to the overhead distribution system. Its air is distributed through the wemac outlets. The front evaporator discharges air upward aft with a fixed grille that biases a percentage of the airflow either forward or aft. The air is driven across the evaporator coils with electrically powered centrifugal blowers. Electrical power comes from a circuit breaker in the main J-box in the tail cone.

Wemac Boost (XL Only)

Additional airflow through the Wemac outlets is available via the aft evaporator fan, on the forward side of the aft pressure bulkhead (in the aft vanity section). This fan is activated by rotating the A/C–FANS switch on the panel to either the WEMAC BOOST HIGH or LO position. The fan draws cabin air through the non-operating aft evaporator and forces it into both cool air distribution lines, creating a greater flow through all Wemac outlets. An inline flapper-type check valve is in the Wemac boost system to prevent reverse flow of cool air when the blower motor is not operating.

NOTES

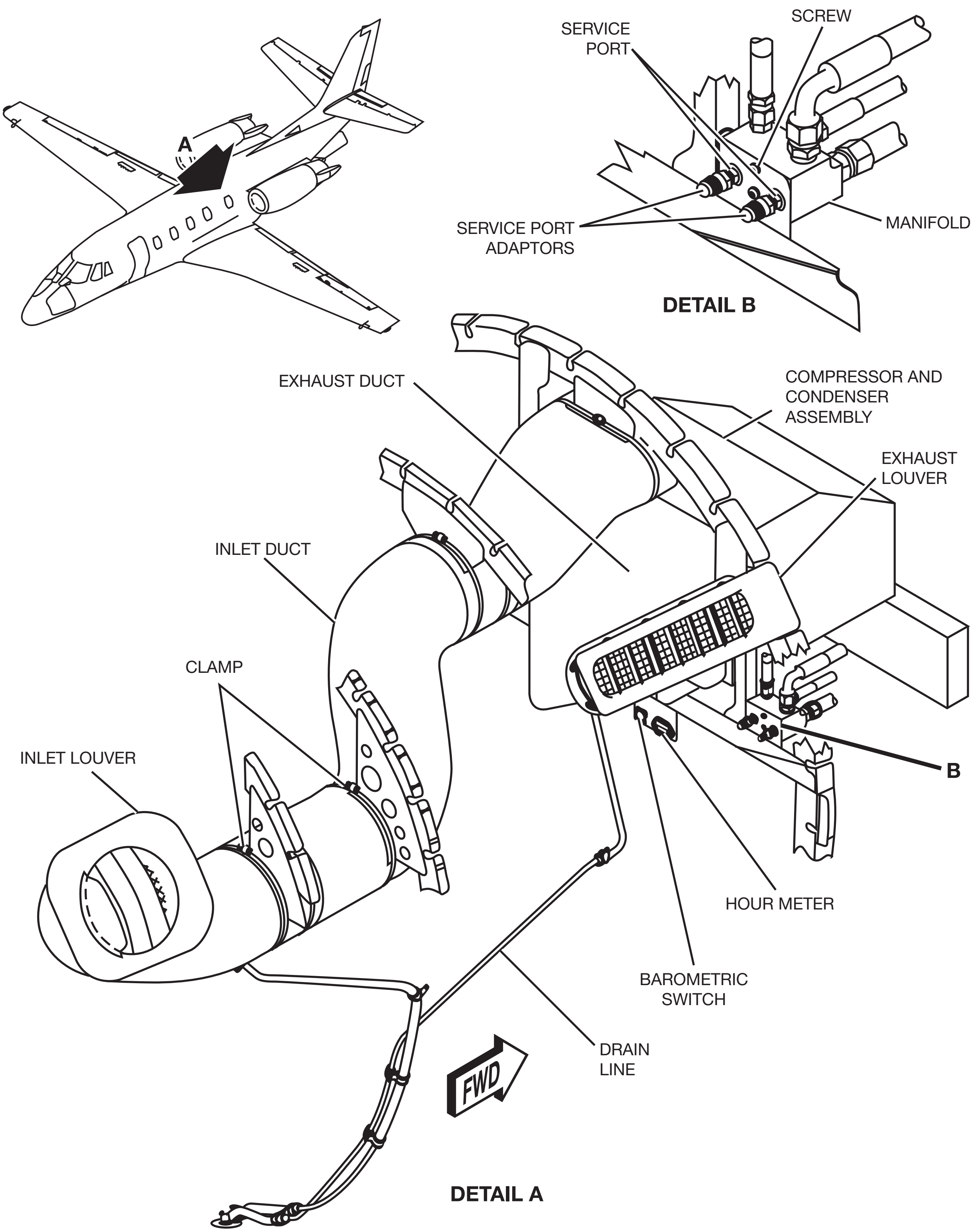


Figure 21-27. Compressor Module Assembly

Sight Glass and Service Ports

There is a sight glass is in the receiver/dryer for a visual check of the charge in the system. The service ports are on the right of the condenser (Figure 21-27). The high pressure port has a larger adapter than the low pressure port. The servicing equipment prevents inadvertent reverse connection.

OPERATION

An electric motor drives the vapor cycle cooling system compressor which pumps refrigerant through the system. The hot gaseous refrigerant from the compressor is condensed into a liquid, by airflow through the condenser. The cooled liquid refrigerant is expanded to a low temperature gas via expansion valves at each evaporator. Cold gas in the evaporators removes heat from the cabin air while it circulates through the evaporators via the evaporator fans.

On the model 560 XL, controls for the air conditioning system are below the copilot primary flight display. The A/C–FANS single rotary switch controls the air conditioning fans. The left side of the switch is labeled A/C with two positions labeled LO and HIGH. With the switch positioned to the LO position, both evaporator fans run at low speed. With the switch in the HIGH position, both evaporator fans run at high speed. There is an indicator light above the rotary switch to provide an indication of compressor operation. The right side of the rotary switch is labeled WEMAC BOOST. The HIGH and LO positions on the right side control only the aft evaporator fan to the appropriate speed. The compressor and forward evaporator fan do not operate with the switch positioned to the right side. Additionally, a barometric switch shuts down the system above 18,000 ft. Also, the aircraft is equipped with automatic load shedding. In flight, both generators must operate in order for the compressor drive motor to operate. In the event that a generator fails, the compressor automatically disconnects from the power source. On the ground, the system is powered either by an auxiliary ground power cart, or by operating either engine. The circuit breaker for the compressor controller is on the left circuit breaker panel and is labeled A/C.

NOTES

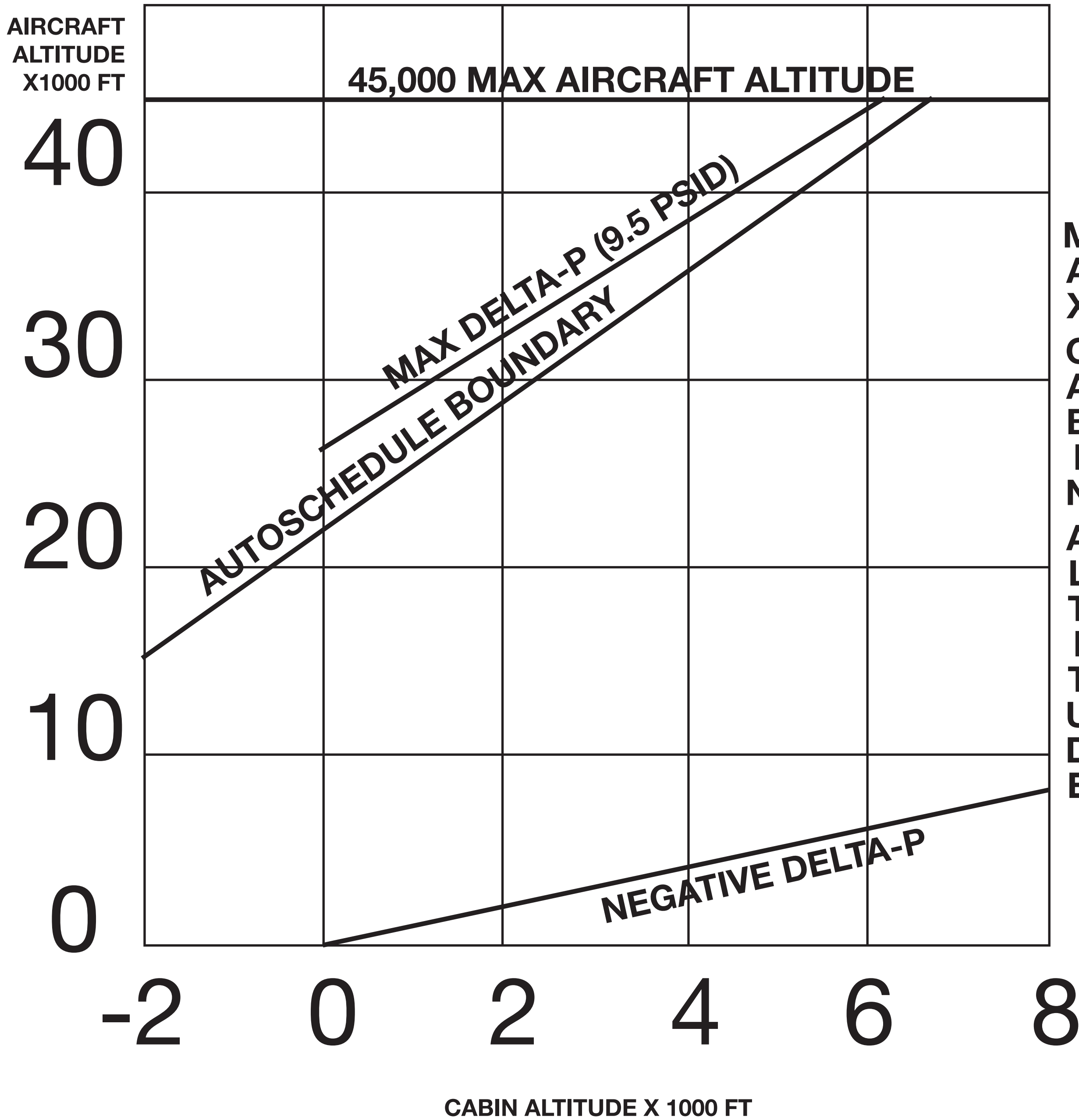


Figure 21-28. Kollsman Auto Pressurization Schedule

PRESSURIZATION CONTROL

DESCRIPTION

This section provides maintenance information for of the environmental system used to control the pressure in the crew and passenger compartments (Figure 21-28).

The pressurization control components responsible for the control of the pressurized area are:

- Cabin altitude controller
- Manual valve
- Outflow valves

The Kollsman pressurization system includes:

- Digital autoschedule controller
- One primary and one secondary outflow valve
- Manual toggle valve

Each outflow valve features an independent maximum differential pressure safety relief and a maximum altitude safety valve. Solenoid on the primary valve enable the cabin altitude controller to smoothly change the operating point of both valves. A common pneumatic connection between valves balances the outflow between them.

The controller is a 100% solid state design. Separate displays provide landing pressure altitude and cabin rate information. System maintenance and testing are facilitated with an aircraft diagnostic capability, that isolates discrepancies on the line-replaceable unit level. Integral landing altitude and cabin rate displays are automatically dimmed in accordance with the lighting voltage.

The pneumatic outflow valves use 23 psi of regulated bleed air for control pressures. Connections between the outflow valves and panel components are limited to three wires and one pneumatic line.

In the event of a power or controller failure, the outflow valves can be controlled manually using the pneumatic toggle valve. The toggle valve is connected to a static line and supplies either static or cabin pressure to the outflow valves for control pressure.

NOTES

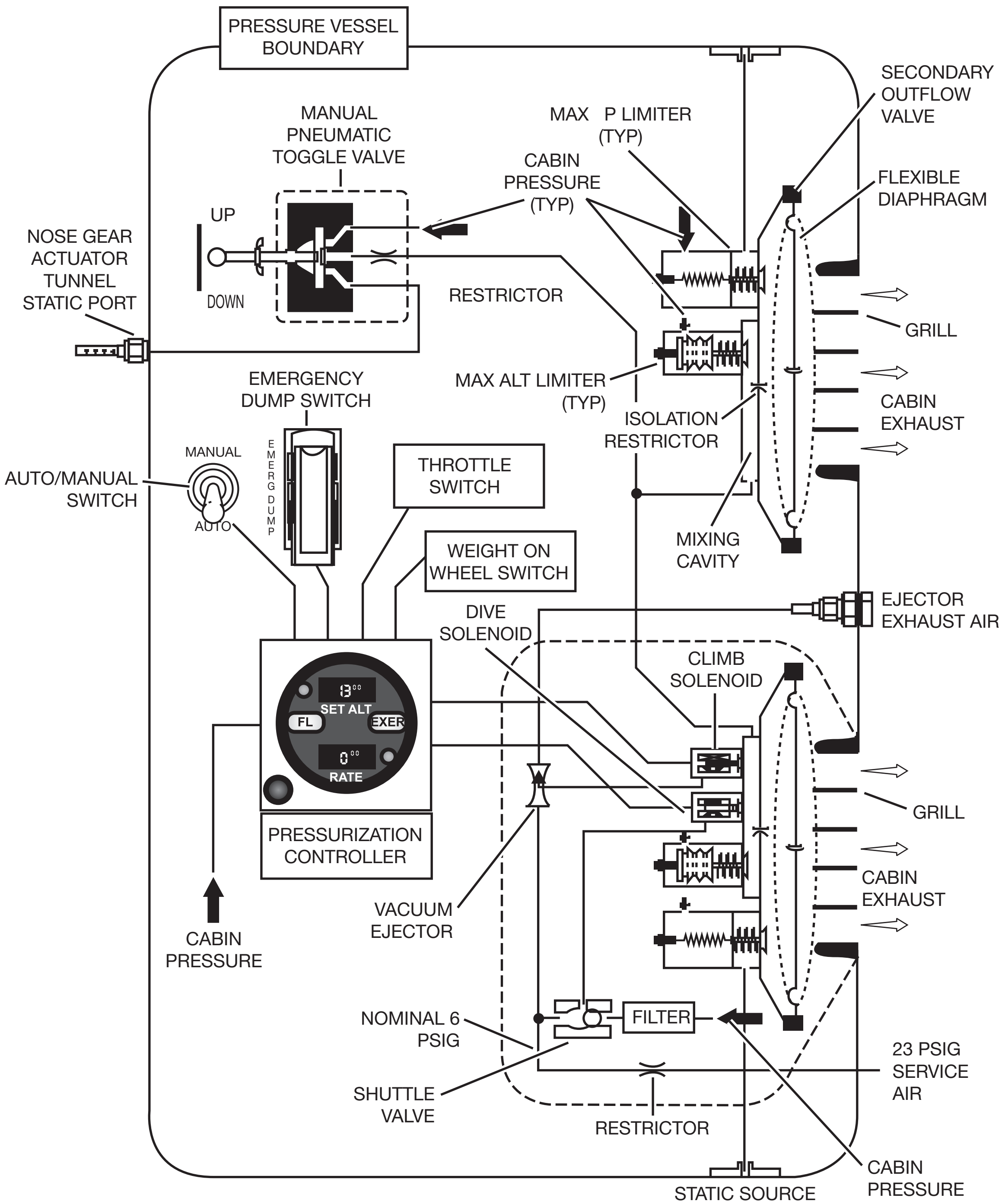


Figure 21-29. System Schematic

COMPONENTS

Outflow Valves

There are two outflow valves at the aft pressure bulkhead on the left side of the aircraft (Figure 21-29): one is the primary outflow valve and the other is the secondary outflow valve. There are two electrical solenoids (climb and dive) attached to the primary outflow valve, along with a vacuum ejector/shuttle valve assembly and filter.

The primary outflow valve regulates the flow of exhaust air using regulated bleed-air pressure and a vacuum. An integral vacuum ejector generates the vacuum, which exhausts to static air. The primary and secondary valves are pneumatically connected together, forcing the secondary valve to duplicate the action of the primary outflow valve.

The outflow valves are not spring-loaded (open or closed). Each outflow valve is constructed with a reinforced flourosilicone diaphragm covering a 4-inch diameter outlet grill. The diaphragm is a sealed reference pressure chamber. Air trapped in this chamber functions as the regulating "spring", which determines the operating point of the valve. Solenoid pilot valves (NC), on the primary outflow valve, are modulated by the controller to change the reference chamber pressure; thereby changing cabin altitude. A common pneumatic connection between the primary and secondary valve reference chambers ensures balanced outflow between outflow valves.

Each outflow valve features an independent maximum differential pressure safety relief connected to static pressure, along with a maximum altitude safety relief valve.

Maximum Differential Pressure Relief Valve

Each outflow valve has an independent cabin pressure relief function which compares cabin pressure to outside static air pressure. The automatic mechanical feature releases outflow valve control chamber air as the cabin-to-outside air pressure differential reaches 9.5 ± 0.1 psid.

The maximum differential pressure safety relief valve consists of two compartments separated by a moveable diaphragm. One compartment is vented to cabin pressure, the other to static pressure. A calibrated spring regulates the movement of the diaphragm with increasing differential pressure. As differential pressure approaches the maximum value, the diaphragm opens a Schrader valve into the outflow valve reference chamber. The outflow valves then open as required to prevent excessive cabin to outside differential pressure.

Maximum Altitude Safety Relief Valve

Each outflow valve has an independent altitude-limit function. The automatic mechanical altitude-limit incorporates an evacuated bellows, which expands as cabin pressure decreases. At a preset absolute pressure, the bellows unseats a Schrader valve and allow cabin air pressure into the outflow valve control chamber. This allows the outflow valves to close as required to limit the cabin pressure to a maximum altitude of 14,500 ± 500 feet. Orifice size provides the maximum altitude safety limit valve with authority over the solenoid pilot valves.

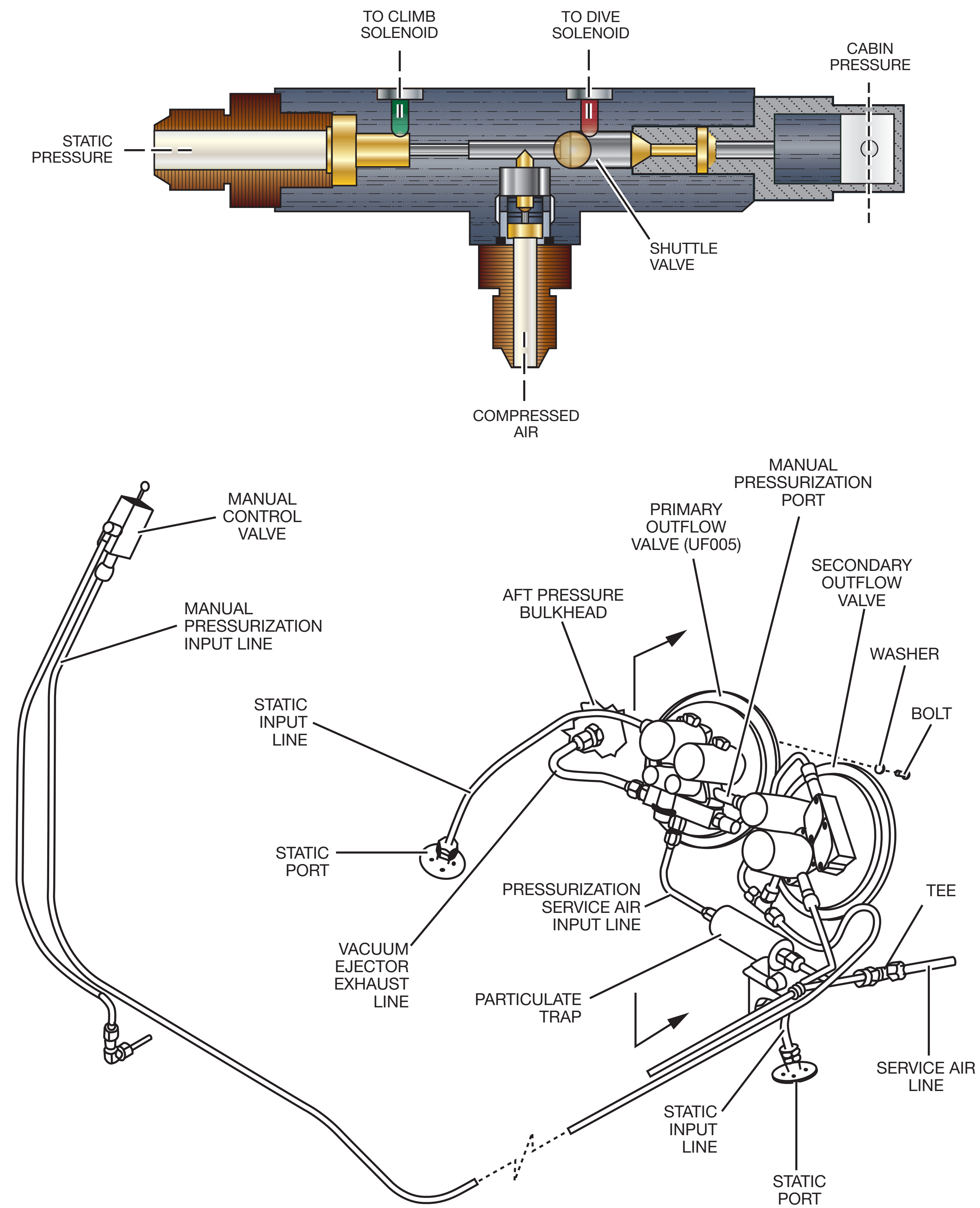


Figure 21-30. Vacuum Ejector/Shuttle Valve

Controller

Cabin Pressure Range .... –2,000 to 14,000 ft

Maximum Cabin Rate (climb) ...... 600 fpm

Maximum Cabin Rate (dive)....... –500 fpm

Electrical Power .... 28VDC, 1.2 A max

Lighting... 5 VDC

Mounting Data.......................... Rear Mount Through Panel

Weight.. 3.3 lbs

Operational Range

Cabin Altitude .... –2,000 to 14,000 ft

Maximum Range, Aircraft Altitude Input.............. –2,000 to 53,000 ft

Cabin Rate Indicator

Operational Range .... –2,000 to 2,000 fpm

Primary Outflow Valve

Maximum Cabin Altitude................................ 14,500 ±500 ft

Maximum Differential Pressure................................... 9.5 ± 0.1 PSI

Secondary Outflow Valve

Maximum Cabin Altitude............................... 14,500 ± 500 ft

Maximum Differential Pressure .... 9.5 ± 0.1 psi

Vacuum Ejector Assembly

The vacuum ejector assembly is a component of the primary outflow valve only (Figure 21-30). It provides a pressure source for operation of the dive solenoid, and generates a vacuum for operation of the climb solenoid.

A check valve in the vacuum ejector assembly determines the pressure source for the dive solenoid. The dive solenoid is operated from the regulated bleed air input (for cabin to static differential pressures below 6 psi) and from filtered cabin air for differential pressures (above 6 psi).

Cabin Dump Switch

An EMER DUMP ON–NORM switch is on the cockpit tilt panel, next to the controller. It can be manually actuated to reduce cabin pressure at anytime. The EMER DUMP ON–NORM switch actuates the primary outflow valve climb solenoid to pull air out of the outflow valve control chambers. The maximum altitude limit valves prevents complete cabin depressurization above 14,500 feet altitude.

The EMER DUMP ON–NORM switch is protected from accidental operation by a lift-lock toggle.

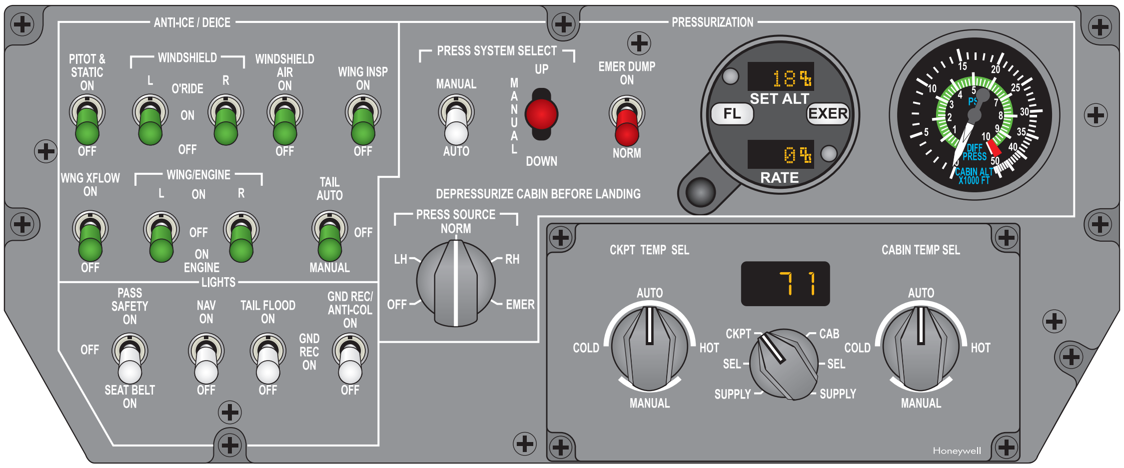
Cabin Altitude Pressure Switch

There is an altitude pressure switch in the pilot side console. It is factory set to actuate at a cabin altitude of 10,000 ft., illuminating the red CABIN ALT. 10,000 ft. annunciator. The switch is set to close at an increasing cabin altitude between 9,650 ft and 10,350 ft. It opens at a decreasing cabin altitude prior to reaching 9,000 ft. Since this is a red annunciator, the Master Warning light also illuminates.

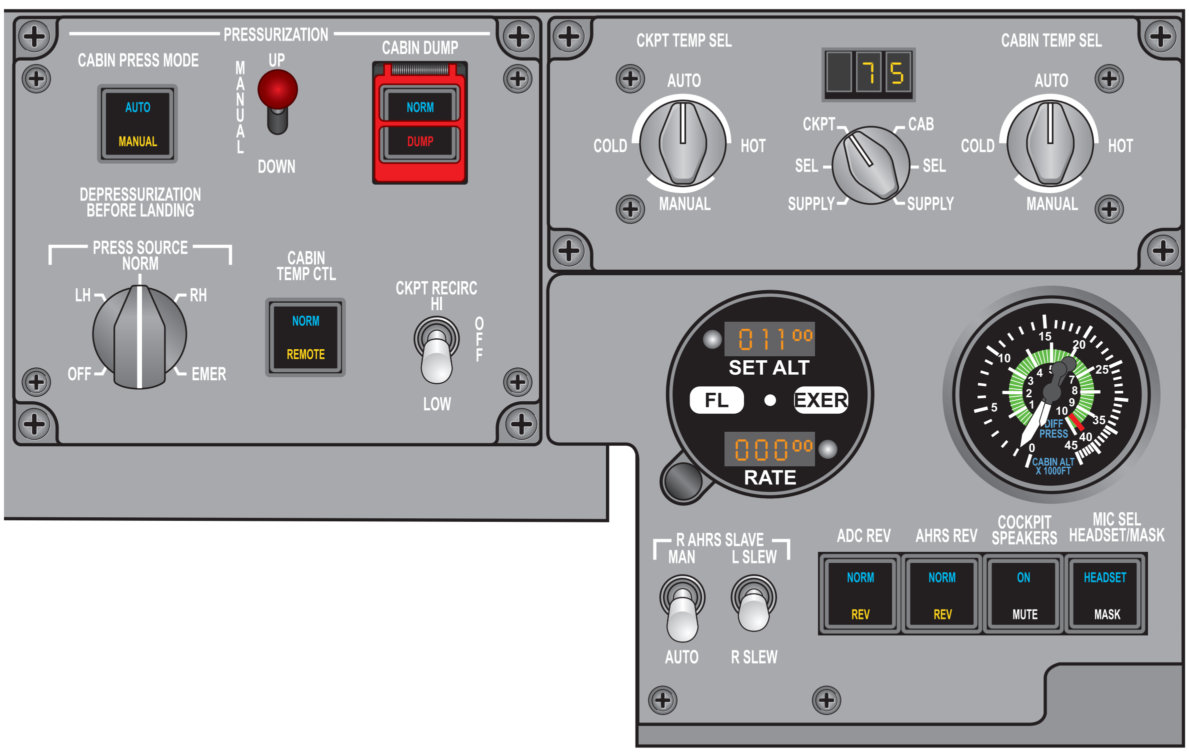
Manual Toggle Valve

A toggle valve provides manual control of cabin pressure in case of an electrical power failure or other emergencies. To establish manual control, place the MANUAL/AUTO switch in the manual position, deactivating the controller solenoid valve outputs. The outflow valves now responds to the manual pneumatic toggle "cherry-picker" valve.

The manual-toggle valve is a three-way/three position valve with a spring that returns to the center (closed) position. The manual toggle valve sup-plies static (climb) or cabin (dive) pressure to the outflow valves.



XL/XLS



XLS+

Figure 21-31. Pressurization Controls

The cabin rate of change, through the manual toggle valve, is controlled only by the amount of control pressure available and length of time operating the toggle valve. The more control pressure available and the longer the valve is operated, the quicker the rate of change.

Controller

The Kollsman controller is a 100% solid state design incorporating:

- Internal cabin pressure transducer
- Microcontroller
- Integral altitude and rate displays
- Maintenance functions
- Selectable configuration databases

The controller regulates the outflow valve setpoint via electrical signals to the climb and dive solenoids on the primary outflow valve (Figure 21-31).

Controller inputs include cabin pressure in pneumatic form, aircraft altitude and barometric corrections via ARINC-429 bus, discrete inputs from aircraft squat, dump, throttle, and auto/manual switches, 28 VDC power, 5 VDC lighting power, and landing pressure altitude set by the pilot using the select knob. Based upon these inputs, the controller produces climb and dive solenoid commands.

The microcontroller implements the autoschedule depending on its inputs. Control signals are transmitted to solenoid drivers, which generate the climb and dive voltages sent to the primary outflow valve. During Maintenance Diagnostic Mode, climb and dive commands are displayed on miniature green and red LEDs on the face of the controller, permitting fault isolation and reducing unnecessary removals.

The controller features two digital displays. Cabin rate in feet per minute (fpm) is always shown on the lower RATE display. The upper SET ALT display shows selected landing pressure altitude during normal operation, and either selected cabin altitude or aircraft flight level data during isobaric operation. The SET ALT display is adjusted by the pilot using the altitude select knob.

The 00 in the right most character of both displays signifies hundreds. Therefore a SET ALT display of "1500" indicates a selected landing pressure altitude of 1,500 ft. Likewise a RATE of "–300" indicates a cabin altitude rate of –300 fpm.

Absence of aircraft altitude information on the controller ARINC-429 input causes the con-troller to switch to isobaric operation. A yellow LED in the upper left corner of the controller face is continuously illuminated whenever the controller is operating in isobaric mode. The left character in the SET ALT display shows either a ^{C}A or ^{F}L icon, signifying selected cabin altitude or flight level mode.

The controller has two push button switches marked “FL” and “EXER”. The FL push button switches between selected cabin altitude and flight level modes, during isobaric operation. The EXER push button can perform an on-ground system test or test the SET ALT and RATE displays.

A third switch, recessed behind the center of the controller face, initiates the controller maintenance mode diagnostics. Miniature red and green LEDs in the lower right and upper left corners of the controller face are activated during the maintenance mode to facilitate on aircraft troubleshooting. A yellow LED in the upper left corner flashes during maintenance mode. Maintenance mode can be accessed when the squat switch indicates that the aircraft is on the ground.

The controller blanks both displays and illuminates a red LED in the upper left corner of the controller face when an internal failure is detected. The red LED distinguishes this “failure detected” mode from the power off condition.

OPERATION

Ground/Flight Modes

Power On, Warm up

Specified accuracy shall be obtained after more than a 5 minute warm up from ambient temperatures of –15°C. During controller warm up, the RATE display shows a false rate indication.

Ground/Taxi Mode

On the ground—with either throttle below approximately 62° TLA—both outflow valves are kept fully open.

Pre-Pressurization Mode

The controller commands the outflow valves to a partially closed position whenever the aircraft squat switch indicates “on ground” when both throttles are greater than approximately 62° TLA. This action initiates pressurization system control of cabin altitude, eliminating pressure bumps at take off.

The controller pressurizes the cabin at –100 fpm toward a cabin altitude of 200 feet below field elevation. Approximately 20 seconds is required for the valves to close sufficiently for full cabin regulation.

The controller exits to auto schedule mode when the squat switch indicates in flight.

Flight Mode

The cabin pressure altitude is maintained by controlling the exhaust airflow rate out of the cabin. The cabin exhaust airflow rate is controlled by the position of the modulating diaphragm in the primary and secondary outflow valves. Varying the pressure in the control chamber behind the diaphragm positions the diaphragm.

The primary and secondary outflow valve control chambers are connected together by a tube and a flow-limiting orifice in each outflow valve.

On Ground Depressurization Mode

The controller operates in the “on ground” depressurization mode when power is applied with the squat switch that indicates “on ground” upon: exiting take-off, pressurization mode, or when the squat switch transitions from in flight to on ground.

The controller provides 30 seconds of controlled depressurization at 1,000 fpm upon initiation of the on ground, depressurization mode. The valves are fully opened after the period of controlled depressurization.

Auto Control

The primary outflow valve has two normally closed solenoids which allow air to enter into and out of the valve control chambers. When the cabin dive solenoid is energized open, cabin air is allowed to pressurize both control chambers and drive both valves toward closed.

When the cabin climb solenoid is energized, air pressure is pulled out from both valve control chambers via the vacuum ejector built into the primary outflow valve—driving both valves open. The solenoid airflow is restricted, so it cannot overpower the maximum altitude limit valve, the maximum DP valve or the pressurization environmental press system select manual toggle valve, which is on the cockpit tilt panel.

The solenoids receive short 28.5 VDC electrical surges from the controller causing the solenoids to momentarily pop open and generate gradual pressure changes in the control chambers. Audible clicks are produced when the solenoids pop open and are heard when the engines are off. The system responds rapidly to minor cabin pressure variations and corrects them before passengers and crew experience discomfort.

NOTES

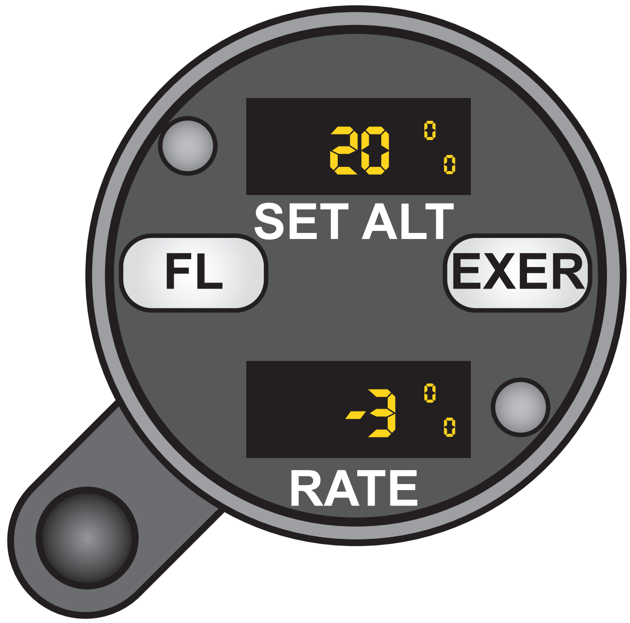


Figure 21-32. Controller: Autoschedule

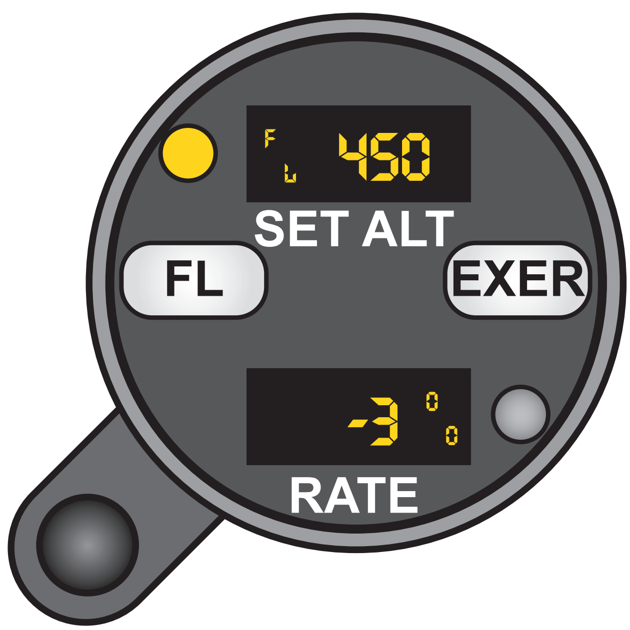


Figure 21-33. Controller: Flight Level Isobaric Mode

When the pressurization select manual/auto switch is set to AUTO, the pilot selects the landing field altitude prior to flight (Figure 21-32). In flight, the controller continually generates an "auto-schedule" based upon:

- Departure field elevation
- Maximum altitude achieved in current flight (per air data sensor) and
- Operator input landing field altitude

The controller defines the pressure rate of change and the cabin altitude based on auto-schedule and air data sensor-indicated altitude.

The controller dispatches 28.5 VDC surges through the primary outflow valve solenoids to obtain a specific cabin pressure response. The auto-schedule completely depressurizes the cabin at the set landing altitude, +1500 feet during landing.

High Control Auto Control

The controller goes into high altitude mode (HAM) when the aircraft is landing on or departing from an airfield between 8,000 feet and 14,000 feet. The primary function of the HAM is to prevent nuisance high cabin altitude annunciation and to minimize the amount of time the cabin altitude spends above 8,000 ft while the aircraft is above FL 250. When the HAM mode activates, the controller outputs a signal that the aircraft systems use to delay the high cabin altitude warning (occurring normally at 10,000 feet) until the cabin altitude reaches 14,500 feet. A new signal occurs simultaneous with deployment of the cabin oxygen drop boxes. To minimize the amount of time the cabin spends above 8,000 ft, the maximum cabin dive and climb rates are increased. The cabin rates are modified as a function of the airfield altitude—proportional, according to the need. At airfields of 8,000 ft and below the normal maximum rates of +600/–500 ft/min apply. When operating out of a 14,000 ft airfield, the maximum rates are increased to +2500/–1500 ft/min. If the ARINC 429 bus signal is lost, the max rates drop back to the default (+600/–500 ft/min.).

In landing at a high altitude airport, cabin pressure altitude does not exceed 8,000 feet, before the aircraft altitude descends below FL 245. Upon descending below FL 245, the cabin altitude climbs at the increased climb rate until the aircraft reaches the selected landing altitude.

Isobaric Control

The controller automatically switches from auto control to isobaric control if the air data sensor information is interrupted (Figure 21-33). A yellow warning indicator on the pressurization controller display face illuminates to advise of this change.

The pilot-selected landing field altitude on the controller display is replaced with a selected flight level that allows the pilot to set the desired aircraft cruising altitude. The controller regulates the cabin pressure rate of change and the cabin pressure altitude, in reference to the selected flight level, to maintain near maximum differential pressure.

21 AIR CONDITIONING

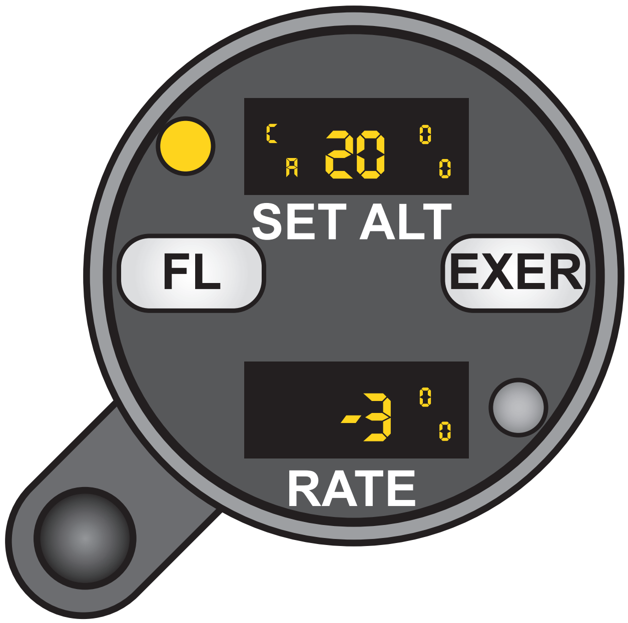


Figure 21-34. Controller: Cabin Altitude Isobaric Mode

The pilot may recall the selected landing field altitude by pressing the flight level (FL) push button on the controller. The selected flight level on the controller display face is replaced with cabin altitude (CA) (Figure 21-34), allowing the pilot to set the desired cabin altitude prior to landing. The controller controls the cabin pressure rate of change to maintain the cabin pressure rate of change to maintain the displayed cabin altitude.

The pilot may "flip-flop" the flight level and cabin altitude displays at anytime by pressing the FL push button on the controller.

Once the air data sensor information resumes, the controller automatically switches back to the auto control flight mode extinguishing the yellow warning indicator.

Manual Control

When the pressurization system select MANUAL/AUTO switch is set to MANUAL, the electric power that opens the climb and dive solenoids is removed. To control the cabin pressure altitude, the pilot must sliding the MANUAL UP/DOWN pressurization control valve up or down.

UP (or cabin climb position) allows the outflow valve control chamber air to vent overboard into the unpressurized nose wheel well— opening the outflow valve—thus causing the cabin altitude to climb.

DOWN (or cabin dive position) allows cabin air pressure into the outflow valve control chamber, closing the outflow valve, causing the cabin altitude to dive.

The cabin pressure rate of change is limited by the orificing in the MANUAL UP/ DOWN pressurization toggle valve and cannot be adjusted by the pilot. It is restricted so that it cannot overpower the maximum DP valve. However, it can override the solenoid valves.

Air from the MANUAL UP/DOWN control valve on the tilt panel is passed through a tube, immediately forward of the aft pressure bulkhead, and teed into the tube between the two outflow valves.

NOTES

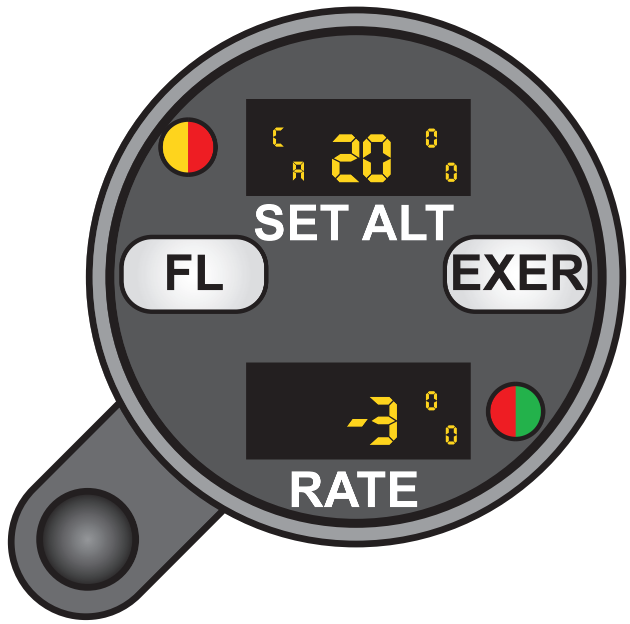


Figure 21-35. Controller: Maintenance Mode

DIAGNOSTICS

CABIN PRESSURIZATION BUILT-IN TEST

Preflight Exer Mode Test

1. Press and hold the EXER system exercise button on display face of controller for two minutes as cabin pressurizes to 200 feet below field elevation (Figure 21-35).
2. Release the button to terminate exercise, display test and gradually depressurize cabin.

NOTES

Table 21-1. BUILT-IN "DIAG" INDICATIONS

ALT SELECT SETTING	"CLIMB" (GREEN)	"DIVE" (RED)	CURRENT (RED)	OUTFLOW VALVE	SOLENOID	DIAGNOSIS
AMBIENT ALT	OFF	OFF	OFF	NO CHANGE	BOTH OFF	NORMAL
AMBIENT ALT	ON STEADY	OFF	ON STEADY		OUT OF CALIBRATION	CONTROLLER
AMBIENT ALT	OFF	ON STEADY	ON STEADY		OUT OF CALIBRATION	CONTROLLER
1,000 FT ABOVE AMBIENT ALT	PULSING OR ON	OFF	PULSING OR ON	OPEN OR MOVING OPEN ON	"CLIMB" SOLENOID PULSING OR	NORMAL
1,000 FT ABOVE AMBIENT ALT	OFF	ON		NO CHANGE	BOTH OFF	FAULTY CONTROLLER
1,000 FT ABOVE AMBIENT ALT	ON	ON				FAULTY CONTROLLER
1,000 FT BELOW AMBIENT AIR	OFF	PULSING OR ON	PULSING OR ON	CLOSED OR MOVING CLOSED ON	"DIVE" SOLENOID PULSING OR	NORMAL
1,000 FT BELOW AMBIENT AIR	OFF	PULSING OR ON	OFF	NO CHANGE	BOTH OFF OR OPEN SOLENOID	OPEN-CKPT TO "DIVE" SOLENOID,
1,000 FT BELOW AMBIENT AIR	OFF	OFF	OFF	NO CHANGE	BOTH OFF	FAULTY CONTROLLER

Ground Maintenance Test

Maintenance personnel are provided with BIT modes, which assist in isolating system faults of the controller or primary valve (Table 21-1). This feature is activated on the ground by depressing a hidden button on the pressurization controller face (on the tilt panel). The button is behind a hole between the FL and EXER buttons. It can be depressed by using a slender non-conductive tool.

Upon entering this maintenance mode, the top display shows “MANT” and the bottom display provides a menu option for different maintenance functions. A Yellow Warning indicator in the upper left corner of the display face continuously flashes when in maintenance mode. Using the altitude select knob on the controller, the user may scroll through the menu of maintenance functions (“DIAG”, “CALB”, and “TIME”). The EXER push button on the controller activates the function that appears on the bottom display. The FL push button deactivates the function and also exits the Maintenance mode.

Diag Function

This function disables the squat switch input inside the controller and allows the controller to operate on the ground in ISOBARIC CONTROL of cabin altitude. This allows maintenance personnel to set a cabin altitude on the controller with or without bleed air while observing the solenoid drive, solenoid current and observing actual outflow valve operation. A Green and Red “Solenoid Indicator” in the lower right corner of the controller face lights up when the respective “Climb” (green) and “Dive” (red) solenoid valve switches in the controller allow current through the solenoids. In addition, a separate red “Current Indicator” in the upper left of the controller face provides an indication whenever either climb or dive solenoid is drawing current. Also listen for solenoid clicking sounds. Allow the controller to warm up 5 minutes or until the rate display shows 0.0, before performing any functional or troubleshooting tests.

Calb Function

The CALB function periodically corrects calibration drift. The controller does not engage this function until the controller has warmed up for at least 15 minutes. With the cabin depressurized, engines OFF, bleed air OFF, and cabin door open, set the ambient pressure altitude in the top of the controller display using the altitude selector knob. After verifying that the setting is correct, press the push-button behind the hole between the FL and EXER buttons on the controller face, to start the recalibration cycle. Upon completion, the unit exits the CALB function and return to the maintenance mode menu.

Time Function

The TIME function indicates the total elapsed time that power has been applied to the controller. The upper display indicates total time in x10 Hr units (e.g. displayed 345 = 3,450 hrs). The elapsed time is not resettable.

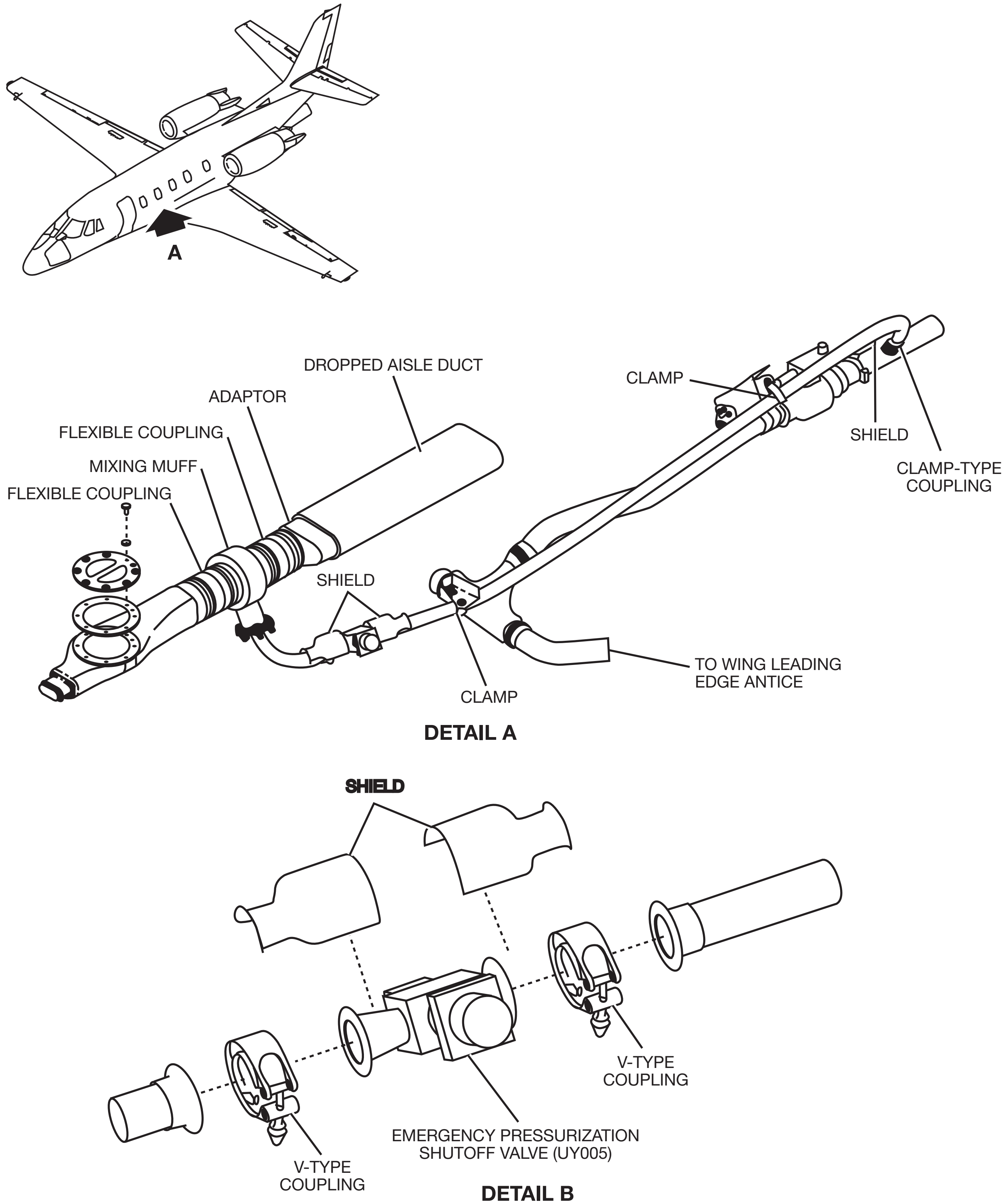


Figure 21-36. Emergency Pressurization System

EMERGENCY PRESSURIZATION

The emergency pressurization bleed air supply is taken from the left wing anti-ice bleed air distribution system (Figure 21-36). The system furnishes the cabin/cockpit with emergency pressurization air when called upon. The emergency pressurization system includes:

- Emergency pressurization shut off valve
- Check valve
- Mixing muff
- Necessary ducting

During an emergency pressurization operation, hot engine bleed air is released into the mixing muff. Bleed air from the engine is too hot to be released to the cabin without some cooling, so the high velocity hot air pulls open the check valve, allowing cool cabin air into the under-floor ducting to lower the temperature.

NOTE

A wire modification on aircraft that have complied with SB560XL-21-19 and SNs 5603 and subsequent prevent emergency pressurization at cabin altitudes that are less than 14,500 feet.

NOTES

QUESTIONS

1. The bleed air temperature between the ACMs compressor and the turbine reaches 420°F, which of the following lights illuminate?
   A. ACM O'PRESS
   B. EMER PRESS
   C. ACM O'HEAT
   D. ACM O'HEAT and EMER PRESS

2. The bypass valve or ECU low temp valve on the ACM:
   A. Mixes hot conditioned air with cool ram air for temperature control
   B. Is plumbed to the emergency pressurization system to mix conditioned cabin air with hot bleed air whenever the emergency pressurization valve is open
   C. Mixes hot bleed air with conditioned air from the ACM, to prevent freezing of the water separator
   D. Does not operate in the manual temperature position

3. At what temperature does the AIR DUCT O'HEAT CKP or CAB annunciator illuminate?
   A. 300°F
   B. 250°F
   C. 200°F
   D. 270°F

4. Which of the following components prevents the formation of ice in the water separator?
   A. Supply duct temperature sensor and controller
   B. Low limit sensor, ECU low limit control valve, and controller
   C. Bleed air ejector nozzle
   D. Primary and secondary heat exchangers

5. During engine run-up, the maintenance technician placed the pressure source selector in the EMER position. The EMER PRESS annunciator illuminated and the cabin began to pressurize.
   A. This condition is normal and no action required
   B. Replacement of the left squat switch is necessary
   C. There is no electrical power to the emergency pressurization valve and it is failed open
   D. Both B and C

6. With the pressure source selector placed in the NORMAL position and only the right engine running:
   A. Both left and right flow control valves are energized open
   B. Right flow control valve is energized open and the left remains closed
   C. Left flow control valve remains closed because the valve requires 7–10 psi of air to open
   D. Both B and C

7. When checking the cabin temperature controller fault codes, an E5 and E6 code is displayed:
   A. Replace the cockpit temperature control valve
   B. Replace the right temperature control valve, under the aft luggage floor
   C. Ensure the temperature selector is in MANUAL and check again
   D. Ensure the temperature selector is in AUTO and check again

8. Maximum differential pressure is controlled by:

A. Outflow valves when the cabin altitude is greater than atmospheric altitude
B. Cabin altitude limit controller when the cabin altitude reaches 14,500 ± 500 feet
C. Outflow valves at a preset point when cabin pressure is lower than atmospheric pressure
D. Outflow valves at a preset point when cabin pressure is greater than atmospheric pressure

9. If a continuous on ground indication occurs in the Kollsman system while in flight:

A. Aircraft depressurizes at a normal rate
B. Pilot must switch to manual and use the manual toggle valve
C. Pressure in the cabin does not change
D. Reduce power below 62° TLA to maintain normal pressure

10. To prepressurize the aircraft on the ground:

A. Both throttles must be above 62° TLA
B. Both throttles must be below 62° TLA
C. Only the right throttle must be 62° TLA
D. Only the left throttle must be above 62° TLA

11. Which statement is true concerning the Kollsman pressure system?

A. Only the primary outflow valve has climb and dive solenoids
B. Maintenance diagnostic mode can only be used in flight
C. Uses 23 psi of air press to the climb solenoids
D. Does not have emergency dump capabilities

12. At what altitude does the vapor cycle compressor motor, if installed, shutdown?

A. 14,500 feet
B. 18,000 feet
C. Compressor does not operate in flight
D. Compressor only shuts down if one generator is switched off above 18,000 feet

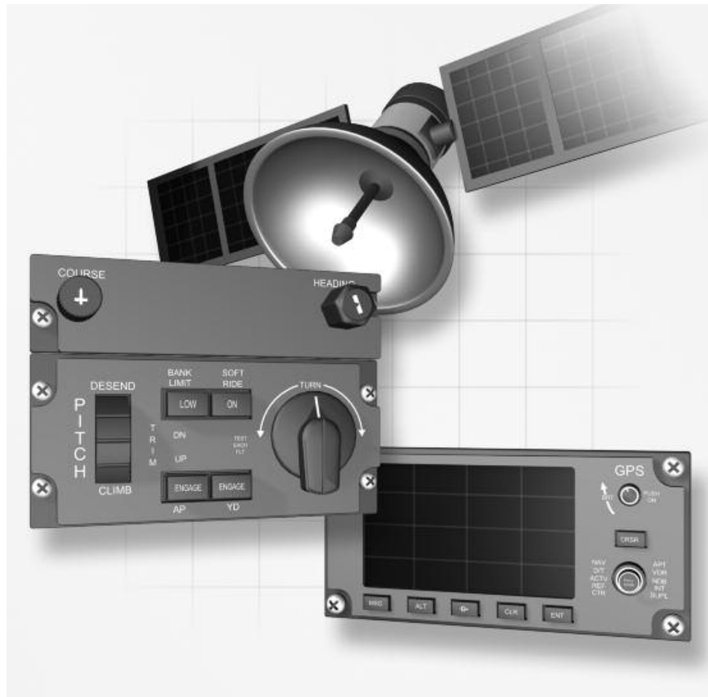
CHAPTER 22
AUTOFLIGHT

CONTENTS

Page

ILLUSTRATIONS

CHAPTER 22
AUTOFLIGHT



INTRODUCTION

The autopilot section describes the portion of the system controlling the flight path of the airplane through adjustment to pitch, roll, or yaw, autopilot servos and associated cables. This section provides maintenance information on the autopilot servo, autopilot controller, servo bracket, cable drum, and servo cables. Individual servos are installed to control aileron, rudder, and elevator surface positions. The autopilot system is integrated with the flight director.

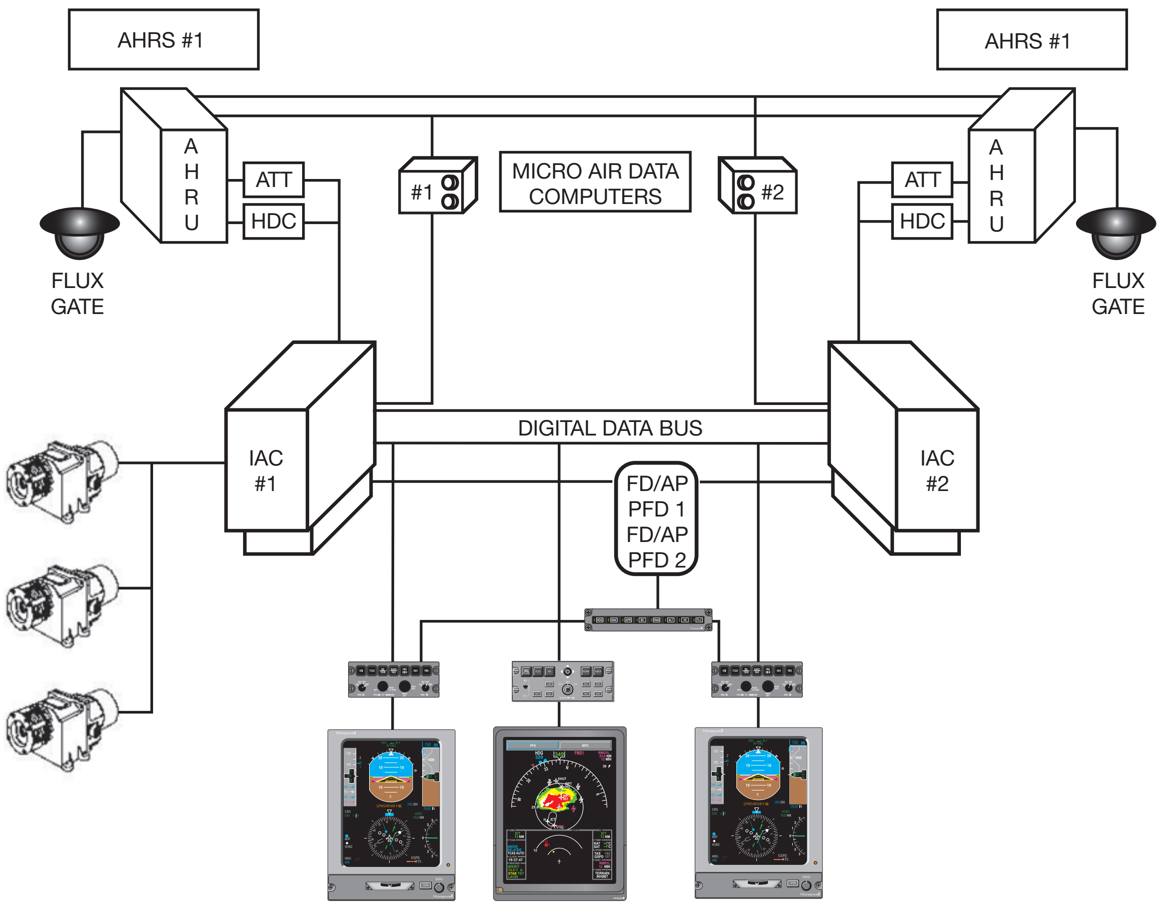


Figure 22-1. Primus 1000 Block Diagram

AUTOMATIC FLIGHT CONTROL SYSTEM—XL/XLS

The automatic flight control system (AFCS) consists of two IC-600 display guidance computers—one MS-560 flight director/autopilot mode selector, one PC-400 autopilot controller and three flight control SM200 servos (pitch, roll and yaw) (Figure 22-1).

Features of the autopilot system include:

- Yaw damping
- Elevator trim
- Heading hold
- Pitch hold
- Bank limit modes
- Touch-control-steering

The coupling of flight director modes with autopilot engagement is also featured.

DESCRIPTION

Three flight maneuvering options are available to the pilot, manual operation, automatic operation or manual control using the autopilot.

Manual

The pilot can hand-fly the airplane with the controls when the autopilot is disengaged. The desired flight mode is selected on the MS-560 mode selector and the necessary flight path command is displayed on the primary flight display (PFD). The pilot then flies the airplane using the commands displayed.

Automatic

When A/P ENGAGE is pressed on the PC-400 autopilot controller, autopilot couples to the mode selected on the MS-560 flight director mode selector. The autopilot then flies the airplane automatically while the pilot monitors its performance on the PFD.

Semi-Automatic

The pilot can fly the autopilot by using either the pitch wheel and turn knob or touch control steering. Touch control steering, when engaged, de-clutches the pitch and roll SM200 servos. The pilot is then in manual control of the airplane, and cannot use the pitch wheel or turn knob with touch control steering engaged. Using the pitch wheel or turn knob cancels the engaged vertical or lateral flight director mode.

The PC-400 autopilot controller (on the pedestal) provides engagement control for the autopilot, yaw damper, and low bank angle, as well as manual control of the airplane through the autopilot.

The controller includes:

- Turn knob
- Pitch wheel
- Push-on/push-off illuminated engage switches for the autopilot
- Yaw damper
- Low bank angle

The touch control steering (TCS) buttons, as well as the autopilot/trim disconnect buttons (AP/TRIM), are situated on the pilots and copilots control wheels (Figure 22-3).

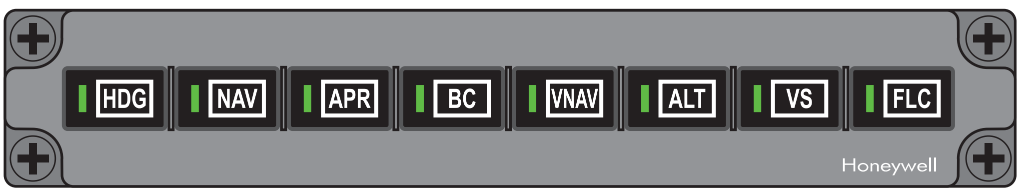


Figure 22-2. Flight Director Mode Selector

OPERATION

Flight Director Modes

The MS-560 flight director/autopilot mode selector provides flight director modes that can be coupled with autopilot (Figure 22-2). Mode annunciation is provided on the mode selector (the selected mode switch illuminates when pressed on). Flight director mode annunciations are also integral to the primary flight displays.

HDG

Selects/deselects the heading select mode, The command bars on the PFD are positioned to track the location of the heading bug. While in a heading mode, a lower bank limit can be selected with the bank limit button on the PC-400 autopilot controller. Low bank is automatically selected above 34,000 feet (10,363 m) mean seal level.

NAV

Arms/deselects navigation mode, the flight director computer can arm, capture, and track the selected navigation signal sources:

- VOR (very high frequency omnidirectional range)
- LOC (instrument landing system localizer)
- FMS (flight management system)

When APR is selected, the NAV select also annunciates.

APR

Arms/deselects approach mode. The appropriate gains are selected to arm and capture the lateral deviation signal for VOR APR, LOC, BC, and both lateral and vertical navigation signals for ILS to meet approach criteria.

VNAV

In the VNAV (vertical navigation) mode the system can arm and capture a VOR/DME (distance measuring equipment) or (FMS) based vertical profile enabling a coupled climb or descent to a waypoint altitude. The pilot enters the vertical profile data, using the multi function display (MFD) VNAV menu.

ALT

This selects the altitude hold mode, and overrides all active pitch flight director modes. When the altitude is captured in the pilot flight display altitude select display window the system maintains that altitude.

VS

This selects the vertical speed hold mode, and the system maintains the current vertical speed. A new vertical speed can be selected and maintained using either the autopilot pitch wheel or the TCS button. The vertical speed target is displayed on the PFD.

FLC

This selects the flight level change mode, and the system maintains the current indicated airspeed or permits a new indicated airspeed to be selected, using the pitch wheel, or the TCS button. The indicated airspeed target displays on the primary flight display. FLC can be used with the FMS vertical navigation to maintain a FMS-supplied speed target.

BC

This selects the back course mode. The flight director computer tracks localizer back course.

Autopilot Operation

Autopilot Modes

When particular mode switch is pressed on the MS-560 flight director mode selector the mode is engaged and coupled. The mode switch illuminates when that mode is engaged and coupled.

Figure 22-3. Outboard Control Horn

Yaw Damper Mode

The yaw damper mode provides yaw stabilization and turn coordination through rudder control. The yaw damper engages by pressing Y/D ENGAGE on the autopilot controller or by engaging the autopilot.

Autopilot Engage Mode

When the autopilot is engaged, the elevator, aileron and rudder servo clutches engage and the autopilot controls the airplane by changing the position of the control surfaces through the servos. The autopilot can be engaged in any reasonable attitude. If autopilot is engaged when the flight director is in standby mode, the autopilot provides three-axis stabilization (with the roll axis in the heading-hold mode, the pitch axis in the pitch-hold mode, and yaw damper mode automatically engaged). The autopilot couples to the flight director mode and maintains the commanded pitch and roll attitude, when engaged.

The autopilot is disengaged by the following methods:

- Actuating of the autopilot disengage switch on the control wheel
- Pressing AP engage switch on the operating electric trim
- Pressing the go-around switch

Heading-Hold and Pitch-Hold Modes

The autopilot is in the heading-hold mode and the airplane heading is maintained when the turn knob is in detent, the roll attitude is less than 6°, and no lateral flight director modes are engaged. The autopilot pitch axis is in the pitch-hold mode when no vertical flight director modes are engaged.

Pitch Wheel Mode

Rotating the pitch wheel results in a change of pitch attitude proportional to the rate of rotation of the wheel. This permits positive control of pitch attitude changes. If the autopilot is coupled to a lateral and vertical flight director mode and the pitch wheel is moved, the engaged vertical flight director mode cancels and the autopilot is in the pitch sync mode.

Turn Knob Mode

Rotation of the turn knob out of detent results in a roll command. The resulting roll attitude is proportional to and in the direction of the rotation of the turn knob. If the autopilot couples to lateral and vertical modes and the turn knob moves out of detent, the engaged lateral mode cancels and the autopilot is in the heading hold mode.

Touch-Control Steering Modes (TCS)

A switch on the control wheel allows the pilot to manually control the airplane attitude through control wheel column movements (Figure 22-3). When the TCS switch is pressed and held, the elevator and aileron servo clutches disengage, and the pilot is free to fly the airplane manually without opposition from the autopilot. When the TCS switch is released without a vertical mode having been selected on the flight director, the existing pitch attitude is held. If the airplane is at a roll attitude above 6° without a lateral mode selected when the switch is released, the roll attitude is maintained. If the roll attitude is less than 6° when the switch is released, the existing airplane heading is held. TCS allows the pilot to modify the commanded flight path from the flight director. For example, when the autopilot is coupled to an AIR DATA hold mode (altitude hold, vertical speed hold or FLC) or pitch-sync mode, TCS can be used to manually change the vertical flight path through pitch attitude or power change. Upon release of the switch, the new reference is held. If the autopilot was coupled to a lateral mode during the use of TCS, the system remains coupled to the lateral mode when the TCS switch is released.

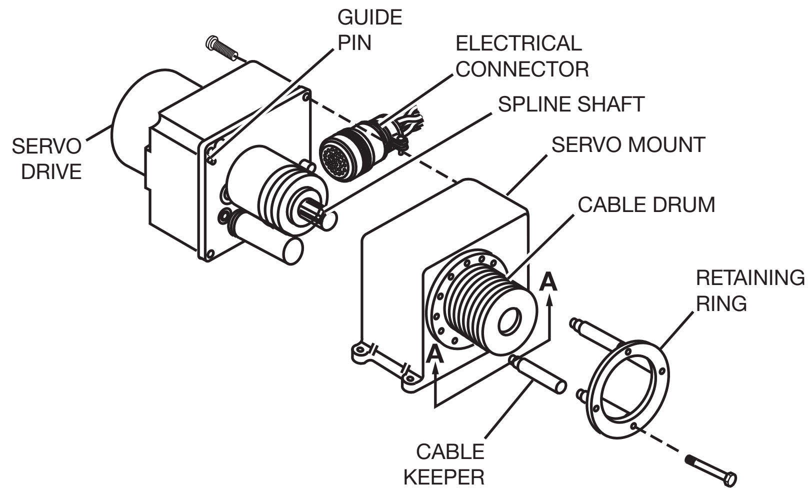


Figure 22-4. Autopilot Servo

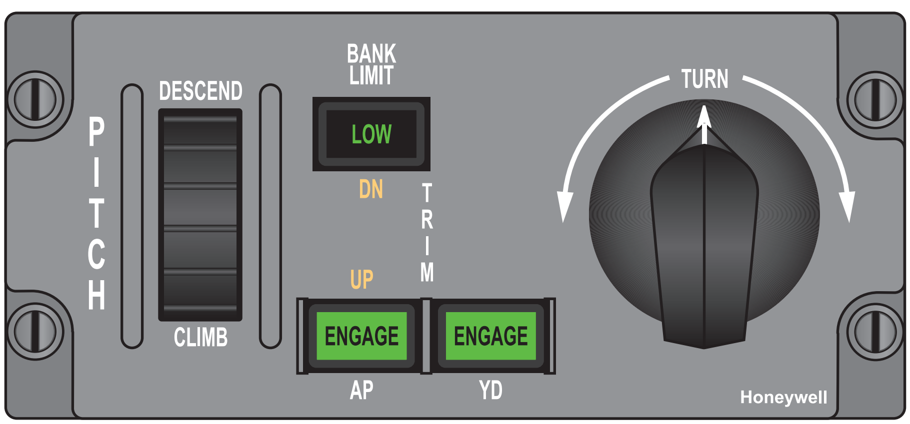


Figure 22-5. Autopilot Controller

Bank Limit Switch

When in the HDG mode, this switch decreases the flight director roll command bank limit to 17°.

DIAGNOSTICS

Autopilot Servo

Description

This section provides maintenance information on the autopilot servo actuators and the servo cables (Figure 22-4). The servo actuator is comprised of a servo mount and a servo drive. An individual servo is installed in each axis and controls aileron, elevator and rudder surface positions.

Servo Mount

The servo mount is a cast aluminum housing with a cable drum assembly. The drum accepts the splined shaft of the clutch assembly. The shape of the servo mount housing provides protection for the clutch assembly and synchro when the servo drive assembly is installed in the servo mount.

Removal of the servo mount requires disconnecting servo cables. Consequently, servo cable rigging is required upon reinstallation of the servo mount. Refer to Chapter 27, Aileron and Trim Tab—Adjustment/Test, Rudder and Tab System—Adjustment/Test or Elevator and Tab System—Adjustment/Test.

Servo Drive

The servo drive includes:

- Motortachometer
- Clutch assembly
- Synchro and power gear train

The servo drive translates electrical inputs into a rotational mechanical output, that drive the servo mount cable drum. Each servo drive contains an integrated DC torque motor tachometer that drives the output engage clutch through a power gear train. The motor position also transmits to a synchro through an instrument gear train. This assembly, with a spline output on the clutch, mates with the drum and bracket. The tachometer rate signal feeds to the autopilot computer servo amplifier.

The pitch, roll and yaw servos are electrically driven and provide surface displacement proportional to input signals. Each servo includes an engage clutch that disengages the servo output shaft, and leaves it free to rotate when the autopilot is turned off. The output shaft connects through the servo drum to the airplane control cables.

Autopilot Controller

Description

This section includes maintenance practices for the PC-400 autopilot controller. Maintenance practices include removal/installation of the autopilot controller and the autopilot disengagement and warning test (Figure 22-5).

The PC-400 autopilot controller provides the pilot with the ability to manually introduce turn and pitch commands to the autopilot computer, and to select operational modes of the automatic flight control system through the autopilot computer.

The autopilot controller is a unit on the pedestal that has:

- PITCH knob and generator assembly
- Detent TURN switch and variable resistor assembly
- Three momentary action, annunciating and push button switches

All items are identified by the nomenclature on an edge lighted panel. The TRIM UP or DN indicator is used only as an annunciator.

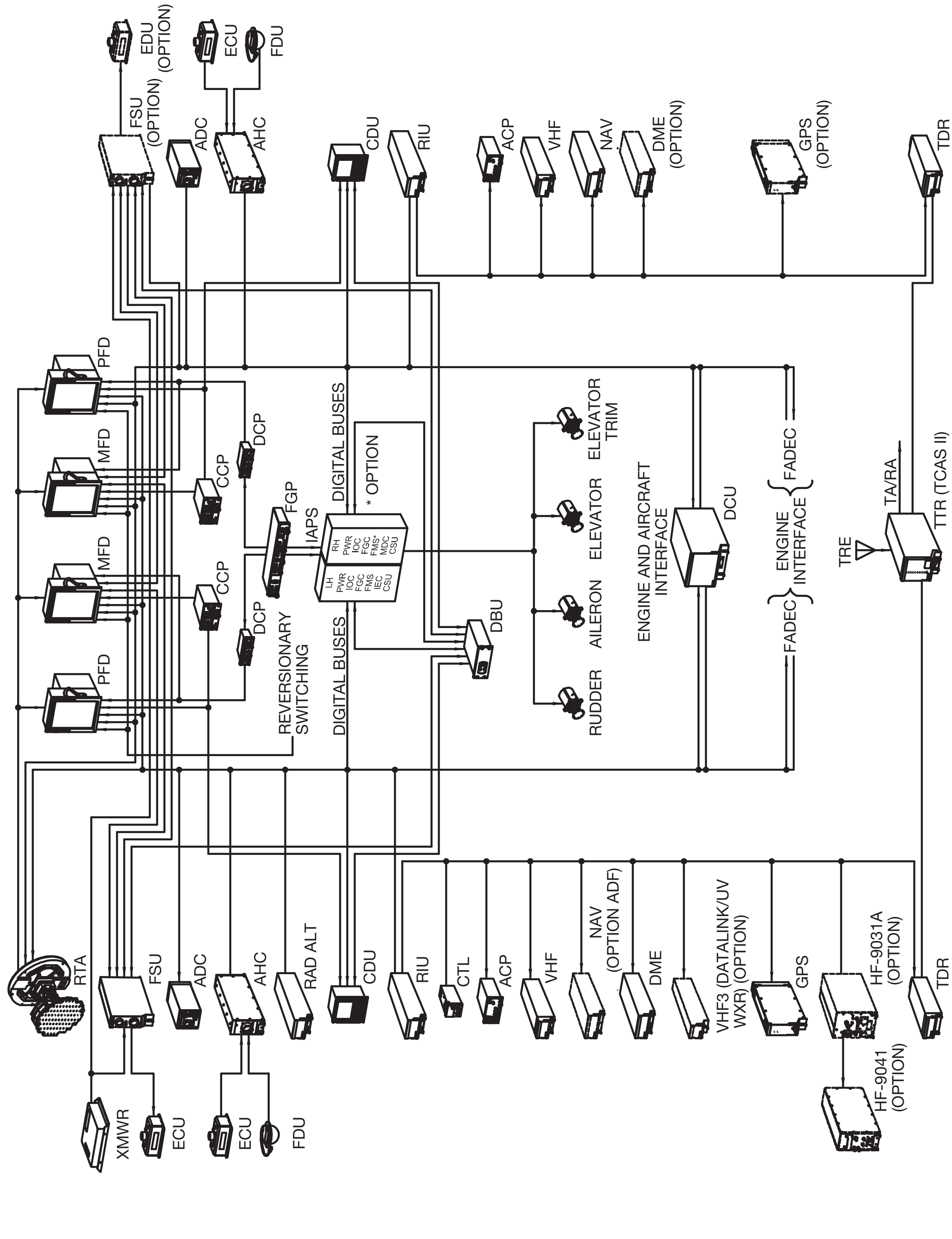


Figure 22-6. Avionics System Block Diagram

AUTOMATIC FLIGHT CONTROL SYSTEM— XLS+

DESCRIPTION

The Collins flight guidance system has a FGP-3000 flight guidance panel (FGP) that is installed in the upper center instrument panel in the fire tray. The FGP has lateral and vertical mode selection switches, heading and course control knobs, a speed knob, an altitude alert/altitude preselect knob, a VS pitch control wheel, and autopilot and yaw damper controls. The autopilot and yaw damper controls include autopilot engage, yaw damper engage, couple switch, and yaw damper/autopilot disconnect. The FGP also has dual flight director controls (Figure 22-6).

The FGC-3000 flight guidance computer modules are a component of the integrated avionics processor system (IAPS) that is installed inside the ICC- 31 11 integrated card cage (ICC) in the right side nose avionics compartment. The FGC modules control the data for the Collins flight guidance system. The FGC modules send the commands to the aileron, elevator, and rudder autopilot servos for the three-axis autopilot control.

NOTES

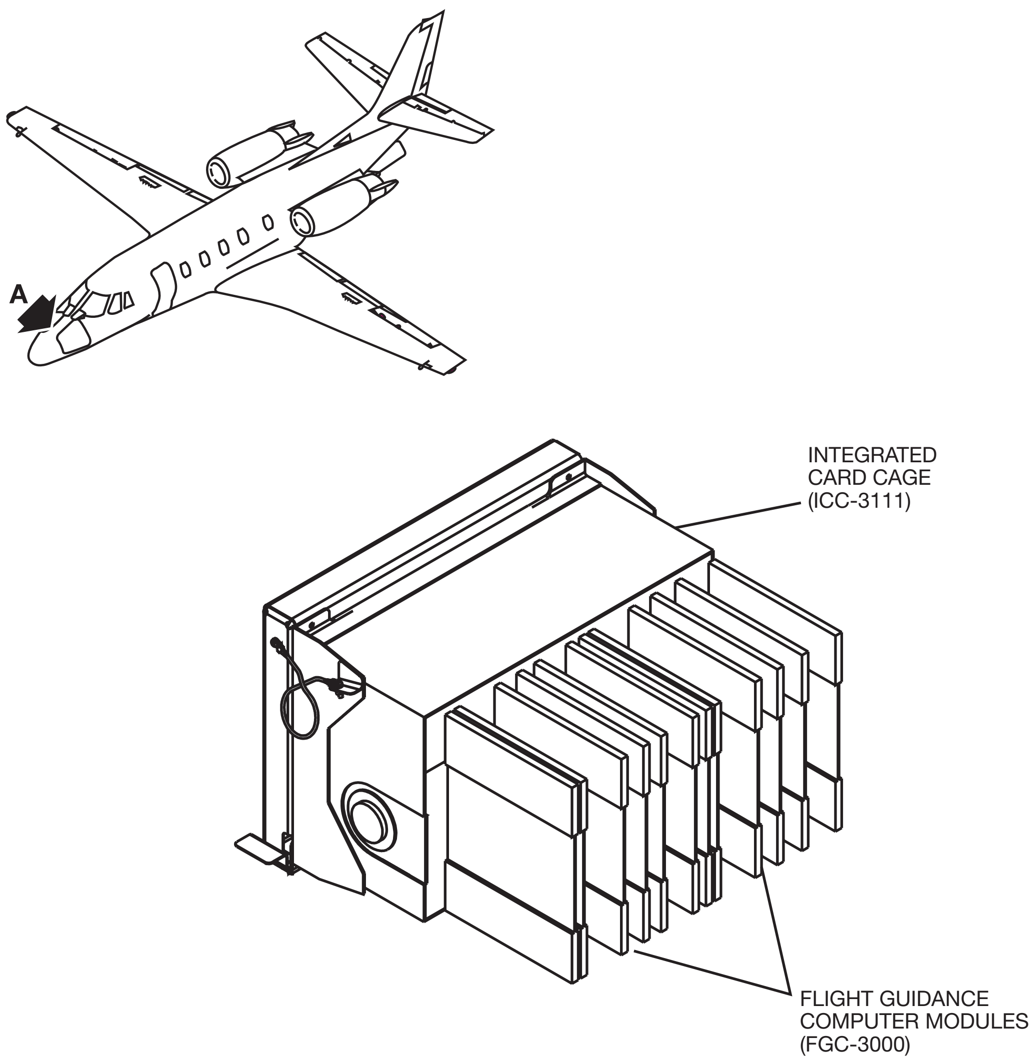


Figure 22-7. Pro Line 21 Flight Guidance Computer Modules (FGC-3000)

COMPONENTS

Flight Guidance Computers

Two FGC-3000 flight guidance computers are located in the IAPS card cage and operate together to provide three-axis autopilot and pitch trim functions as well as providing independent flight guidance computations (Figure 22-7).

The flight guidance computers receive critical attitude heading system data directly from the attitude heading computers; and receives air data system, radio sensor system, and flight management systems data through the IAPS input/output (110) processor cards. Flight control system mode and autopilot information is displayed on the active flight displays.

Flight guidance computers independently calculate command output and together apply redundancy monitored servo drive to aileron and elevator servos, monitor elevator servo torque, and automatically generate pitch trim output. Flight guidance computers apply rudder commands to the rudder servo. Flight director steering commands and autopilot modes are provided to flight displays for annunciation.

NOTES

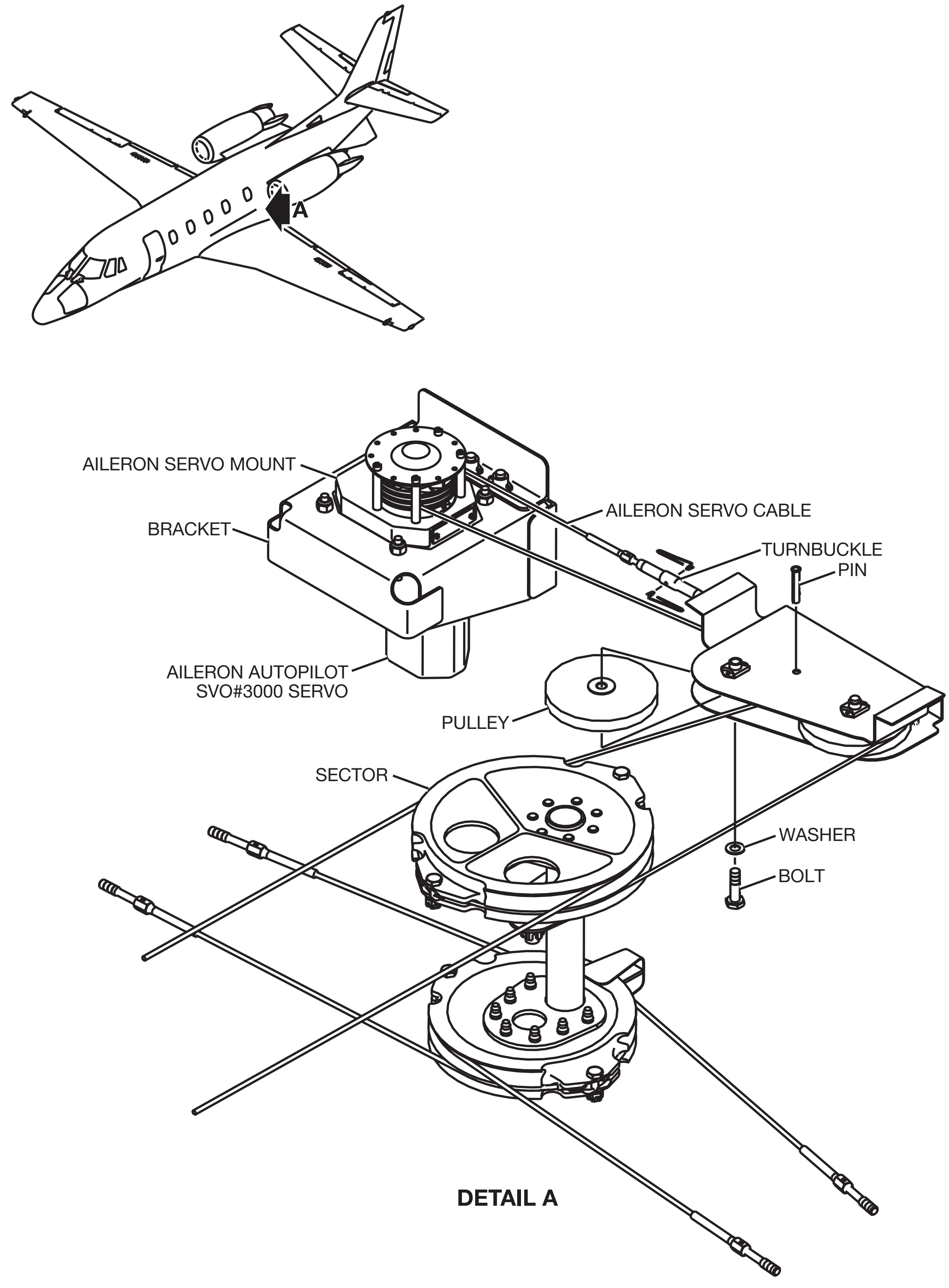


Figure 22-8. Aileron Servo Installation

AUTOPILOT SERVOS

DESCRIPTION

The autopilot system provides auxiliary control of the ailerons, elevators, and rudder. Electrical servos are mounted in the aircraft to drive the flight control airfoils (Figure 22-8).

An aileron servo underneath the fuselage, aft of the wing's rear spar, is connected to the aileron cable network by a servo cable system.

The elevator servo in the tail section aft of the pressure bulkhead connects to the elevator control cables.

The rudder servo in the tail section connects by servo cables directly to the rudder control cables.

COMPONENTS

Servo Bracket and Cable Drum

The servo mount is a cast aluminum housing that has a cable drum assembled to it. The drum accepts the splined shaft of the clutch assembly. The shape of the servo mount housing provides protection for the clutch assembly and synchro when the servo drive assembly is installed in the servo mount.

Removal of the servo mount requires disconnecting servo cables. Consequently, servo cable rigging is required upon reinstallation of the servo mount.

OPERATION

The servo drive is comprised of a motor-tachometer, clutch assembly, synchro and power gear train. The servo drive translates electrical inputs into a rotational mechanical output to drive the servo mount cable drum. Each servo drive contains an integrated DC torque motor tachometer which drives the output engage clutch through a power gear train. The motor position is also transmitted to a synchro through an instrument gear train. This assembly, with a spline output on the clutch, mates with the drum and bracket. The tachometer rate signal is fed to the autopilot computer servo amplifier.

The pitch, roll and yaw servos are electrically driven and provide surface displacement proportional to input signals. Each servo includes an engage clutch, which disengages the servo output shaft and leaves it free to rotate when the autopilot is turned off. The output shaft is connected through the servo drum to the airplane control cables.

Torque limiting of the autopilot servo Is accomplished electrically. The autopilot computer servo amplifier includes a torque limiter and monitor circuit. The current limiter limits the current supplied to the autopilot servo drive motor. Because motor torque is proportional to motor current, the torque is also limited. Normal override at the control wheel drives the servo against the torque established by the torque limiting circuit.

The current monitor system acts as a backup for the current limiters. The limits of the current monitor system are slightly higher than those of the current limiters. If a current limiter fails, the current increases above the level allowed by the current limiter. This increase causes the monitor to disengage the autopilot.

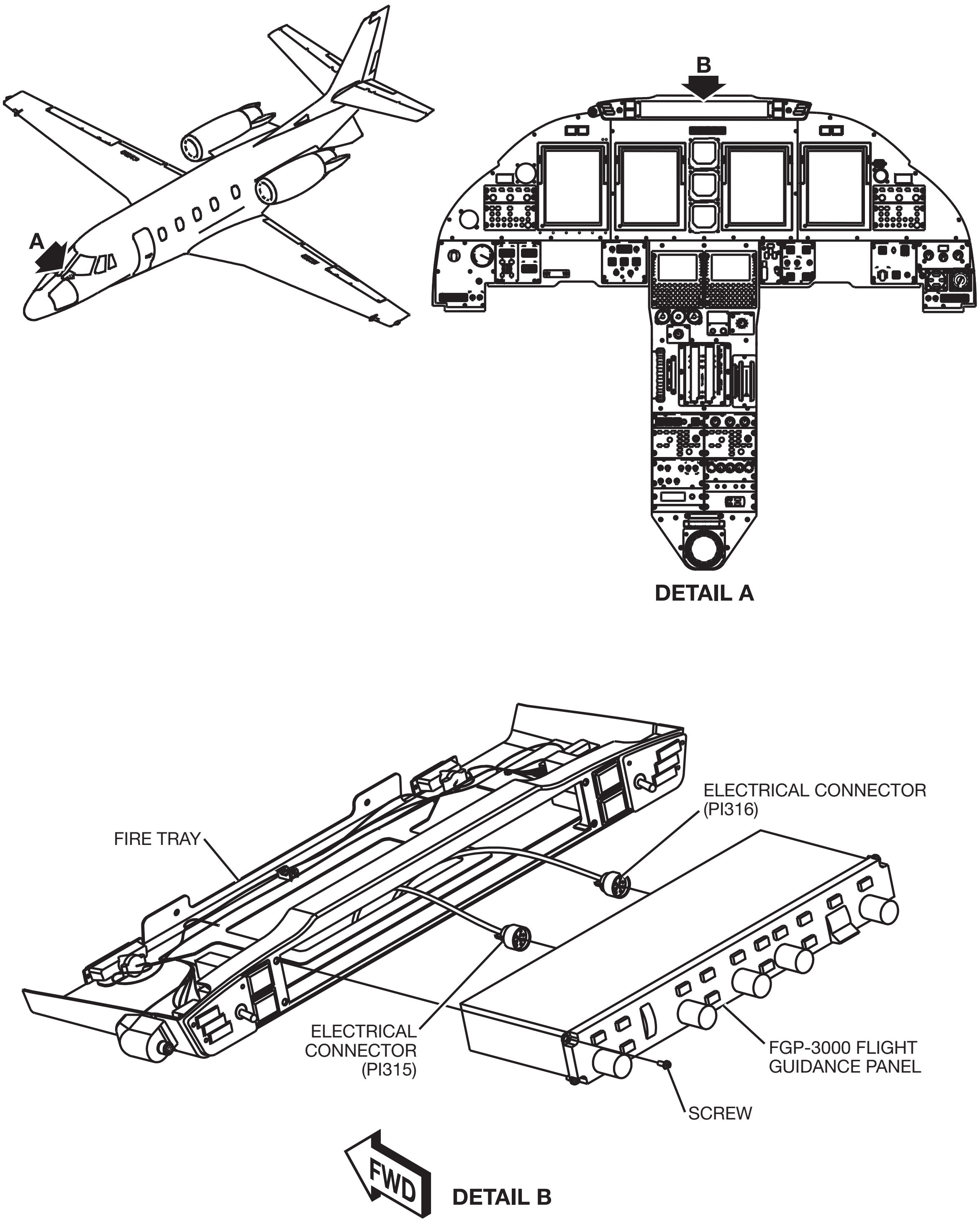


Figure 22-9. Collins Flight Guidance System (FGP-3000)

CONTROLS AND INDICATIONS

Flight Guidance Panel

The Collins flight guidance system has a FGP-3000 FGP that is installed in the upper center instrument panel in the fire tray (Figure 22-9). The FGP has lateral and vertical mode selection switches, heading and course control knobs, a speed knob, an altitude alert/altitude preselect knob, a VS pitch control wheel, and autopilot and yaw damper controls. The autopilot and yaw damper controls include autopilot engage, yaw damper engage, couple switch, and yaw damper/autopilot disconnect. The FGP also has dual flight director controls.

FD buttons—Selects the flight director ON/OFF.

Course knobs—Allows pilot and copilot to independently select courses on their displays. Direct select button located in the center of the course knob automatically points to a previously tuned radio station (VOR or LOC) with no deviation.

VS button—Selects can deselects vertical speed mode.

VNAV—Turns the vertical navigation mode ON/OFF.

Pitch wheel—Allows flight crew to adjust the pitch angles of the aircraft with the autopilot is engaged.

FLC button—Selects or deselects the flight level change enabling/disabling a speed command for climbs and descents.

1/2 BANK button—Reduces the commanded bank angle to 15°.

HDG button—Connects the command bars to the heading bug.

HDG knob—Sets the heading bug on all main displays. The PUSH SYNC button located in the center of the heading knob brings the heading bug to the current aircraft heading.

APPR button—Enables the approach mode of the FGS.

BIC button—Arms the localizer back course mode of the FGS.

ALT button—Selects and deselects altitude hold mode.

ALT knob—Selects an altitude (see in each PFD above the altitude scale) for capture.

YD button—Engages and disengages the yaw dampener.

AP XFR button—Allows the pilot to connect the autopilot to either the pilot or copilot FGC.

AP button—Engages and disengages the autopilot.

YD/AP DISC bar—Disconnects the autopilot and yaw dampener.

CHAPTER 23
COMMUNICATIONS

CONTENTS

ILLUSTRATIONS

CHAPTER 23
COMMUNICATIONS

INTRODUCTION

This chapter describes and provides maintenance information for systems and components that furnish a means of communicating from one part of the aircraft to another, or between the aircraft and other aircraft or ground stations. Also included are the passenger address and voice recording systems. Each Model 560 Excel aircraft is delivered with a complete set of avionics wiring diagrams. These diagrams, which are to be carried aboard the aircraft, must be used in conjunction with this manual when performing maintenance on aircraft. Technical publications, available from manufacturers of components and systems, must be utilized as required for maintenance of those components and systems.

GENERAL

Various antennas are used on the model 560XL/XLS for navigation and communications. The aircraft has a high frequency system to provide long range communications, a digital telecommunications system, an automatic flight information system (AFIS), passenger address, and entertainment systems. Information is also provided on the audio integrating system, static discharging system, cockpit voice recorder, and integrated radio system.

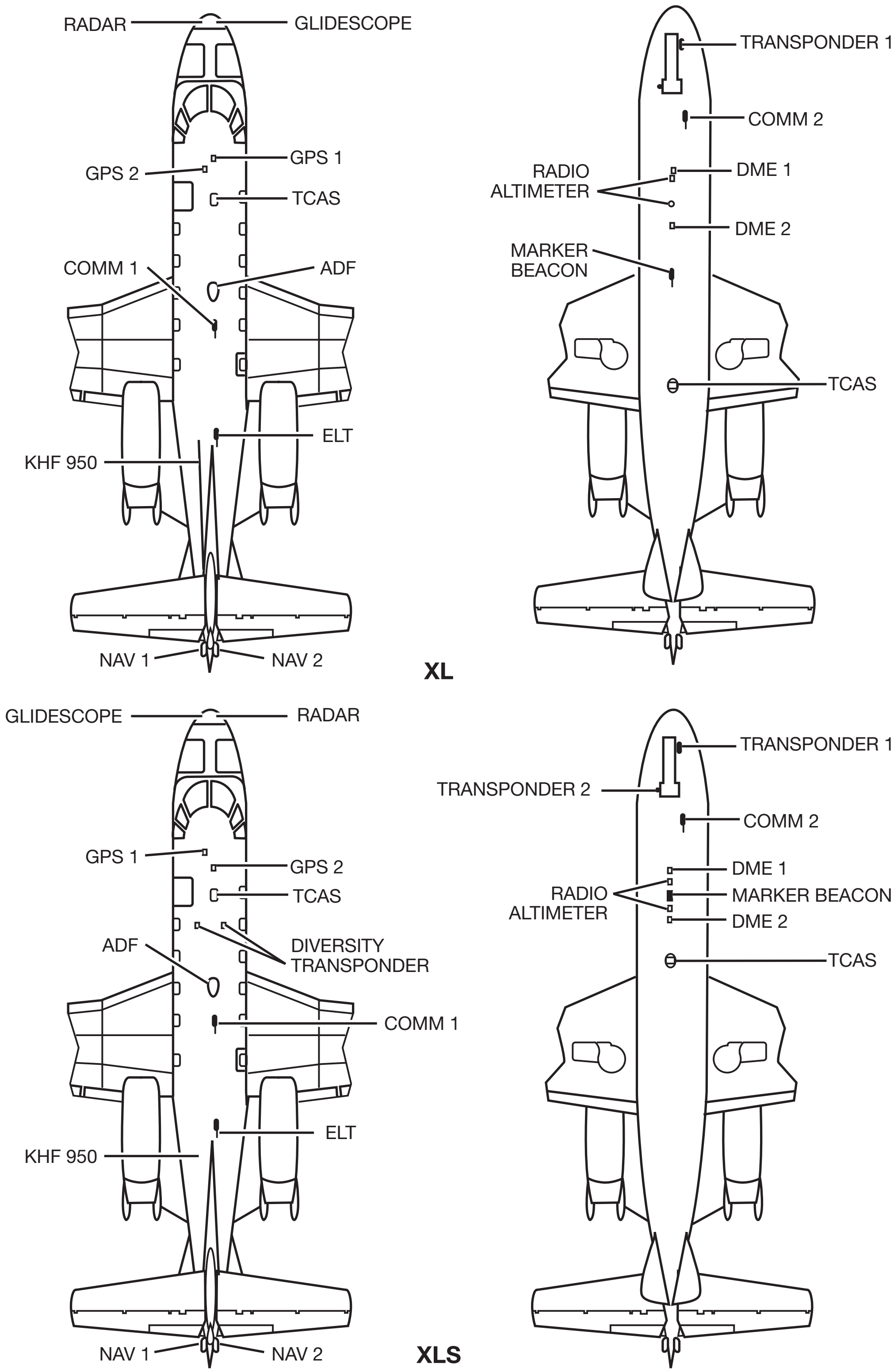


Figure 23-1. Antenna Locations (XL/XLS)

COMMUNICATION AND NAVIGATION ANTENNAS

DESCRIPTION

This section identifies specific antennas and their locations (Figures 23-1 and 23-2).

The antennae on the model 560 XL/XLS/XLS+ include:

- Radar antenna
- Glideslope antenna
- Transponder 1 & 2
- Diversity transponder
- Global positioning system (GPS) 1 and 2
- Communications (COMM) 1 & 2
- Distance measuring equipment (DME) 1 and 2
- Traffic alert and collision avoidance system (TCAS)
- Automatic direction finder (ADF)
- Marker beacon (MB)
- Radio altimeter
- Emergency locator transmitter (ELT)
- KHF 950

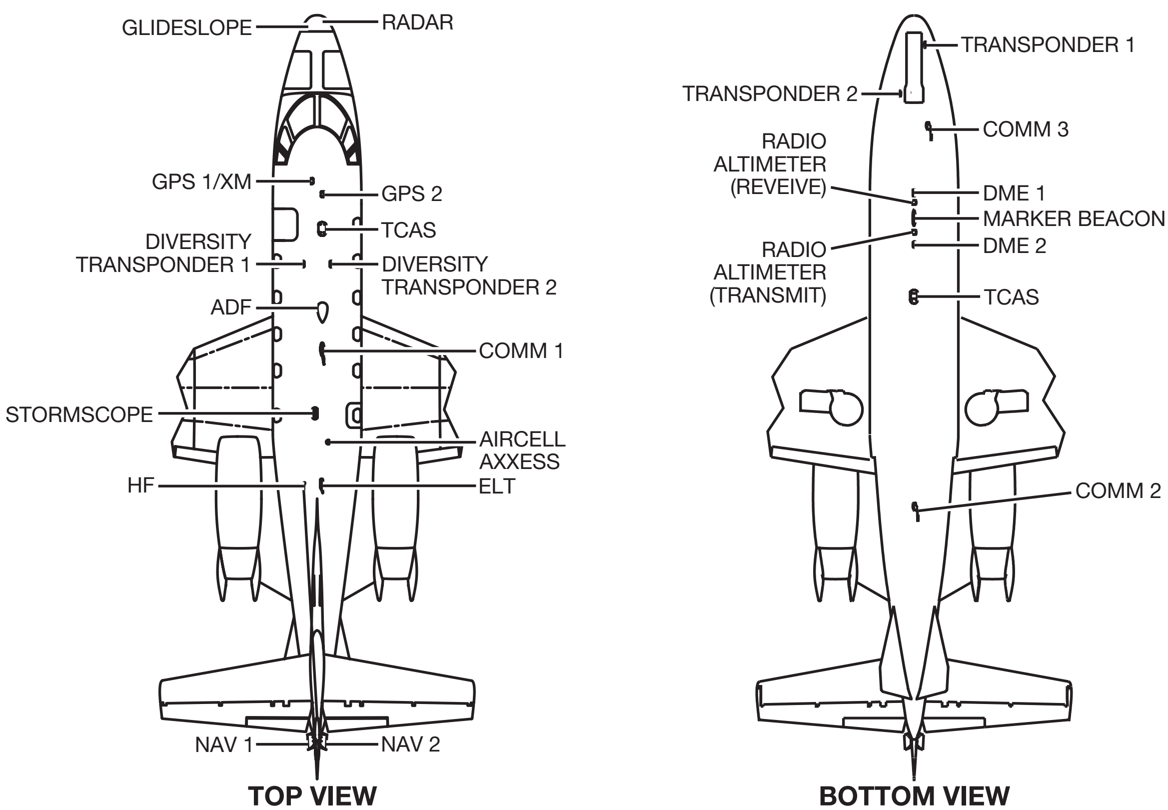


Figure 23-2. Antenna Locations (XLS+)

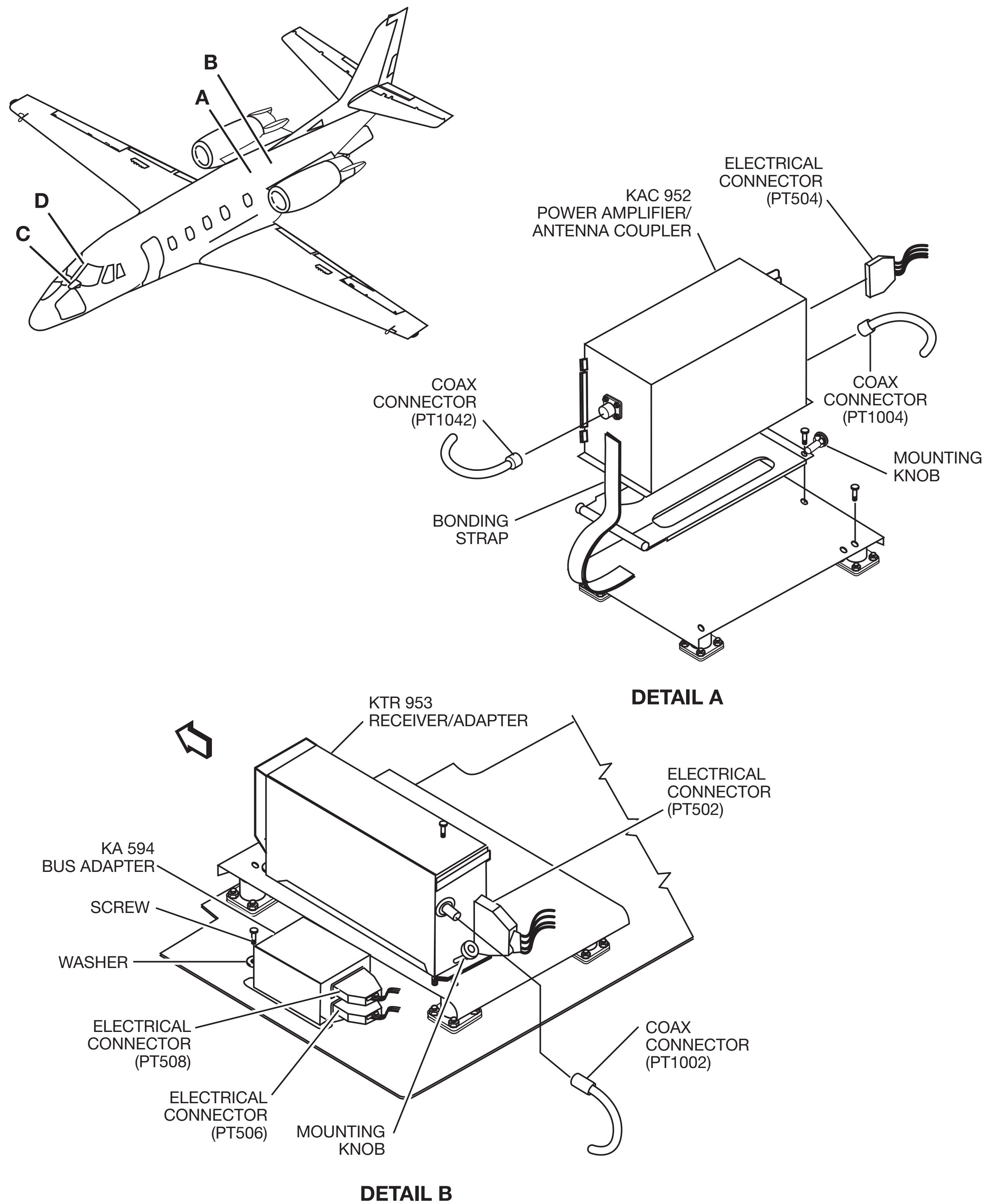


Figure 23-3. KHF-950 HF System Components

KING KHF 950 HF SYSTEM

DESCRIPTION

The King KHF 950 high-frequency (HF) system provides long range communication between air-to-ground and air-to-air stations (Figure 23-3). The system transmits and receives amplitude modulation (AM) and upper side band (USB). Lower side band (LSB) is no longer used in most countries and has been disabled. When LSB is selected with the emission mode switch, the control panel displays an E, unless LSB has been enabled. The KHF 950 system is capable of simplex or semi-duplex operation. In simplex operation, the system transmits and receives in the same frequency. In semi-duplex operation, the system transmits in one frequency and receives in another. The KFS 594 control panel allows the user to define 19 channels, and to program transmit and receive frequencies for these channels. The emission mode selector enables the selection of AM, USB, and 176 International Telecommunication Union (ITU) semi-duplex maritime channels. ITU channels are permanently stored and are accessed when the emission mode switch is positioned in A3J or TEL position.

The KHF 950 HF provides air-to-air and air-to-ground voice communications. HF is capable of transmitting and receiving 280,000 operating frequencies in the 2.0 through 29.9999 MHz range.

The King KHF 950 system consists of:

- KFS 594 control panel
- KA 594 bus adapter
- KTR 953 receiver/exciter
- KAC 952 power amplifier/antenna coupler, and HF relay box.

COMPONENTS

The KA 594 bus adapter is behind panel 321CT in the tail cone baggage compartment.

The KTR 953 receiver/exciter is behind panel 321CT in the tail cone baggage compartment. It provides reception capabilities, and a low power transmit signal to the KAC 952 power amplifier/antenna coupler.

The KAC 952 power amplifier/antenna coupler is behind panel 321CT in the tail cone baggage compartment. It contains a solid-state amplifier which increases the signal from the receiver/exciter to 150 watt peak envelope power, for single sideband or 35 watts (for AM equivalent operation). The KAC 952 also contains a microprocessor-controlled antenna coupler that tunes the antenna to any frequency.

HF relay box is in the copilot side console. It configures the HF audio to transmit or receive.

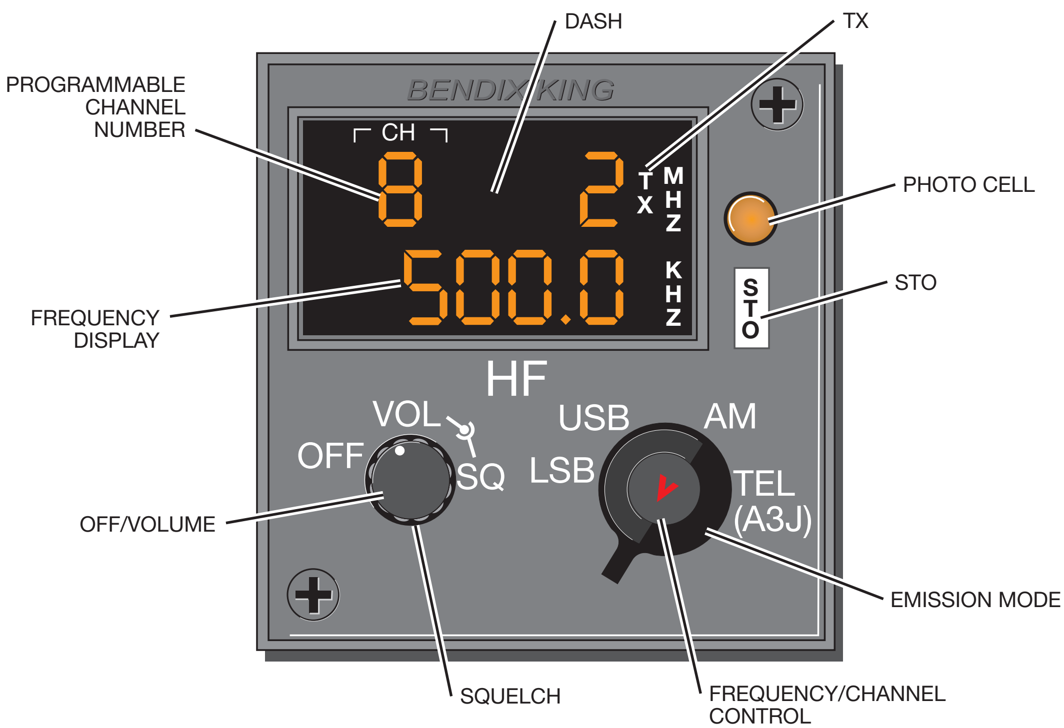


Figure 23-4. HF Controller

CONTROLS AND INDICATIONS

The KFS 594 control panel is on the center instrument panel (Figure 23-4). It works in conjunction with the KA 594 bus adapter to operate the HF system.

Off/Volume—A small knob in the lower left corner of the control panel that controls on/off switch and volume of received audio.

Squelch—A large knob on the lower left corner of the control panel that selects the threshold of the received signal, above which audio should be enabled.

Emission Mode—A large knob on the lower right corner of the control panel that controls emission modes of the radio. When LSB, USB, or AM is selected, the radio is set to the corresponding mode and the control head displays directly-selectable frequency, on one of 19 user programmable channels. When A3J or TEL is selected, the radio operates in corresponding mode and the control head displays an ITU channel.

Frequency/Channel Control—A small knob on the lower right corner of the control panel that when pushed in, moves a cursor (flashing digit) from left to right. When the knob is rotated, the digit selected by the cursor increases or decreases based on the direction of knob rotation.

STO—This switch preforms three functions:

- User is allowed to listen for signals on the transmit frequency in a duplex channel by depressing STO when in channel mode and not in program mode (program mode is noted by flashing dash in the space adjacent to channel number CH). The control panel displays the transmit frequency and illuminates a TX indication.
- Pressing STO the while transmit switch is activated causes the transmission of a 1000 hertz (Hz) tone. This is used to break squelch of some stations.
- Pressing STO while in program mode enters a selected frequency into the channel selected.

NOTE

The transmit switch must be momentarily activated and the antenna allowed to tune before a signal may be transmitted.

If the antenna wire is not properly tensioned, it is necessary to replace the entire antenna wire. It is not possible to salvage antenna wire, because removal requires severing the wire.

Proper electrical bond of all HF system units to aircraft structure ground is of prime importance for proper operation. Bonding to anodized or painted surfaces is not acceptable. Bonding surfaces shall be sanded free of paint or anodize film; and should be joined using screws with washers to ensure maximum surface contact over as large an area as possible.

When performing transmitting tests, proper operating procedures must be used. The operator must be licensed in accordance with Federal Communications Commission rules and regulations. All transmissions shall be identified using the aircraft tail number.

WARNING

Do not touch the antenna or antenna feedline when the radio is transmitting. Painful RF burns may result from high RF voltages.

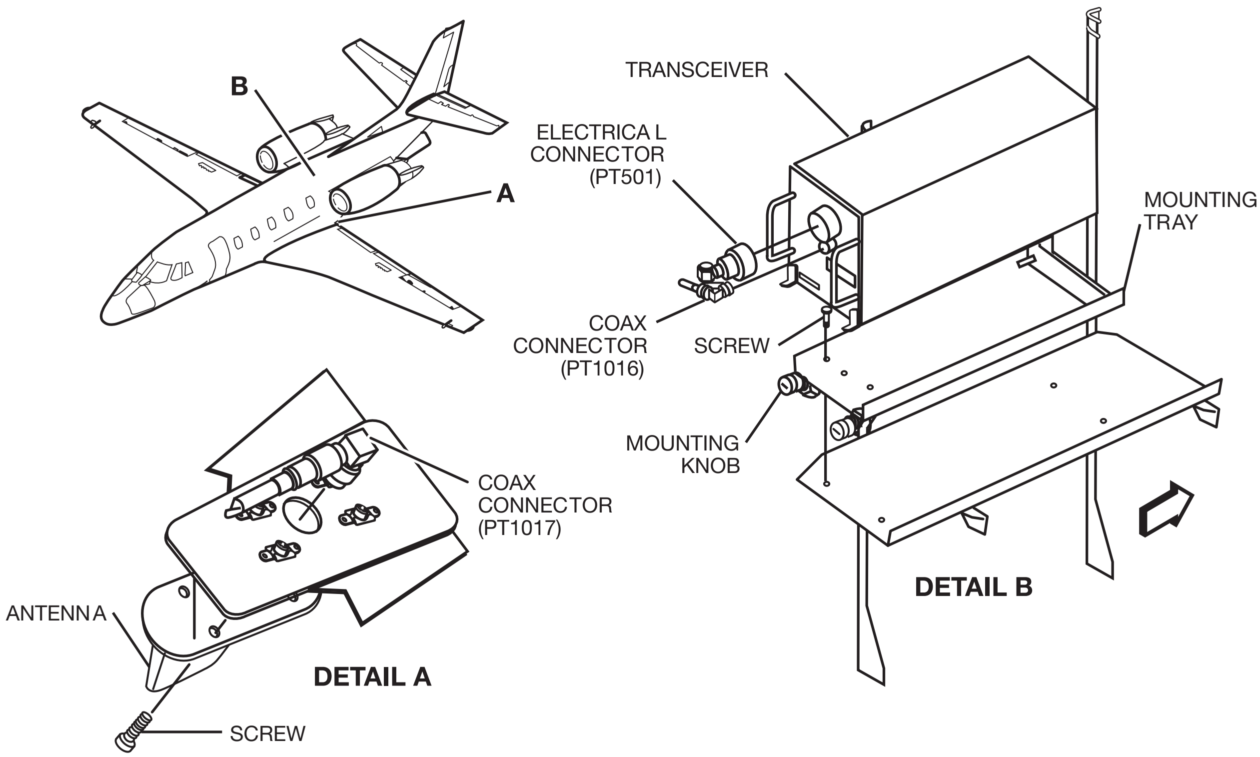


Figure 23-5. Magnastar Components

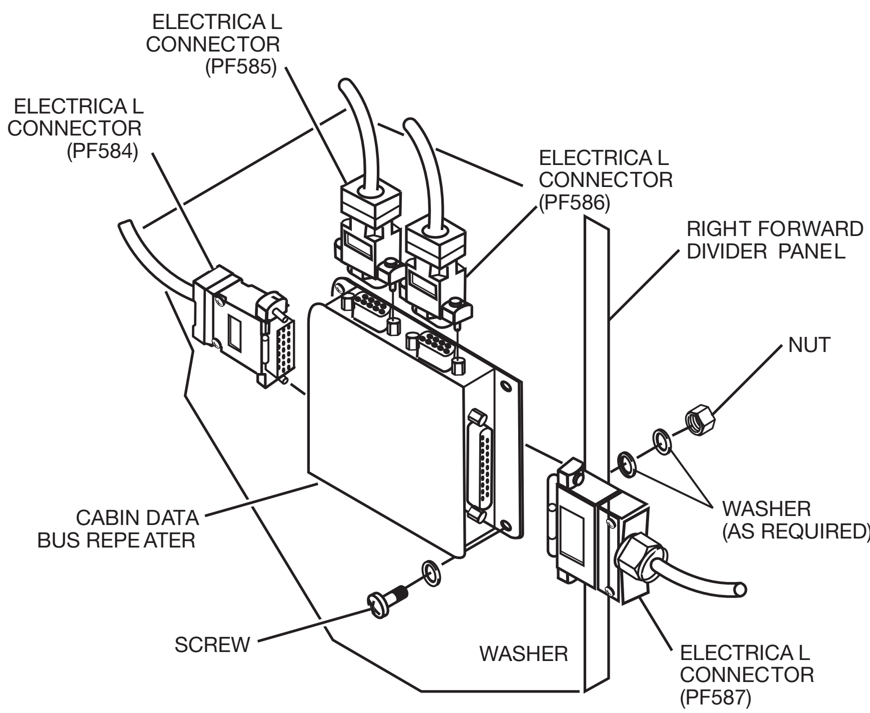


Figure 23-6. Cabin Distribution Components

MAGNASTAR C-2000 DIGITAL AIRBORNE TELEPHONE SYSTEM

DESCRIPTION

The MagnaStar C-2000 digital airborne telephone system is a digital telecommunication system that allows two simultaneous calls of any type:

- Voice, data, fax
- Interphone calls (seat-to-seat)
- Conference calls
- Speed dialing

The system includes digital radio components and cabin distribution components.

COMPONENTS

MagnaStar Antenna

The antenna for the MagnaStar C-2000 digital airborne telephone system is used for both transmit and receive functions (Figure 23-5).

Duplexer

The duplexer combines the transmit and receive lines into a single antenna. It is in the baggage compartment (forward right side) at FS 405.50.

Airborne Radio Telecommunication Unit

The airborne radio telecommunication unit (ARTU) is a full-duplex radio that operates over a frequency range of 849 to 851 MHz for receive functions and 894 to 896 MHz for transmit functions. The ARTU is in the baggage compartment (forward right side) at FS 389.50. A mounting tray allows removal or installation of the ARTU.

Cabin Distribution Components

Cabin distribution components are those components within the cabin that allow interface by a local network to other user equipment (Figure 23-6). A cabin data bus controls the support of:

- Up to nine handsets
- Seven cabin distribution bus repeaters (CDBRs)
- Call alerter switch (CAS)

Cabin Distribution Bus Repeater

The cabin distribution bus repeater (CDBR) interfaces the handset(s) and the CAS to the telephone system. Each CDBR can interconnect two handsets, or one handset and one CAS connected to the high-speed cabin distribution bus.

CONTROLS AND INDICATIONS

Call Alerter Switch

The call alerter switch (CAS) provides 10 switches that are programmable to respond to uplink calls or to any installed handset. The switches can be used to control lights and chimes, etc.

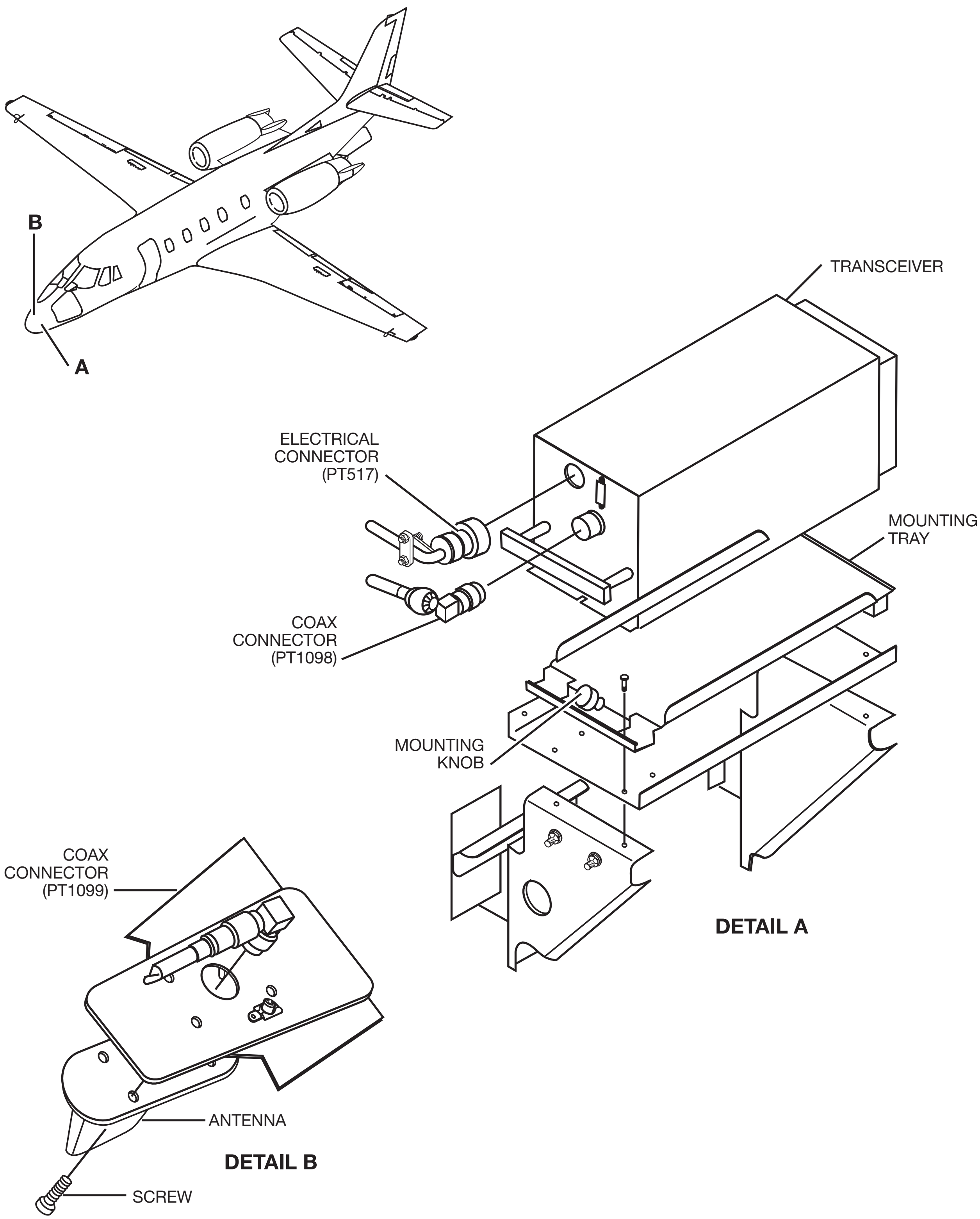


Figure 23-7. Flightfone 800 Components

FLITEFONE 800

DESCRIPTION

The Flitefone 800 radiotelephone is a digital telecommunication system that allows two simultaneous calls of any type:

- (Voice, data, fax)
- Interphone calls (seat-to-seat)
- Conference calls
- Speed dialing
- HF connection
- SATCOM connection.

COMPONENTS

The digital radio components include the RT-800 transceiver, an AT-801 antenna and the WH-800 handset(s) (Figure 23-7).

RT 800 Transceiver

The RT-800 transceiver is a full-duplex radio that operates over a frequency range of 849 to 851 MHz for receive functions; and 894 to 896 MHz for transmit functions. It also provides a forced-air internal-mounted cooling fan. The transceiver is on a tray in the tail cone baggage compartment (at FS 405.50 RBL 22.32).

AT-801 Antenna

The AT-801 omnidirectional blade-type antenna is 3.5 inches tall and is on the bottom of the aircraft at FS 405.50 RBL 27.48.

WH-800 Handsets

All operations of the Flitefone 800 system are performed using the WH-800 handset(s). Each handset has a display and a telephone-style keypad. The keypad is used to dial calls and select various options. Information is provided through a liquid crystal display (LCD). A credit card reader is built into the handset.

NOTES

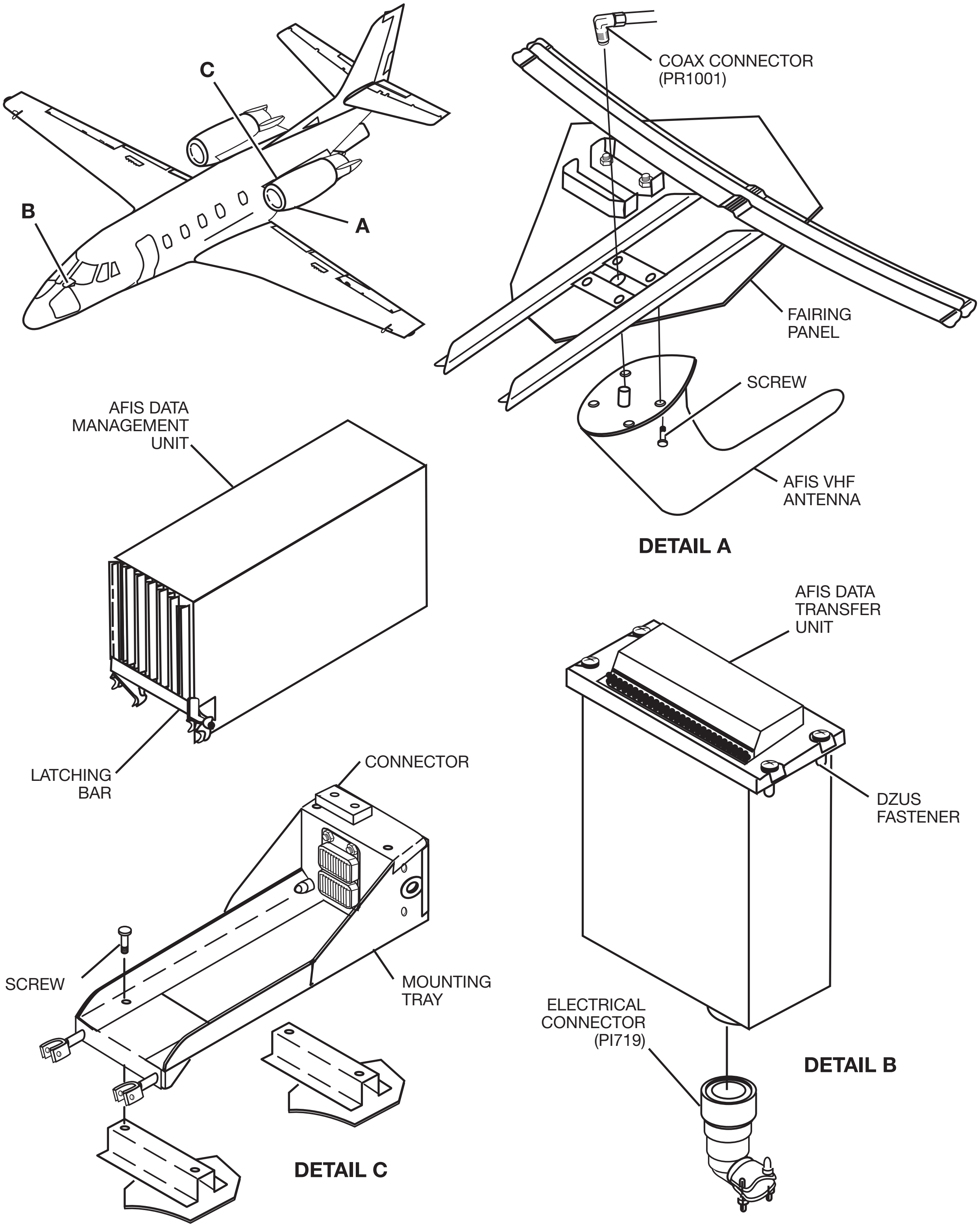


Figure 23-8. Global Automatic Information System (AFIS)

GLOBAL AUTOMATIC FLIGHT INFORMATION SYSTEM

DESCRIPTION

An automatic flight information system (AFIS) provides automatic (VHF) communications with the Global Data Center. The Global Data Center is a ground based computer facility that provides flight planning, aviation weather, and message forwarding services on subscription basis. The Global Data Center has three ways to communicate:

- Via data quality telephone lines to personal computer
- Directly to aircraft via VHF communications network
- Via satellite network.

The global automatic flight information system (AFIS) consists of (Figure 23-8):

- Data management unit (DMU)
- Data transfer unit (DTU)
- VHF AFIS antenna

DMU interfaces with flight management system (FMS). FMS provides operational control and information display for AFIS. DMU transmits and receives data by a data quality VHF transceiver, which is part of DMU.

The DMU computer formats information and presents it to flight management system. DMU automatically tunes its internal VHF transceiver to appropriate ground stations, for the purpose of transmitting and receiving data from the Global Data Center. DMU is behind the upper forward niche panel in the tail cone baggage compartment.

DTU is in the center pedestal and provides a means of updating the DMU data base.

VHF AFIS antenna is on the lower aft of the aircraft at FS 416.72. The system uses this antenna when VHF network is enabled. Enabling or disabling VHF or satellite network is done on the FMS control display unit.

NOTES

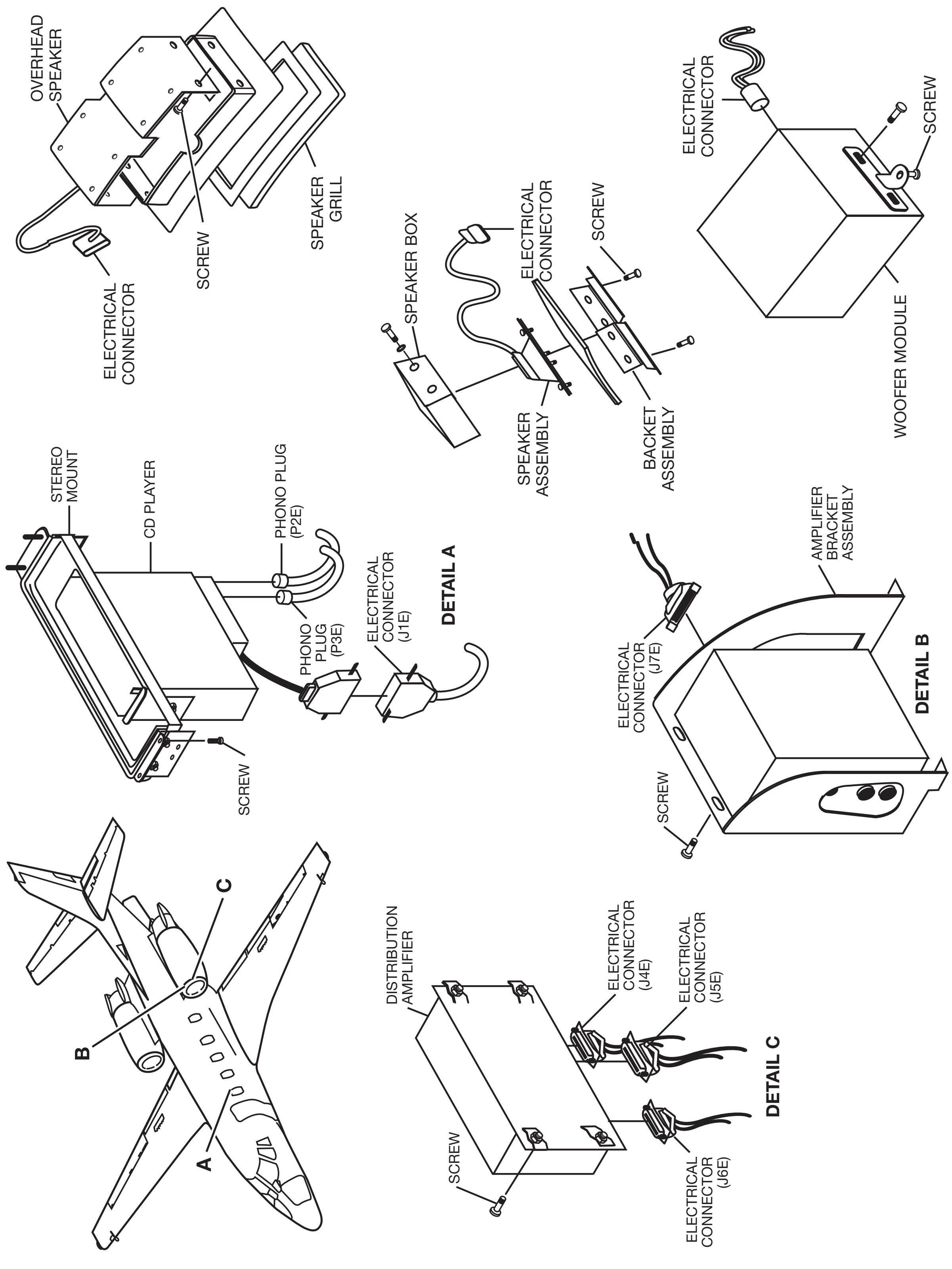


Figure 23-9. Passenger Address and Entertainment

PASSENGER ADDRESS AND ENTERTAINMENT

DESCRIPTION

This section describes systems used to address and entertain the passengers, including: stereos, speakers, chimes and controls.

This section is subdivided into two parts: one part provides information about systems classified as passenger entertainment; and the other part provides information about systems classified as passenger address (Figure 23-9).

PASSENGER ENTERTAINMENT

The stereo system includes:

- CD player
- Audio amplifier
- Distribution amplifier
- Four overhead speakers
- Two woofer modules.

PASSENGER ADDRESS

Passenger Address (Aircraft 5001 through 5036)

The standard passenger address system utilizes four speakers in the cabin left and right upper sidewalls. The controls are in the flight compartment on the audio control panel. The crew (pilot or copilot) may select passenger speaker to address the passengers.

Another passenger address system is the audible chime connected with the seat belt and "no smoking" signs.

Passenger Address (Aircraft 5037 and Subsequent)

The standard passenger address system utilizes four speakers in the cabin left and right upper sidewalls. An additional speaker is in the right aft vanity divider. The controls are in the flight compartment on the audio control panel. The crew (pilot or copilot) may select passenger speaker to address the passengers.

Another passenger address system is the audible chime connected with the seat belt and "no smoking" signs.

NOTES

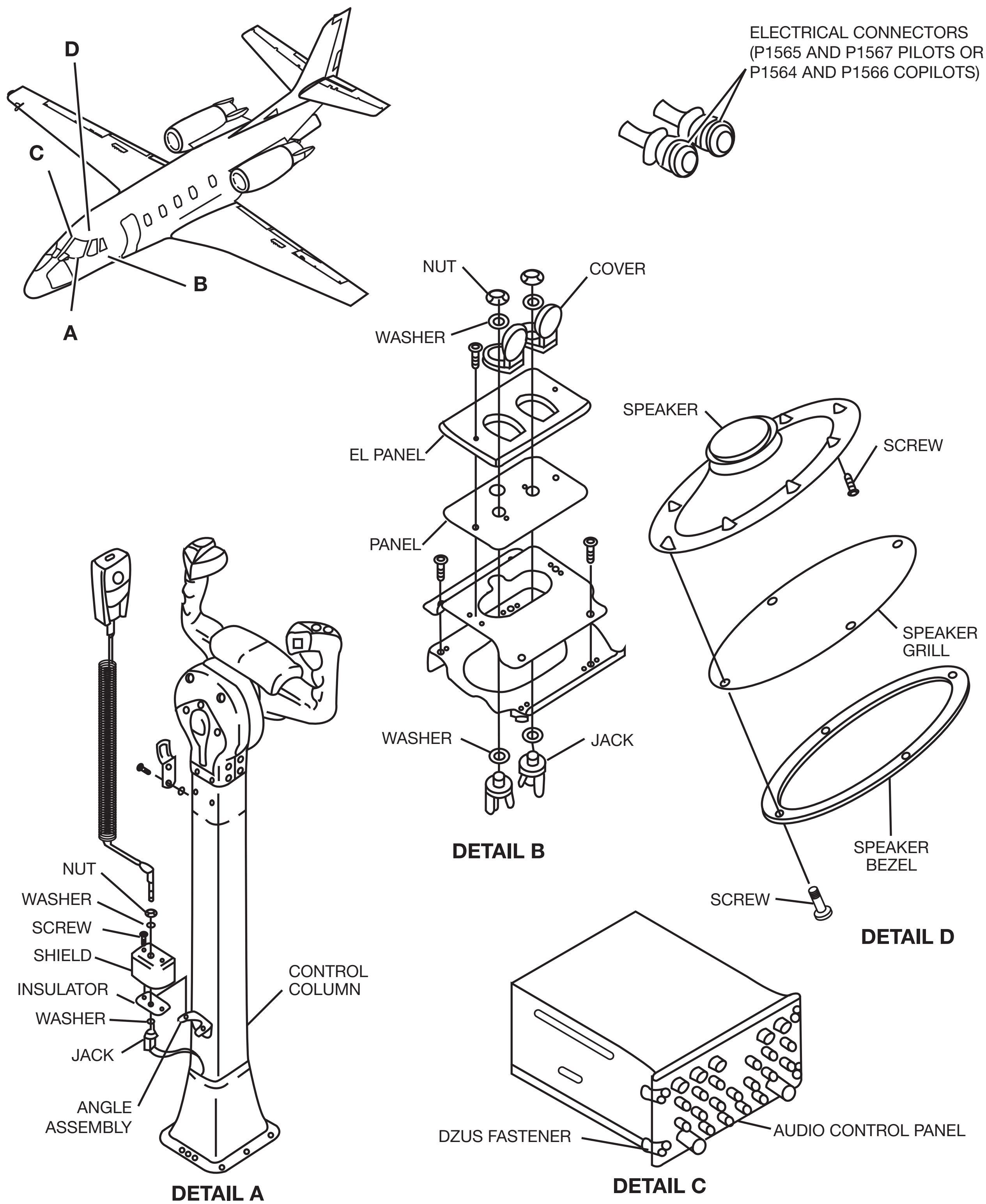


Figure 23-10. Audio Integrating System

AUDIO INTEGRATING SYSTEM

DESCRIPTION

The audio integrating system consists of (Figure 23-10):

- Two self-contained audio control panels
- Two flight compartment speakers
- Jacks for pilot/copilot headsets and microphones

The audio panel receives digitized audio, through a high speed digital audio bus, from each side of aircraft. Each audio panel reconstitutes headphone and speaker audio, for selected sources from digital audio bus. This enables crew members to individually select and regulate the volume of a selected radio.

Audio control panels are in the pilot and copilot instrument panels. These panels provide audio control—both transmission and reception—for communication and navigation equipment in aircraft.

Three jacks on the left console and three jacks on the right console provide headphone/microphone connections.

Two jacks provide for the headphone/microphone and one jack provides for oxygen mask microphone connection.

A microphone jack, located at the forward side of each control column, provides the operator with the option of using a hand-held microphone for audio transmission.

NOTES

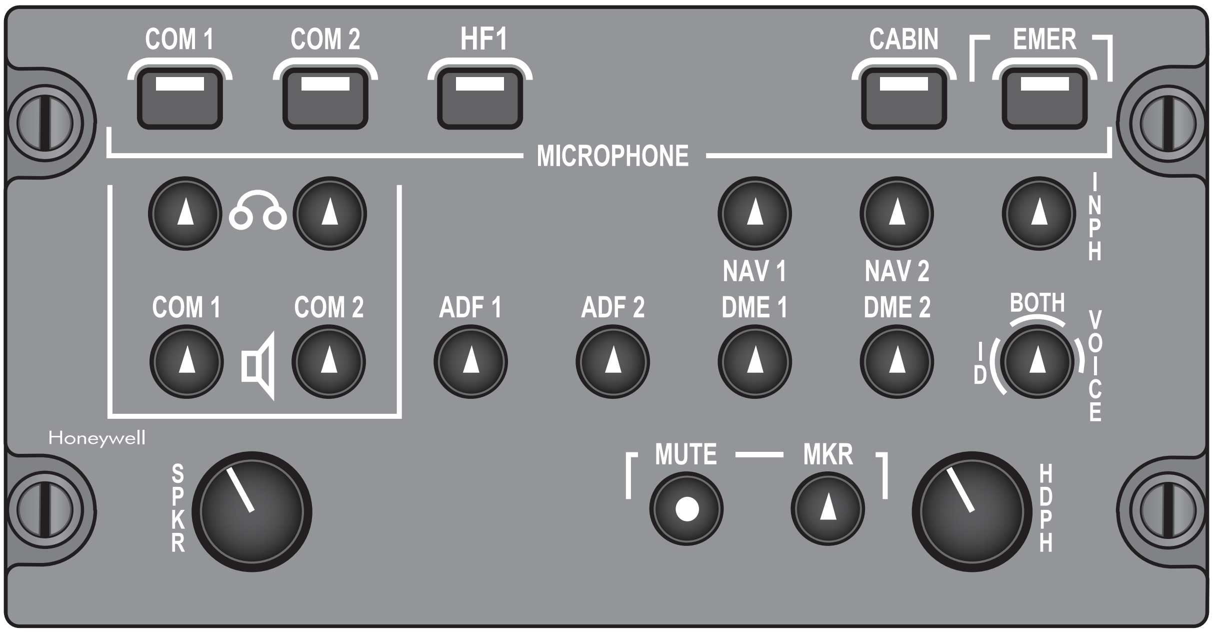


Figure 23-11. Audio Amp

CONTROLS AND INDICATIONS

The microphone input selection controls are rectangular latching switches along upper edge of the audio panel (Figure 23-10). When the switch is latched in, flight crew microphone audio is directed to a selected transceiver, or to a selected intercom channel. In addition, received audio is routed to speakers and headphones at an internally preset level. Level may be adjusted as desired, using audio source selector buttons. The audio source selector buttons are round latching switches (on the face of the audio panel) providing individual audio source selection. Each control combines switch and volume control functions.

There is an emergency (EMER) communications switch is in the upper right corner of the Honeywell audio panel. When EMER position is selected, a microphone is directly connected to the VHF number one communications transceiver. Communications number one and navigation number one audio are connected directly to the crewmember headphones. All electronic circuitry is eliminated in EMER position. In EMER position, warning audio is still heard through flight crew compartment speakers. Microphone audio, emergency phone audio and warning audio are still available for the voice recorder. EMER position disables all other audio panel modes.

The two knobs on the lower edge of the audio panel are the speaker and headphone master volume controls. They are used to adjust the speaker and headphone volume. These controls work in series with the individual controls.

The ID/BOTH/VOICE switch is on the right side of the audio panel. In the ID mode, audio is filtered to enhance the Morse code identification. In the BOTH position, both voice and Morse code may be heard. In the VOICE mode, audio is filtered to enhance the voice content.

There are controls for the marker beacon receiver at the bottom of the audio panel. They include:

- Marker audio volume control
- Marker sensitivity control
- Marker mute control

OPERATION

To transmit, select the desired transmitter on the audio control panel/key microphone, and speak into the keyed microphone.

The hand-held microphone is keyed by depressing a button on the side of microphone. The microphone has a dual, internal switch. One closes the microphone circuit and one closes the keying circuit.

The headphone/microphone is operated by selecting MIC HEADSET on the microphone switch—on the left meter panel (pilot) or right meter panel (copilot)—and by depressing the microphone switch on the control column. Use of the oxygen mask microphone works the same way: by selecting MIC OXY MASK on the selector. When in OXY-MASK position, interphone audio is present in both the speaker and headset.

The intercom works by placing the interphone switch on both control wheels to the INPH position, and speaking into microphone. The system is for use with headsets, enabling the crew to talk to each other without repositioning switches.

For passenger address, position the microphone selector to CABIN/key microphone and speak into the keyed microphone. Use the hand microphone, or if using headsets, use the control wheel push-to-talk switch.

NOTE

Whenever microphone selector switch is in CABIN position, all audio to cabin is interrupted. This includes stereo and passenger briefing systems.

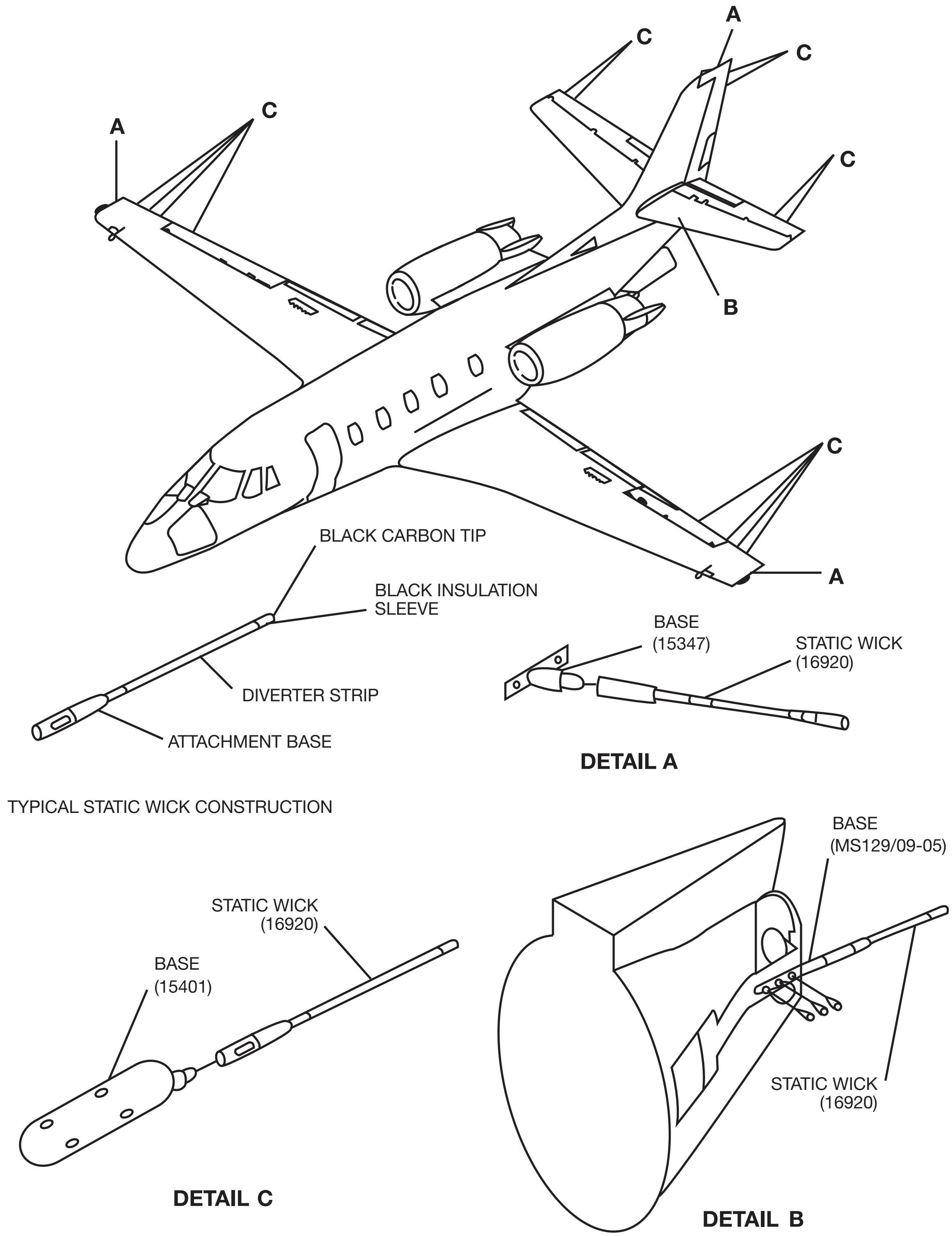


Figure 23-12. Static Wicks

STATIC DISCHARGING

DESCRIPTION

Static dischargers are used to dissipate the static electric charge that accumulates on the aircraft during flight (Figure 23-12). The electric charge is the result of the impingement (on the aircraft) of precipitation and dust particles in the atmosphere. Static dischargers dissipate the accumulated static charge, in order to reduce the noise generated by the associated corona discharge; and to minimize the subsequent noise which is coupled into certain communication and navigation systems. Static dischargers are on outboard trailing edge of wings, wing tips, ailerons, and elevators, vertical stabilizer, tail cone stinger, and rudder.

Static dischargers used on this aircraft are a semiflexible type. Static dischargers are attached to mounting bases, on the aircraft surface.

Mounting bases are not a functional part of the static discharger but serve as static discharger installation devices. Mounting bases are attached to the aircraft surface with screws.

OPERATION

Static dischargers dissipate the static-electric charge that accumulates on the aircraft during flight. Dischargers are a means of controlling the points from which a corona discharge occurs, by keeping a corona threshold level below that of any other point on the aircraft. Dischargers decouple the discharge from the aircraft antenna systems, thus reducing noise coupled into aircraft communication and navigation systems.

NOTES

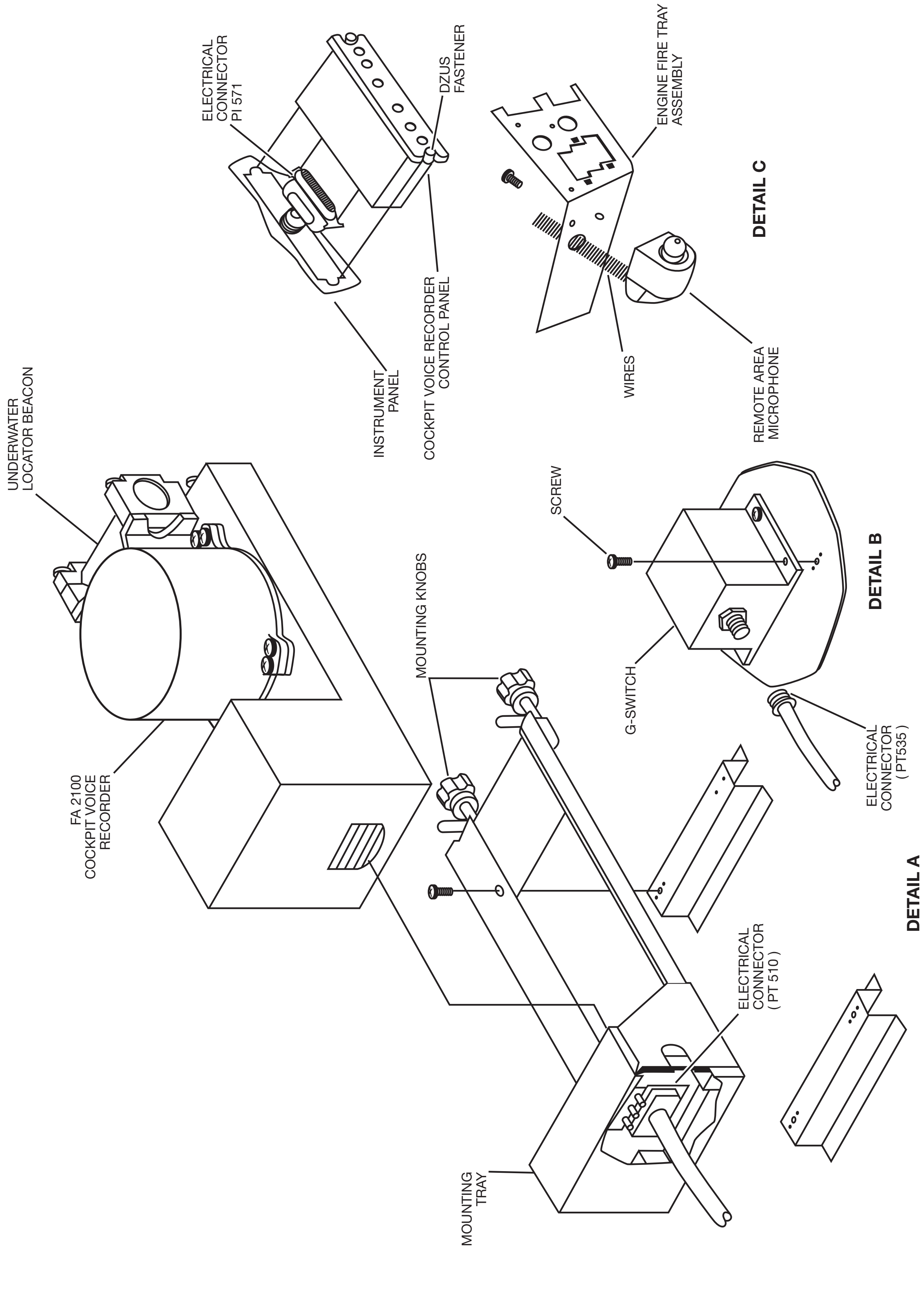


Figure 23-13. Cockpit Voice Recorder

COCKPIT VOICE RECORDER

DESCRIPTION

The Model 560 Excel has either an A200S or FA2100 Fairchild solid-state cockpit voice recorder (Figure 23-13). It can record 30 minutes of high-quality recording for four audio input channels. The voice recorder keeps channels one through three, on channel five in standard quality audio. Channel four (the area microphone) is kept on channel six in standard quality audio. Channels five and six provide 120 minutes of continuous operation.

The voice recorder records and keeps the last 120 minutes of flight crew communication during flight, in case of an aircraft incident investigation. Recordings are made and kept in digital format inside the crash-protected solid-state memory of the voice recorder. Playback is not possible unless the recorder is removed from the aircraft. Recorded communications can be bulk-erased when the aircraft is on the ground. This prevents access to the recordings without approval.

The voice recorder system consists of:

- Voice recorder unit with an underwater locator device
- Voice recorder control
- Remote area microphone
- G-switch
- Electrical relay

The recorder unit has a solid-state recorder assembly and an underwater location device. It is behind the upper forward panel 322BR in the tail cone baggage compartment.

OPERATION

The voice recorder control is found in the right instrument panel and has:

- Test switch
- Bulk erase switch
- Test annunciator
- Headphone jack

There is a remote-area microphone outboard of the pilot thrust reverser annunciators.

A G-switch is included in the cockpit voice recorder system to stop electrical power to the recorder; preventing the recording from being erased after an impact of five G-forces or more. The G-switch is behind panel 322BR in the tail cone baggage compartment.

When the DC POWER BATT switch is placed in the BATT position and CVR, the circuit breaker on left CB panel is engaged, and the CVR is operational.

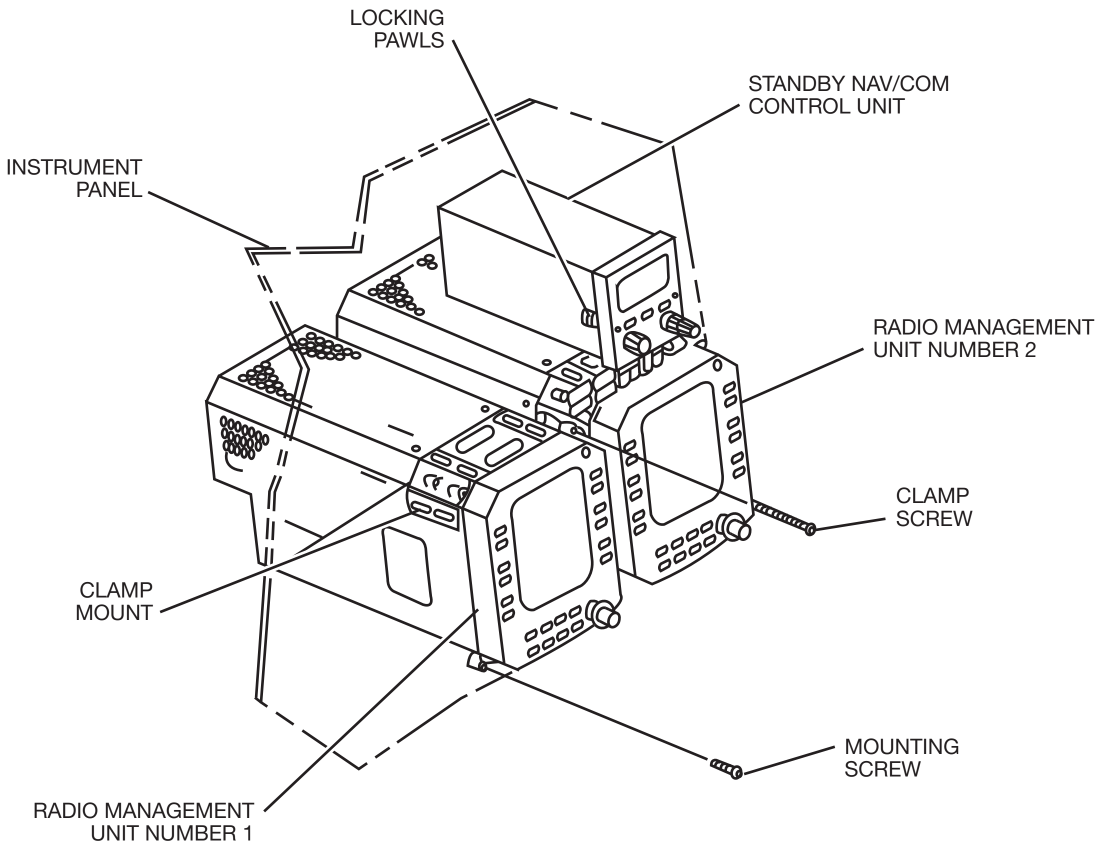


Figure 23-14. Honeywell Radio Controls

HONEYWELL PRIMUS II SZR-850 INTEGRATED RADIO SYSTEM

DESCRIPTION

There are two Primus II SZR-850 Integrated Radio Systems in the aircraft (pilot and copilot). Each integrated radio system consists of two subsystems: RCZ-851 Integrated Communications (COM) Unit, RNZ-850 Integrated Navigation (NAV) Unit, and their associated controls, displays and antennas. Flight crew compartment controls consists of two RM-850 Radio Management Units (RMU), one CD-850 Standby NAV/Com Control Unit, and two audio panels (Figure 23-14).

RMU

The RMU, Standby NAV/COM control unit, and FMS provide frequency and mode control of radios.

RMUs are side-by-side in the center instrument panel. The RMU is the central control unit for the entire radio system. Each RMU is capable of controlling operating mode, frequencies, and codes within all units of the radio system.

The RMU has the ability to switch operation from one side to the other side. The Pilot RMU can control copilot radios; and the copilot RMU can control pilot radios.

The RMU is a color electronic-based controller, featuring function selection by pushing the line-select key next to the parameter to be changed. Selectable parameter, such as VOR station frequency, may be changed by pressing the corresponding line key next to the parameter displayed, and the rotating dual concentric controller tuning knobs.

The RMU display is divided into dedicated windows:

- COM
- NAV
- Transponder
- TCAS
- ADF

RMU has other display modes called pages, which perform additional features and functions for control of the radio system.

NOTES

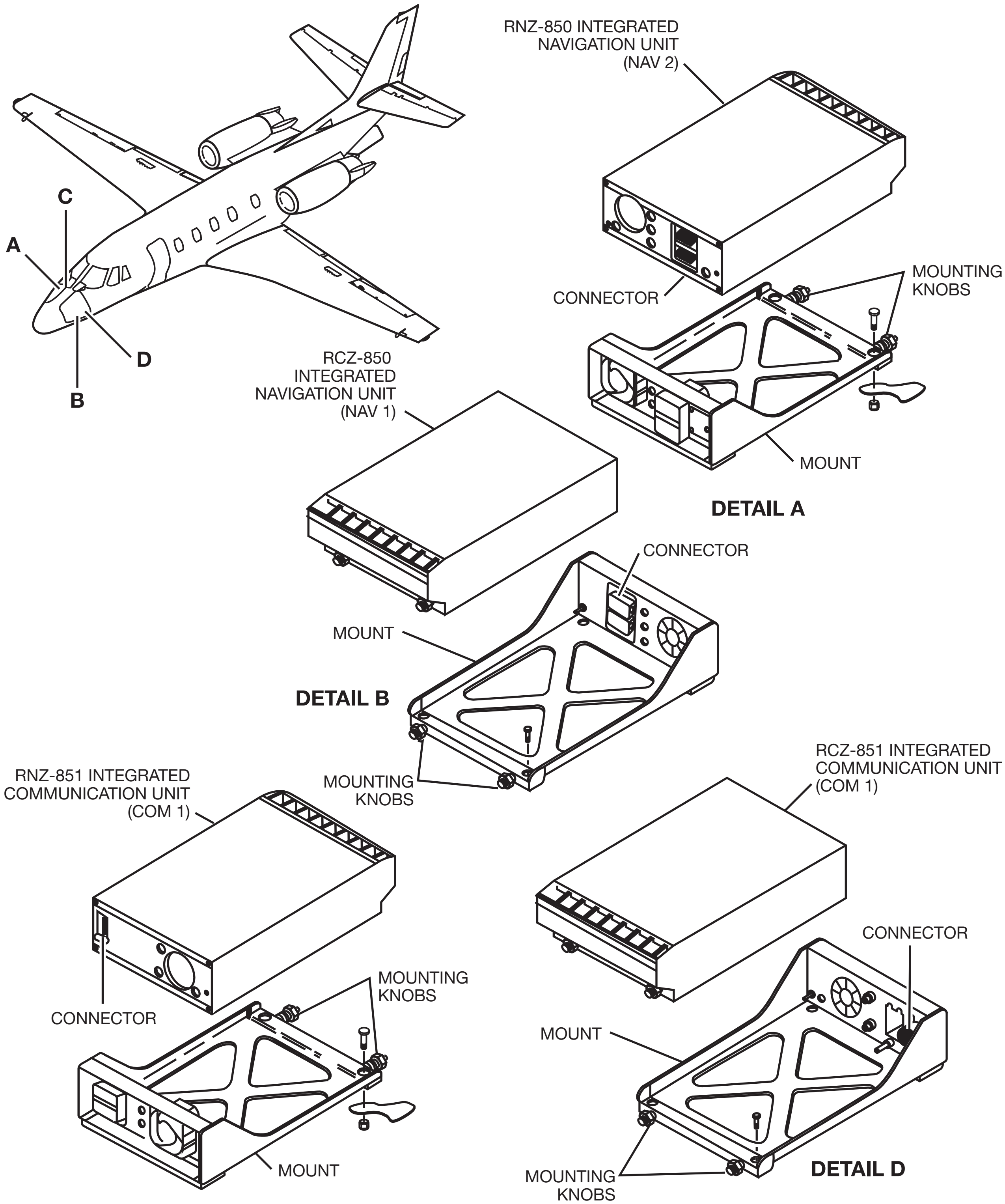


Figure 23-15. Honeywell Integrated Radio System

INTEGRATED NAV UNITS

The RNZ-850 NAV units are in the nose equipment bays (Figure 23-15). The system number one integrated navigation unit (NAV-1) is behind the left nose bay door. System number two integrated navigation unit (NAV-2) is behind the right nose bay door.

Each NAV unit contains:

- NV-850 VHF NAV
- Receiver module
- DM-850 DME transceiver module
- DF-850 ADF receiver module
- Cluster module [radio system bus (RSB)]
- Digitized audio interface

The FMS, when installed, is capable of automatic tuning NAV unit frequencies.

NV-850, VHF NAV receiver module, houses the major navigation functions of:

- VHF omni range (VOR)/Localizer (LOC) receiver
- Glideslope receiver
- Marker beacon receiver

The instrument landing system (ILS) meets Category II instrument landing requirements.

DM-850/distance measuring equipment (DME) module is a six channel scanning DME that tracks four selected DME channels for distance, ground speed, and time to station, as well as monitoring two additional channels for the ident functions. Two of the four channels tracked are dedicated to FMS, when installed.

The flight crew has two channels to:

- Control and display distance
- Time to station
- Ground speed

The FMS preset or standby VOR channel, when selected, provides instant station identification (since it was one of the two monitored channels). DM-850/DME module meets initial approach mode accuracy requirements of P-DME specification, with accuracy that is typically better than 100 feet.

The DF-850/ADF receiver is used in conjunction with the AT-860 ADF antenna. ADF functions over a frequency range of 100 to 1799.5 kilohertz (KHz), in addition to optional operation on marine emergency range of 2181 to 2183 KHz. All frequency ranges are tunable with 0.5 KHz increments.

Cluster module provides RSB communication and digitized audio interface between RMU and radio modules.

INTEGRATED COM UNITS

The RCZ-851 integrated communication (COM) unit, contains internal modules that interface through a cluster module, to the radio system bus for operation (Figure 23-15). Modules within the COM unit are VHF COM transceiver and air traffic control transponder (ATC).

VHF COM Transceiver

The VHF COM transceiver module is a conventional VHF COM transceiver comprised of:

- Receiver
- Synthesizer
- Transmitter
- Power supply
- Audio circuitry

The COM receiver has the unique feature of dual bandwidth, eliminating need for wide and narrow band receivers. Narrow band operation is typically used in more developed areas of the world, while wide band operation is used in lesser developed countries. The COM receiver accommodates a frequency range of 118.00 to 152.00 MHz. Normally COM frequency range is 118.00 to 137.00 MHz. Higher frequency range is normally used in quasi-military operation areas of the world.

There are four ATC modules, which may be included in the RCZ-851 integrated COM unit:

- XS-850 ATC Mode S Transponder
- XS-850A Transponder
- XS-852 Mode S Diversity Transponder
- XI-851 TCAS Interface modules

XS-850 ATC Mode S Transponder

XS-850 ATC Mode S transponder module has encoding/decoding capability, required for Mode S operation; in addition to the capability to operate as a conventional air traffic control radar beacon system (ATCRBS) transponder. Mode S operates with the Federal Aviation Administration (FAA) system, allowing digital addressing of individual aircraft and communication of messages between air and ground. TCAS equipped aircraft transfer data between them through a diversity-type Mode S transponder.

XS-850A Transponder

The XS-850A Transponder module provides only conventional ATCRBS transponder capabilities.

XS-852 Mode S Diversity Transponder

The XS-852 Mode S Diversity Transponder provides full ATCRBS, FAA Mode S, and TCAS data communications capability.

NOTES

XI-851 TCAS Interface

The XI-851 TCAS Interface module allows the integrated communications unit to interface with a separate Mode S diversity transponder and TCAS. The TCAS interface module replaces the XI-851 Mode S transponder module when in the integrated communications unit.

Cluster Module

The cluster module provides RSB communication and digitized audio interface between RMU and radio modules.

NOTES

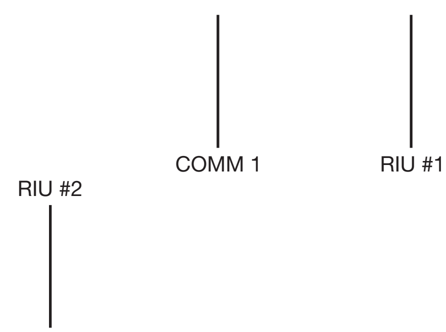


Figure 23-16. COMM and RIU Installation

COLLINS PRO LINE 21

DESCRIPTION

The XLS+ utilizes the Collins dual audio system which is an integrated part of the Collins Pro Line 21 avionics suite. The system has two ACP-4130 audio control panels (ACP) that give the pilots the interface to the audio system and are installed in the pilot's and copilot's instrument panel. The RIU-4110 radio interface units (RIU) interface with the ACPs to give audio to the audio system The system also has dual VHF communication transceivers and an HF communication transceiver.

AUDIO INTEGRATION SYSTEM

Operation

The ACP-4130 ACP give the pilots the primary interface for the Collins dual audio system. The ACP audio selector and volume controls are:

- COM 1
- COM 2
- COM 3 (optional)
- HF (optional)
- PA
- NAV 1
- NAV 2
- DME 1
- DME 2 MKR
- ADF 1 (optional)
- INPH
- V/BOTH/ID
- SPKR
- HDPH

These switches are round push-lock switches. In the outer position, the selected source is sent to the headphones and speaker. In the inner position, the audio will be off for the selected source. In the outer position, you can turn the control to change the volume of the selected source.

The RIU-4110 RID gives the radio data concentration, single communications management, dual audio management, and dual radio control pass-through functions.

As a data concentrator, the RIU has an ARINC-429 connection with each of the radios. Each RID gives dual tuning paths to its on-side radios (primary and secondary).

As the audio management unit, the RIU receives audio inputs from the radios, the ACPs, and other aircraft audio sources and then sends the audio to the audio system. Audio input and output to and from the radios is ARINC-429 digital data or in analog format. All analog signals are converted to digital format at the RIU to give digital mixing and control. The RIUs give dual audio management to the ACPs.

The optional RIU with SELCAL does a check of the audio inputs from the HF communications transceiver for SELCAL tones. The RID gives SELCAL alerts to the pilots when a SELCAL tone is related to the aircraft.

Description

The RIUs also get commands from the DCU to give aural alerts. The aural alerts include Fire Bell, Altitude Alert, Autopilot Disconnect, Landing Gear, Overspeed, Stall, SELCAL, and Phone Call.

Components

ACP-4130 audio control panels (ACP) The ACP-4130 ACP give the pilots the primary interface for the Collins dual audio system (Figure 23-17). Electrically interlocked microphone selector buttons route the microphone to the selected transmitter or interphone.

COM1 , COM2 & HF connect microphone to transmitter of the appropriate radio.

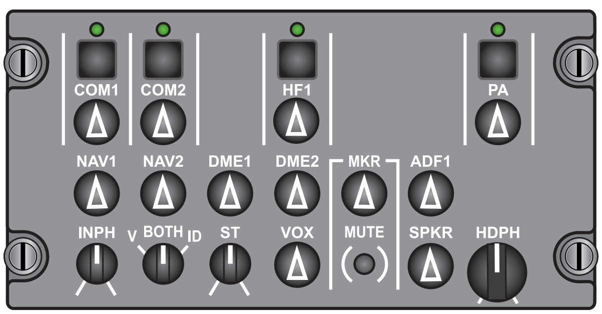


Figure 23-17. Audio Panel

PA connects to the PA system and the audio input from the PA system is connected to the headphones.

A light comes on above the selected microphone switch to indicate it is the active selection and remains on as long as that microphone switch is the active selection. Pushing a microphone selector button automatically deselects the previous selection.

Audio Selector/Preamp Controls

There are 10 audio selector/preamp controls are:

- COM1
- COM2
- HF
- PA
- DME1
- DME2
- ADF1
- MKR
- NAV1
- NAV2

The controls are push-lock switches allowing the flight crew to select the audio source and adjust its gain. In the out position, the selected source is output to the headphones and speaker and the preamp output of the selected source can be adjusted by turning the control. In the in position, audio is off for the selected source.

INPH (interphone) selector/preamp control—Enables crew and service interphone communications. Turning the INPH control adjusts the volume of the interphone audio to the headphone.

SPKR (speaker) selector/volume control—SPKR enables audio to the associated speaker.

HDPH (headphone volume) control—HDPH knob adjusts headphones volume.

V BOTH ID (voice/both/identifier) - controls the voice/identifier code audio input from the NAV and ADF receivers.

ST (sidetone volume) control—Adjusts the local sidetone level to the headphone.

VOX (voice activated) interphone—Enables hot MIC operation for the interphone.

MUTE (marker mute)—Temporarily overrides the MKR selection muting the marker beacon audio.

Microphone jacks on both pilot and copilot control columns supply the flight crew with the option of using a hand-held microphone for audio transmission. The hand-held microphone is keyed when you push the button on the microphone.

The microphone has an internal dual switch. One switch closes the microphone circuit and the other switch closes the keying circuit. The microphone has a potentiometer on the backside to adjust the MIC gain for to prevent feedback.

MIC SELICOCKPIT SPEAKERS Switchlights

MIC SEL HEADSET1 MASK—Switch is used to control the headphone and oxygen mask microphones. The MASK position is used to enable the oxygen mask microphone and disable the headphone microphone. The HEADSET position is used to enable the headphone microphone and disable the mask microphone.

COCKPIT SPEAKERS ON/MUTE—Switch is used to control the transmission of audio to the on-side headphones and speakers. The Collins system has an emergency mode is automatically selected upon loss of power (Figure 23-18).

When emergency mode is active, audio from the VHF COM is transmitted and received directly to the headphones/microphones and no other audio is heard over the speakers or headphones. When emergency mode is not active, audio from all the selected sources is transmitted to the on-side headphones.

INPH—Interphone switch is located on the outboard side of each control yoke and allows the flight crew to communicate with each other via their headphones. Selecting SPKR on the audio control panel allows the flight crew to communicate with the passenger cabin.

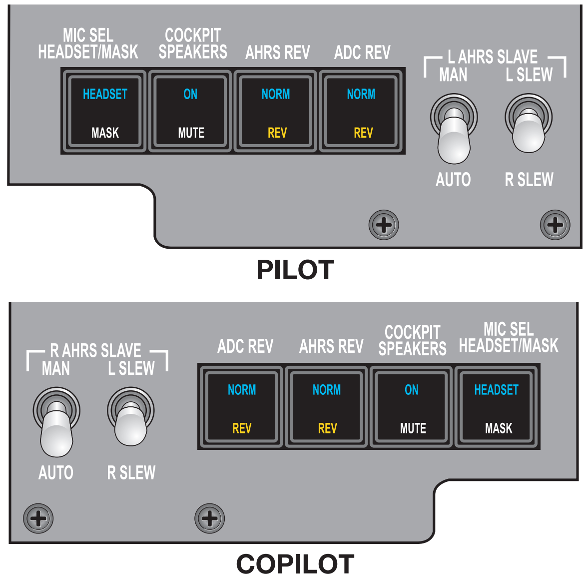


Figure 23-18. Pilot and Copilot COCKPIT SPEAKER and MIC SEL Switchlights

COLLINS DUAL AUDIO SYSTEM

The radio sensor system (RSS) provides the radios and controls/displays used for voice communication, navigation, and operation within the air traffic control (ATC) environment. It is a dual-independent system made up of pilot and copilot side radios/sensors.

Standard (STD) equipment includes:

- Two extended frequency VHF-4000 transceivers
- Two NAV-400014500 receivers (VOR/ILS/MKR)
- Two RIU-4110-4010 RIU
- Two ACP-4130 Audio Control Panels
- One GPS-4000s GPS
- One DME 4000 DME receiver
- Two TTR-4000 ATC Mode-S diversity transponders
- Two TDR-94D TCAS II transmitters

Optional equipment includes:

- Third VHF-4000 COM datalink transceiver
- One or two ADF receivers (located in VHF4000 transceivers)
- One HF-9031A or HF-9041 high frequency (HF) transceiver
- Two transponders with ADS-B capability
- Second DME-4000
- Second GPS-4000s
- One selective call (SELCAL) RIU

The control portion of the system is made up of two CDU-3000 CDU and one CTL-23D back-up NAV/COM control. The RSS provides digital radio data to the electronic flight instrument system (EFIS), navigation systems, and hazard avoidance systems from the IAPS and system bus structure. Each side RSS (pilot and copilot) is functionally isolated and acts as a stand-alone system and can control the cross-side radios/sensors in the event of a control or display failure.

Radio Interface Units (RIU)

The RIU-4110 provides the radio data concentration, single communications management, dual audio management, and dual radio control pass-through functions. As a data concentrator, the RIU has an ARINC-429 connection with each of the radios and gives dual tuning paths to its on-side radios (primary and secondary.)

As the audio management unit, the RIU receives audio inputs from the radios, the audio control panels, and other aircraft audio sources and then sends the audio to the audio system. Audio input and output to and from the radios is ARINC-429 digital data or in analog format. All analog signals are converted to digital format at the RID to give digital mixing and control.

The optional RIU with SELCAL does a check of the audio inputs from the HF communications transceiver for SELCAL tones and gives SELCAL alerts to the flight crew when a SELCAL tone is related to the aircraft.

The RIUs also get commands from the Data Collection Unit (DCU) to give aural alerts. The aural alerts include Fire Bell, Altitude Alert, Autopilot Disconnect, Landing Gear, Overspeed, Stall, SELCAL, and Phone Call.

RIU #1 is located in the left nose avionics bay and RIU #2 is located in the right nose avionics bay.

Central Display Unit

The CDU-3000 provides integrated control of several combinations of aircraft communications and navigation radio subsystems to include the setting of radio frequencies, transponder beacon codes, and system operating modes. The CDU provides primary control of both on-side and cross-side radios from the pilot or copilot position via a radio tuning page (Figure 23-19).

The radio tuning page displays the active and recall frequency for COM1/COM2, the active frequency for NAV1/NAV2, the auto/manual setting for NAV1/NAV2, the active ADF frequency and the ATC code.

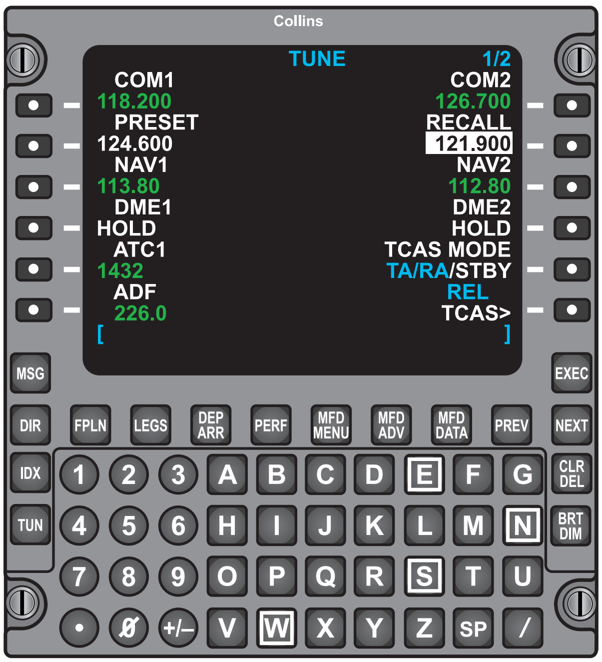


Figure 23-19. Central Display Unit

Cursor Control Panel (CCP)

The CCP-3310 provides radio tuning control in addition to the CDU. Two CCPs are installed, one for the pilot-side and one for the copilot-side (Figure 23-20).

CCP controls:

Radio PUSH SELECT—Located in the center of the RADIO ADV/DATA knob toggles the cursor between the active and inactive states on the CDU. The inactive state consists of the frequency, flight ID, or Radio mode. The active state consists of flashing reverse video on the CDU radio.

RADIO ADV (radio advance) knob—When the cursor is inactive, the RADIO ADV knob is a rotary control used to move the cursor

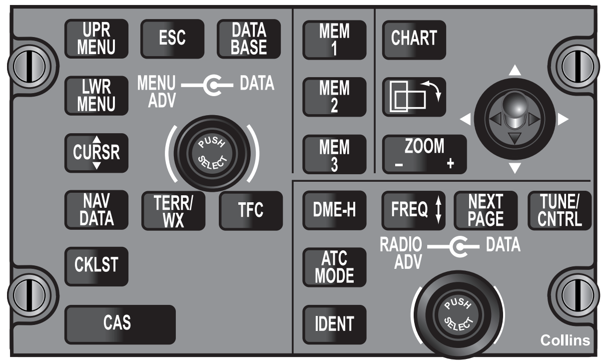


Figure 23-20. Cursor Control Panel

around the page. When the cursor is active and positioned on a frequency, the RADIO ADV knob sets the higher order digits. When the cursor is active and positioned on Flight ID, the RADIO ADV knob moves the cursor a single character at a time.

Radio DATA—When the cursor is active and positioned on a frequency, the radio DATA knob tunes the low order digits. When the cursor is active and positioned on Flight ID, the radio DATA knob changes the highlighted character. When cursor is active and positioned on an item with multiple selections, the radio DATA knob changes the selection.

FREQ (frequency) button—Changes the active and preset frequency when the cursor (active or inactive) is positioned at an active, preset, or recall communication (COM) frequency.

NEXT PAGE button—When the CDU is on the radio TUNE page or radio CONTROL page, the NEXT PAGE button sequences through the available TUNE or radio CONTROL pages.

TUNE/CNTRL button—When the CDU is not on a radio TUNE page the TUNE/ CNTRL button selects the radio TUNE page on the CDU. When the CDU is on a radio TUNE page the TUNE/CNTRL button selects the radio CONTROL page for the radio associated with the current cursor position. When the CDU is on a radio CONTROL page, the TUNE/CNTRL button returns to the radio TUNE page.

CTL-23D Backup Radio Control

The CTL-23D is provided as a backup COM and NAV radio tuning controller in the event that both CDUs fail. It provides control of the COM radio frequency, NAV radio frequency, COM squelch, TX (Transmit) annunciation, and ON/STBY/OFF modes (Figure 23-21).

The CTL-23D controls the pilot-side COM and NAV radio frequencies when the mode select knob is set to the ON position. In the STBY position, the CTL-23D is on but only displays the currently tuned pilot-side COM and NAV radio frequencies.

Figure 23-21. Backup Radio Control

VHF-4000 VHF COMMUNICATION TRANSCEIVER

The Collins dual VHF-4000 communication system is an integrated part of the Collins Pro Line 21 avionics suite and operates in the frequency range of 118.000 MHz through 136.975 MHz. The VHF-4000 can store up to 20 preprogrammed frequencies, show radio diagnostic data, and review or select radio subsystem configurations with the system. The Flight Management System (FMS) is also integrated with the Collins VHF-4000 communication system and has a built-in-test function to make sure that it operates correctly.

The system incorporates a microphone timer that gives you protection from a stuck microphone switch condition. If the microphone switch sticks, there is an auto shutdown function that occurs two minutes after you push the MIC switch.

The #1 receiver is in the left nose equipment bay and the #2 receiver is located above the aft baggage compartment.

HF-9000

The HF-9000 high frequency communication system has 99 programmable preset channels, and 280,000 discrete operational frequencies that range from 2.0 MHz to 29.9999 MHz in 100 Hz steps with selectable RF output power levels of up to 175 watts peak envelope power with an average peak envelope power of 50 watts. Six emergency channels and all 249 ITU maritime radiotelephone network channels are stored in a permanent, nonvolatile memory.

Tuning is done through the CDU on the HF page. Communication is possible with simplex or half duplex operation in upper sideband (USB), lower sideband (LSB), amplitude modulation equivalent (AME), and continuous wave (CW).

SELCAL DATALINK

SELCAL DATALINK - displayed when the SELCAL code is received on the datalink. SELCAL HF 1–2 VHF 1–2–3 0-is played when an HF or VHF message is received for the aircraft (Figure 23-22).

SELCAL DATALINK			
Color	**Inhibited By**		**Debounce**
White	LOPI	TOPI	1 Second

This message is displayed when the SELCAL code is received on the datalink. It produces the SELCAL aural defined in SELCAL HF 1-2 VHF 1-2-3.

Figure 23-22. SELCAL DATALINK CAS Message

NOTES

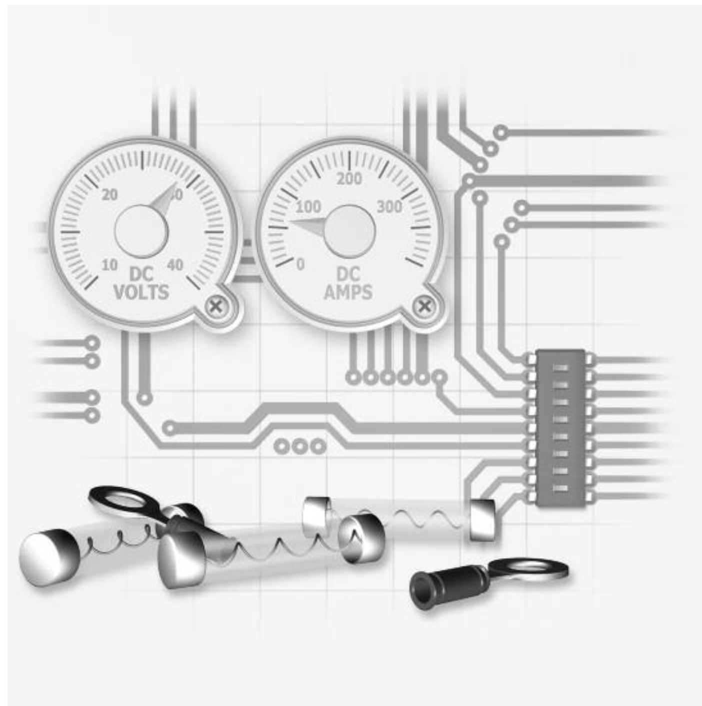
CHAPTER 24
ELECTRICAL POWER

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 24
ELECTRICAL POWER



INTRODUCTION

This chapter describes the electrical power system network used on the Model 560XL/XLS/XLS+. Information is included on direct current (DC) and alternating current (AC) systems. Descriptive coverage of the electrical system consists of power sources, generation, distribution, and system monitoring. Provisions are also made for a limited supply of power during in-flight emergency conditions and for connection of external power while on the ground. References for this chapter and further specific information can be found in Chapters 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 24—"Electrical Power," of the *Aircraft Maintenance Manual (AMM)*.

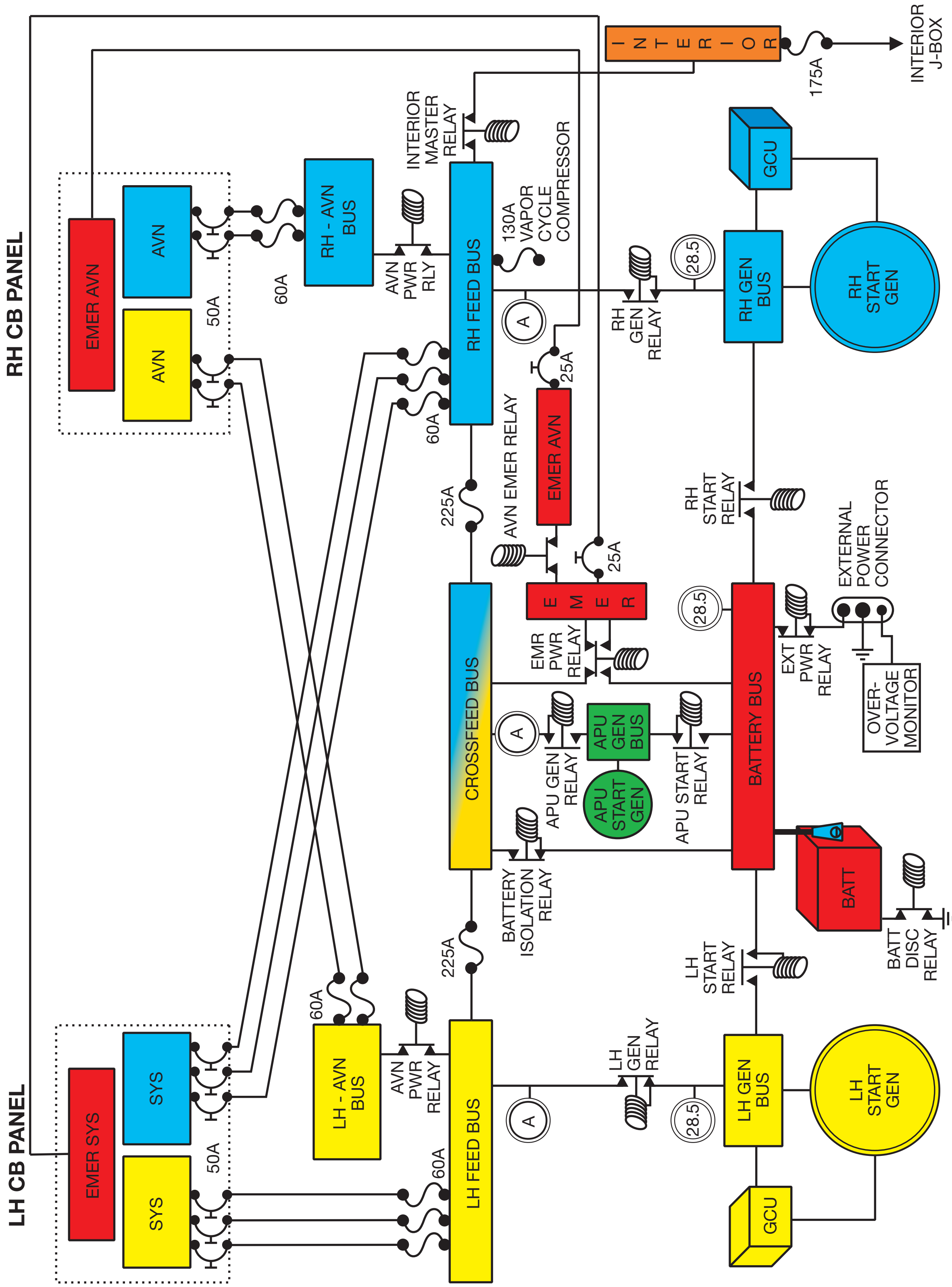


Figure 24-1. XL/XLS Simplified Electrical Bus System

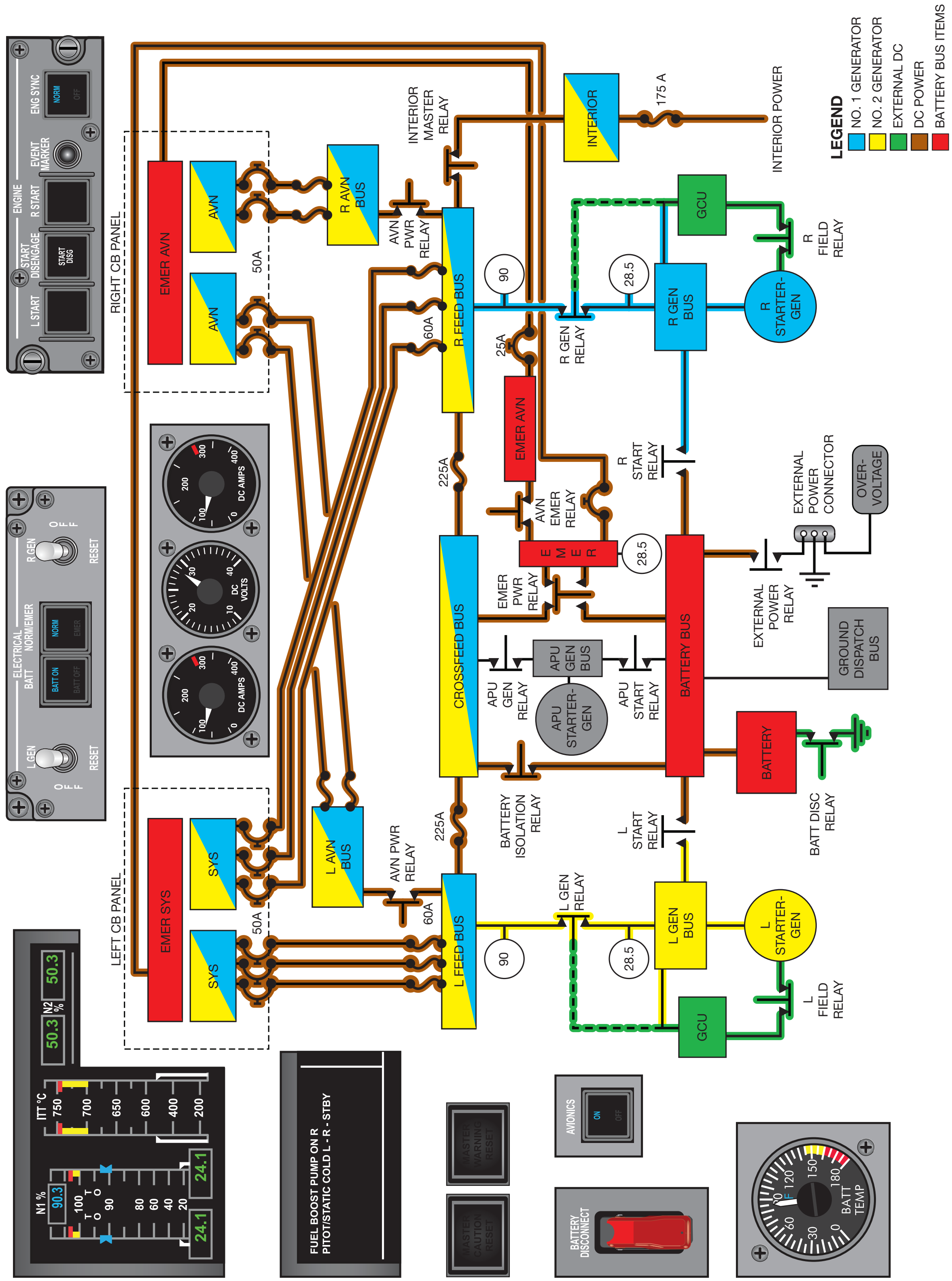
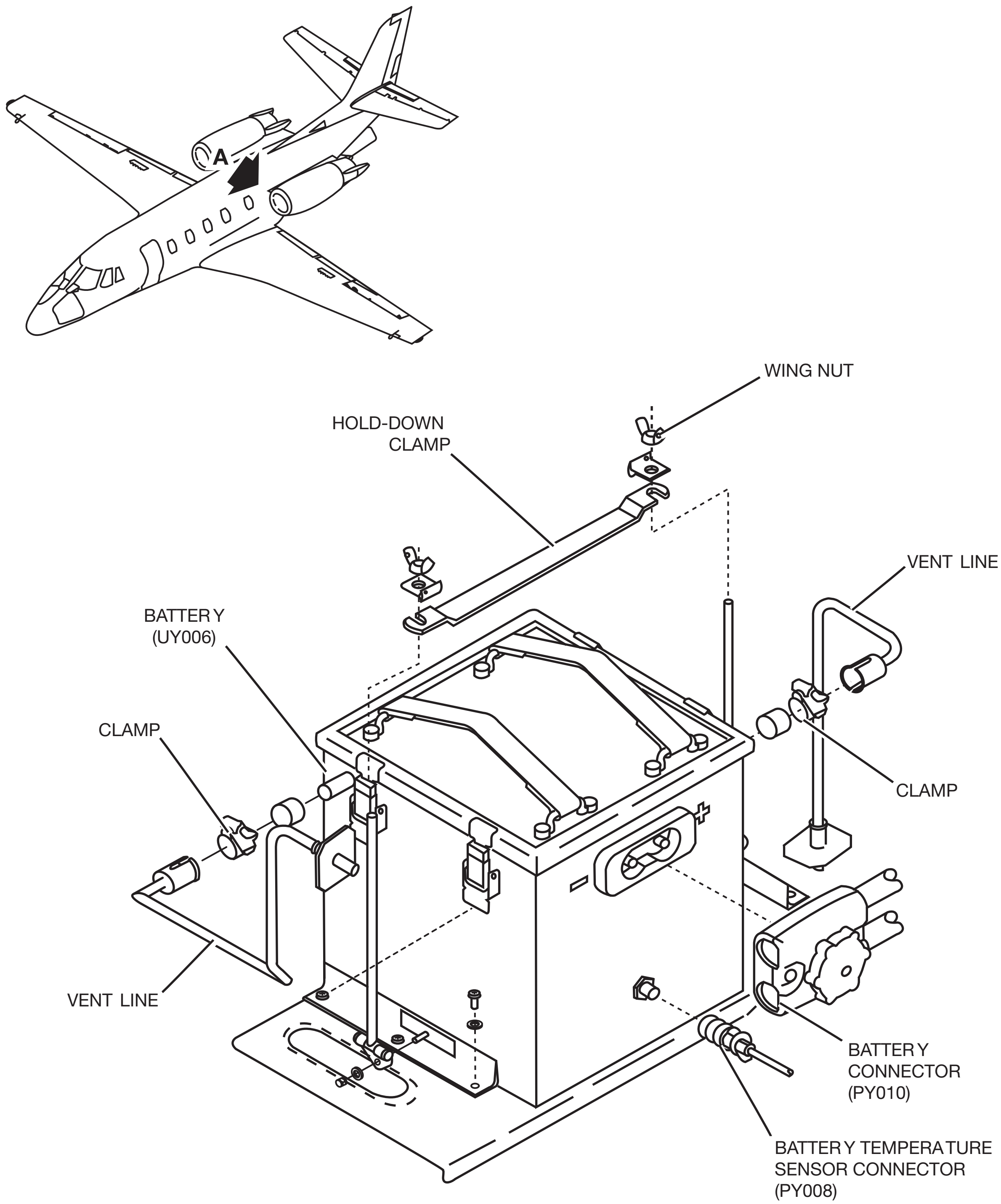


Figure 24-2. XLS+ Simplified Electrical Bus System



DETAIL A

Figure 24-3. Battery Installation

GENERAL

The Model 560XL incorporates DC and AC electrical systems. DC electrical power is required for operation and control of main aircraft systems such as hydraulics, environmental, and anti-ice systems. AC electrical power is required for windshield anti-ice and is provided by an engine-driven alternator. The primary source of DC electrical power is provided by two starter-generators on the engines that are connected in parallel to a common bus system, for equal load sharing. A nickel-cadmium or lead acid battery or an optional onboard auxiliary power unit (APU) provides secondary/backup DC power sources. Provision for connecting an external power supply (EPU) when on the ground is also included.

Current limiters and circuit breakers protect all electrical buses, wiring, and equipment. Backup and emergency power supplies (with associated buses and circuits) are incorporated to provide adequate electrical power for both AC and DC essential equipment during emergency operations. Positioning the battery switch to EMER enables the crew to reduce electrical loads by removing power from nonessential equipment, while maintaining essential electrical power during emergency situations (caused by a loss of primary power). A DC voltmeter, ammeter gauges, annunciator, and master warning switchlights provide monitoring capability for the electrical system.

The electrical system, with source distribution, is illustrated in Figures 24-1 and 24-2.

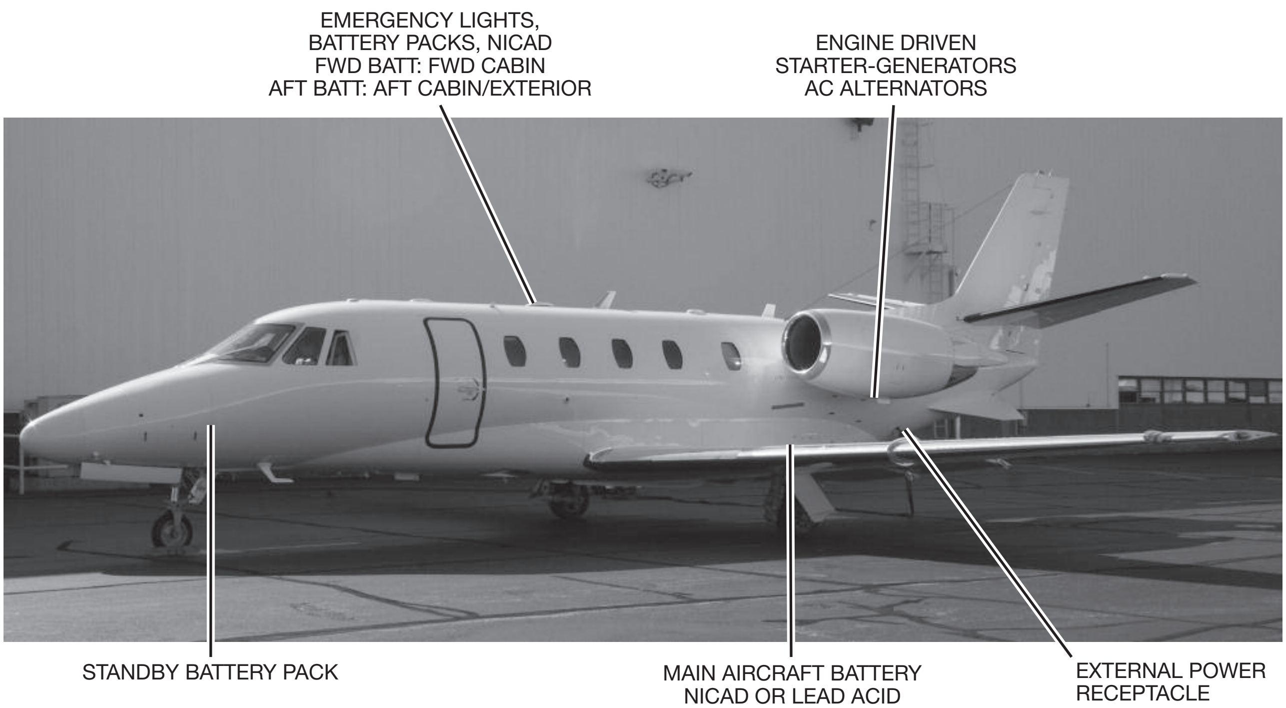


Figure 24-4. Power Source Locations

Table 24-1. EMERGENCY SYSTEM CB PANELS

EMER SYS LH CB Panel (XL/XLS)	
Cockpit floodlights and glare shield lights	Gear control
L and R fan speed (N1)	Hydraulic control
STBY HSI	Stabilizer control
STBY P/S heater	Gear warning
L and R Ignition	Ignition
Flap control	Passenger safety
EMER SYS RH CB Panel	
Audio 1 and 2	Comm 1
Nav 1	AHRS 2
RMU 1	Audio 1 and 2
STBY radio control head	

EMER SYS LH CB Panel (XLS+)	
STBY engine instruments	CVR
Floodlights	Gear control
Dimming	Flap control
STBY P/S heater	Gear Warn
Ignition CH B	Stab control
DCU CH B	Hydraulic control
EMER SYS RH CB Panel	
Audio 1/2*	CDU 1
COMM 1*	STBY Tuner
NAV 1	STBY HSI
XPDR 1	STBY ATT
RIU 1B*	STBY MAG
MFD 1	L-IAPS*
CCP	DCU 1*
PA Amp	GPS 1*
AHRS STBY 1/2	DBU*

Via *GRND DISPATCH

DC POWER

COMPONENTS

Main Aircraft Battery

In addition to the two DC generators, a standard 44 amp/hours, nickel cadmium (NiCad) battery is installed with provisions for an optional lead-acid battery if desired. The battery is inside a dedicated compartment and accessed through a door on the left side of the fuselage just behind the wing fairing (Figure 24-4). The battery is connected to the battery bus by a manual quick connect/disconnect knob on the battery case (Figure 24-5).

Figure 24-5. Battery Compartment

Battery power can only be used for short periods, normally on the ground, for engine starting, and as an emergency power source during in-flight operations.

The battery is limited in its ability to satisfy all aircraft electrical requirements. If operating on battery power only, the electrical system is deigned for the crew to manually shed the majority of the electrical load to prolong battery life. This procedure becomes necessary if both generators are inoperative and the battery is the only source of DC power.

Load shedding allows the battery to provide power to critical systems for a limited time (approximately 30 minutes), otherwise, the battery only powers the entire electrical system for approximately 10 minutes.

NOTES

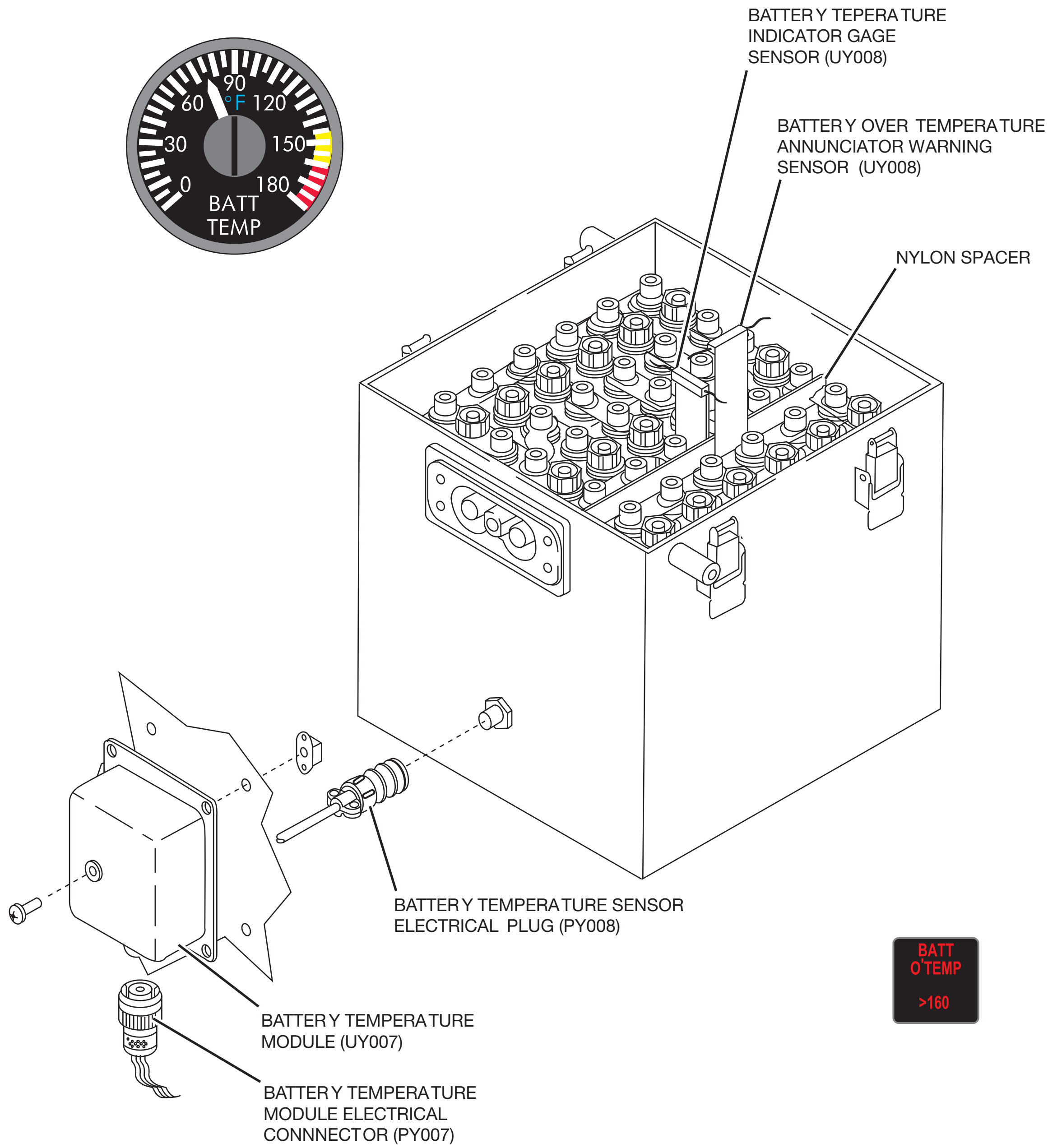


Figure 24-6. Battery Temperature Monitoring

CONTROLS AND INDICATIONS

Battery Temperature Gauge

The battery temperature gauge is installed in addition to the BATT O'TEMP annunciator.

The system provides the flight crew with a continuous indication of battery temperature from 0 to 180°F (–17.77 to 82.22°C) (Figure 24-6).

The battery temperature gauge consists of a battery temperature sensor and gauge. The temperature sensor is installed between cells of the battery to measure the temperature of cells from top to bottom.

The battery temperature gauge is in the right instrument panel. It has a yellow band from 145 to 160°F (62.77 to 71.11°C) and a red band from 160 to 180°F (71.11 to 82.22°C).

The battery temperature gauge operates on 28 volts direct current and it may be operationally checked using the rotary TEST switch.

Battery Overheat Warning

XL/XLS

The battery overheat warning system consists of a battery temperature sensor, a remotely mounted battery temperature module, and a BATT O'TEMP/> 160°F (71°C) annunciator (Figure 24-7). The system is installed to provide the pilot with a visual indication of a battery overheat condition with impending damage.

The battery temperature sensor is installed between cells of the battery to measure temperature of cells at the center of the battery. The BATT O'TEMP/> 160°F annunciator is split horizontally, with the upper half reading BATT O'TEMP and the lower half reading > 160°F.

The battery overheat warning system illuminates a red BATT O'TEMP annunciator when the internal battery temperature exceeds 145°F (63°C). When the internal battery temperature reaches 160°F (71°C) the > 160°F annunciator begins flashing 2 to 3 times per second along with the BATT O'TEMP annunciator.

XLS+

Battery temperature is monitored by a temperature gauge in the pilot side instrument panel. The red BATTERY OVERTEMP >145 and BATTERY OVERTEMP >160 CAS messages appear as appropriate if battery temperature becomes excessive (Figure 24-7). The appearance of either CAS message causes the MASTER WARNING RESET switchlights to flash and an aural "Battery Overtemp" warning to announce.

BATTERY OVER TEMP
Flashes if battery temperature is >145°F. Activates MASTER WARNING lights. If battery temperature increases >160°F, entire light element commences to flash, activates MASTER WARNING lights. This annunciation is triggered by a dedicated sensor independent of the battery temperature gauge. Because the battery temperature gauge uses a separate sensor, the gauge can be used to check the validity of the red annunciator.

XL/XLS ANNUNCATOR

BATTERY OVERTEMP > xxx			
Color	**Inhibited By**		**Debounce**
Red	LOPI	TOPI	8 Second

"xxx" = 145 or 160
This message is displayed when the battery temperature sensor measures above 145°F or 160°F.
This is implemented as 2 messages in the Collins CAS system, one with 145, and the other with 160. However, both messages will not display at the same time. There is an 8 second time delay off for each message. For input characteristics, see Battery Temp Sensor Chart. This CAS message is also accompanied by a "BATTERY OVERTEMP" aural voice alert. The message may also be cross-checked against the Battery Temp gauge on the LH instrument panel.

XLS+ CAS MESSAGES

Figure 24-7. Battery Overtemperature Indications

In addition, the MASTER WARNING RESET switchlight is illuminated with either annunciator. The MASTER WARNING RESET illumination can be extinguished by pressing the MASTER WARNING RESET switchlight.

The rotary test switch may be used to check the operation of the system. The test switch simulates a temperature of 160°F (71°C) or over and causes both annunciators to flash.

NOTE

Do not attempt any kind of start with battery voltage below 24 VDC. This indicates a problem with the battery and maintenance action is required (Table 24-2).

Table 24-2. BATTERY LIMITATIONS

TYPE OF START	COUNTS AGAINST BATTERY
BATTERY START	1
GENERATOR ASSISTED START	1/3
EXTERNAL POWER START	0
APU START	1/3
ENGINE START USING APU	1/3
AIRBORNE START	1

Monitoring

XL/XLS

Battery voltage may be checked with the voltmeter, however the VOLTAGE SEL switch must be in the BATT (spring-loaded) position and the battery isolated from the generators (Figures 24-8 and 24-9). The voltmeter is connected to the BATTERY BUS with the BATT switch in the BATT or EMER position.

Battery voltage is checked by placing the BATT switch to either ON or EMER with the generators offline. If the generators are online, the BATT switch is placed to EMER only to check battery voltage.

NOTE

The voltmeter will not register voltage with the BATT switch OFF. The circuit between the BATTERY BUS and the voltmeter is open to prevent draining the battery if the aircraft is parked for an extended period with the battery connected.

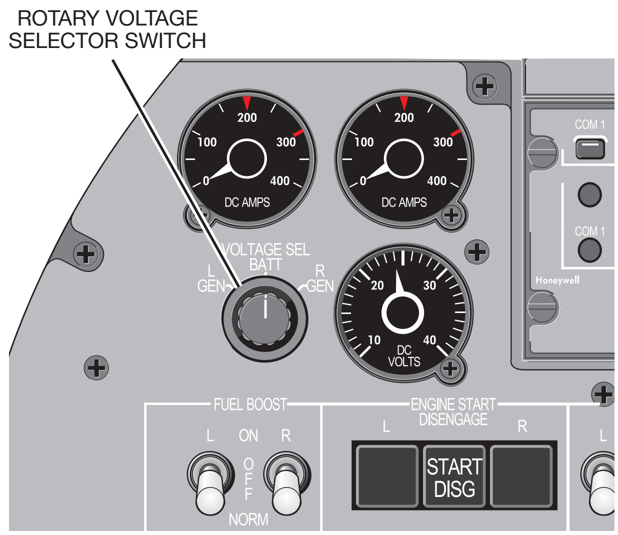


Figure 24-8. Voltmeter and Amperage Gauges

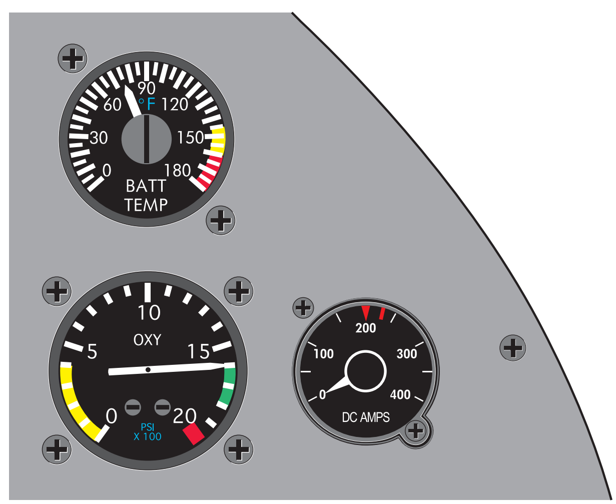


Figure 24-9. Battery Temperature Gauge

XLS+

The voltmeter is connected to the EMER bus. The BATT switchlight must be selected to BATT ON for the voltmeter to be active. The voltmeter displays system voltage when the NORM/EMER switchlight is selected to either NORM or EMER.

Battery voltage can be checked with the voltmeter, however the VOLTAGE SELECT switch must be in the BATT (spring-loaded) position and the battery isolated from the generators. The voltmeter (Figure 24-10) is connected to the BATTERY BUS with the BATT switchlight in the BATT ON or the NORM/EMER switchlight in the EMER position.

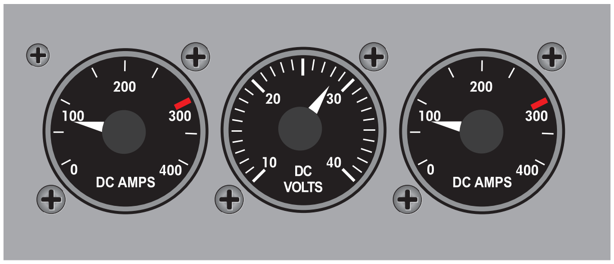


Figure 24-10. VOLTS and AMPS Gauges

Battery voltage is checked by placing the BATT switchlight to either BATT ON and the NORM/EMER switchlight in either NORM or EMER positions with the generators offline. If the generators are online, the NORM/EMER switchlight is placed to EMER only to check battery voltage. Battery temperature is checked by the BATT TEMP gauge (Figure 24-11).

NOTE

The voltmeter does not register voltage with the BATT switchlight in BATT OFF. The circuit between the BATTERY BUS and the voltmeter is open to prevent draining the battery if the aircraft is parked for an extended period with the battery connected.

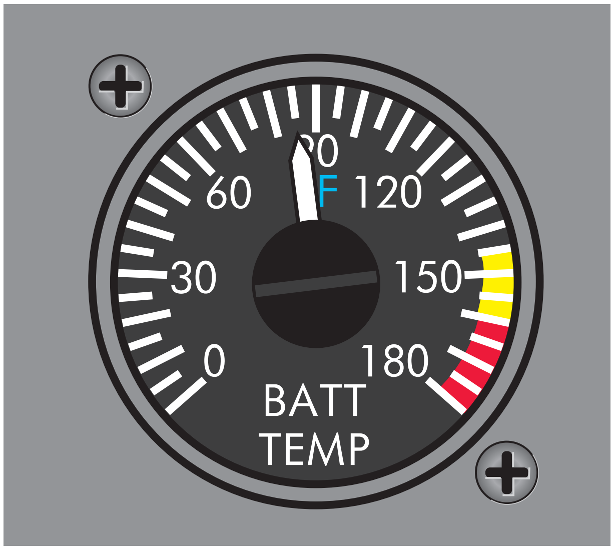


Figure 24-11. BATT TEMP Gauge

NOTES

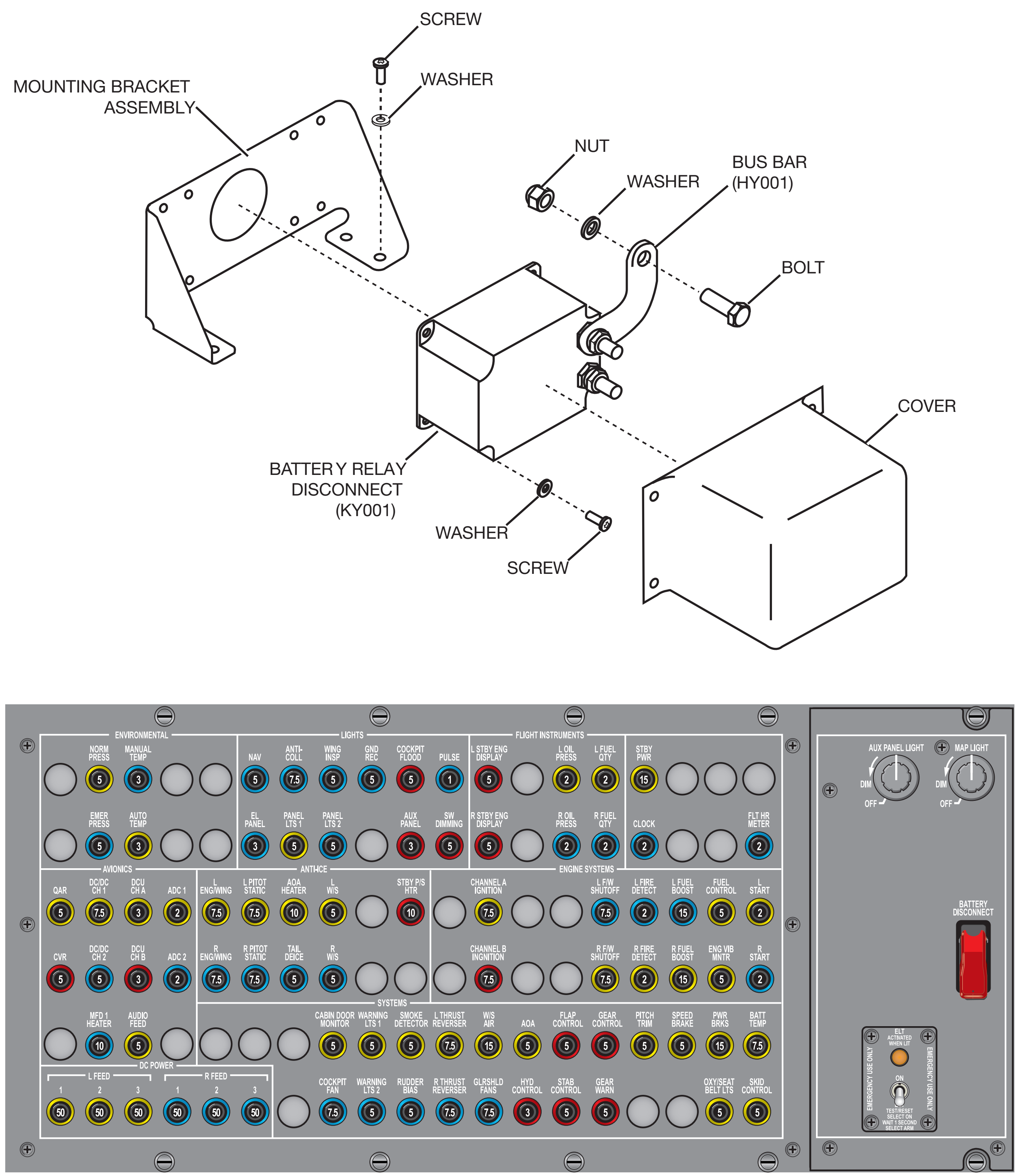


Figure 24-12. Battery Disconnect System

Battery Disconnect Switch

The battery disconnect switch is intended for use in the event of a battery overheat condition and is a cover-guarded switch on the left CB panel (Figure 24-12). Selecting disconnect with the switch energizes the battery disconnect relay.

Battery Disconnect Relay

The battery disconnect relay, when energized, opens the battery ground path to the airframe ground. The relay automatically opens during start when an EPU supplies power to the aircraft. A battery overheat condition is another case in which the battery disconnect relay is energized open. In this case the crew may use the battery disconnect switch to energize the battery disconnect relay. Once the airframe ground is removed, the battery can no longer receive a charge and it cools down. The battery disconnect relay is installed in the battery compartment behind the battery.

NOTE

The battery disconnect switch will operate only if the battery switch is ON (BATT position).

Interior Master Switch

A interior master switch located directly below the battery disconnect switch (XL/XLS) or on the electrical panel (XLS+) is used to secure all electrical power in the cabin (Figure 24-13). This switch is normally activated if an electrical fire should occur in the cabin. Activating the switch opens the interior master relay on the right feed bus, thereby, removing electrical power to the cabin area.

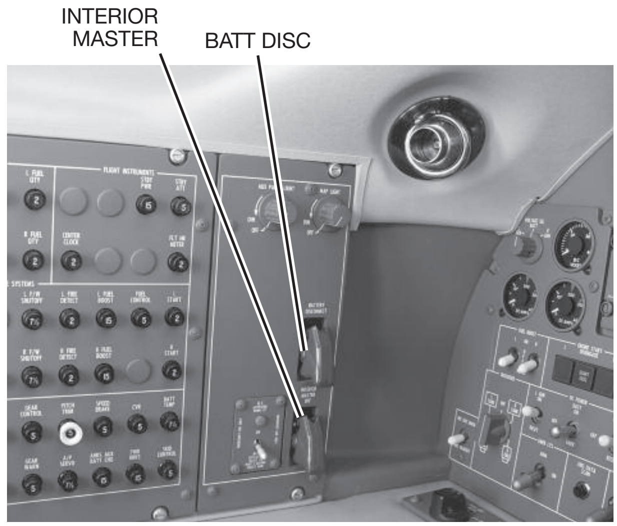


Figure 24-13. Battery Disconnect/Interior Master Switches

NOTES

EMERGENCY BATTERY POWER PACKS

SECONDARY FLIGHT DISPLAY (SFD) BATTERY PACK (XL)

A 28 volt, 2.5 amp-hour, sealed lead-acid battery pack is installed in the left nose compartment. The battery pack can provide approximately 30 minutes of power for emergency operation of the secondary flight display (SFD). The pack is normally charged by the aircraft main DC electrical system through the STBY PWR circuit breaker on the pilot CB panel. The standby SID battery pack can be checked for adequate charge during preflight by a STBY PWR ON–OFF–TEST switch located on the pilot lower switch panel (Figure 24-14).

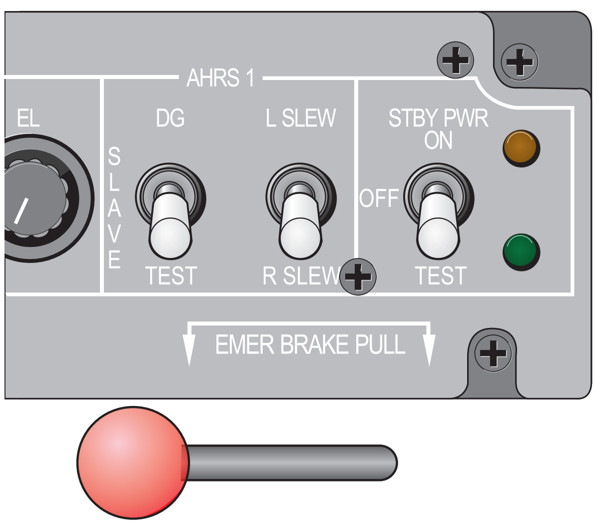


Figure 24-14. Standby Power Switch (XL/XLS)

STANDBY FLIGHT DISPLAY—SECURAPLANE BATTERY PACK (XLS/XLS+)

A 28 volt, 10.5 amp-hours, sealed XL-2410 lead-acid battery pack is in the left nose compartment. The battery pack provides approximately 3.5 hours of power for emergency operation of the standby flight display (SFD). The pack is normally charged by the aircraft main DC electrical system through the STBY PWR circuit breaker on the pilot CB panel. The standby SFD battery pack is checked for adequate charge during preflight by a STBY PWR switch on the pilot lower switch panel (Figure 24-15).

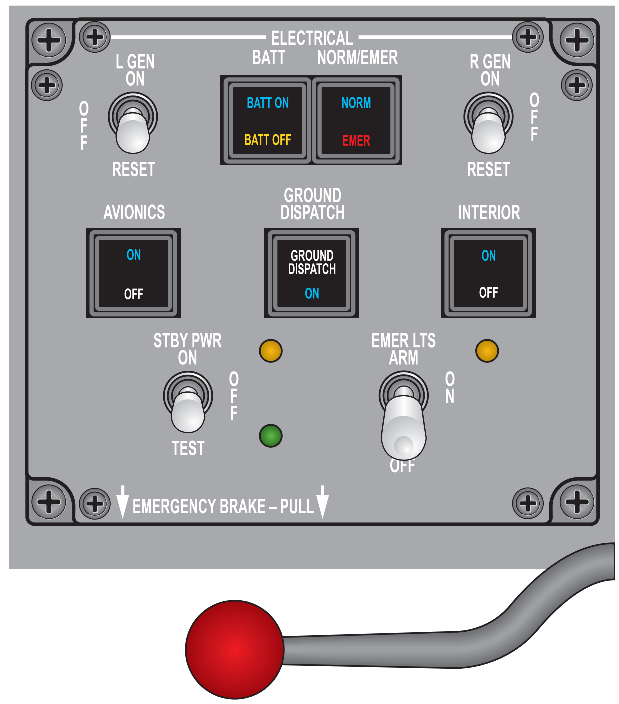


Figure 24-15. STBY PWR Switch (XLS+)

AHRS AUXILIARY BATTERY (XL/XLS)

A sealed lead-acid battery pack is installed in the right nose compartment. This pack is used as an emergency power supply for the attitude heading reference systems (AHRS) if power interruptions occur, provided the STBY PWR switch is ON.

A white AHRS AUX PWR L–R annunciator will illuminate if the emergency battery pack is supplying power directly to either or both AHRS systems. The pack is charged from the

main DC system through the AHRS 1/2 AUX circuit breakers located on the RH CB panel.

The battery pack is capable of providing approximately 30 minutes of operating power directly to both AHRS systems.

EMERGENCY LIGHTING BATTERY PACKS

There are two NiCad battery packs located in the cockpit/cabin area. They are used as a source of power for the emergency exit lights (interior and exterior). One pack is located in the cockpit and one located in the aft cabin. Refer to Chapter 33, "Lighting," for specific information on this system.

DIAGNOSTICS

Battery

The electrolyte in a nickel-cadmium battery is a solution of distilled water and potassium hydroxide. The electrolyte is used only as a conductor and does not react with the plates, like the electrolyte in a lead-acid battery. The state of battery charge cannot readily be determined by a specific gravity reading, since the electrolyte does not change appreciably. For this reason, it is not possible to determine the state of charge of a nickel-cadmium battery by checking the electrolyte with a hydrometer. Nor can the charge be determined by a voltage test due to the inherent characteristic that the voltage remains constant during 90 percent of the discharge cycle. However, a visual indication is beneficial because the plates are porous and absorb the electrolyte while discharging and expel the electrolyte while charging.

The negative plates in the battery are cadmium hydroxide, and the positive plates are nickel hydroxide. During charging, all oxygen is driven out of the negative plates and only metallic cadmium remains. The oxygen dispelled from negative plates is picked up by the positive plates to form nickel dioxide. Toward the end of the charging process, the electrolyte turns into a gas due to electrolysis that takes place in the electrolyte. A slight amount of gassing is necessary to completely charge the battery.

During discharge, the reverse chemical action takes place. The negative plates gradually gain back the oxygen, while the positive plates lose oxygen. Due to this interchange of oxygen, the chemical energy of the plates converts into electrical energy and the plates absorb the electrolyte. For this reason, the level of the electrolyte should be checked only when the battery is fully charged.

CAUTION

The slightest acid contamination deteriorates the nickel cadmium battery. When servicing battery, make certain that servicing equipment is acid free.

Nickel-Cadmium Battery Servicing

A new battery is shipped discharged and contains the proper amount of electrolyte. It does not require leveling even though the battery may appear to have insufficient electrolyte.

The electrolyte, a 30% by-weight solution of potassium hydroxide in distilled water, does not take an active part in the chemical reaction. It is used only to provide a path for the current flow. At 70°F (21.1 °C) the specific gravity of the solution should remain within the range of 1.24 to 1.30.

Another unusual characteristic of the nickel-cadmium battery is that when completely discharged, some cells reach zero potential and charge in the reverse polarity. This action adversely affects the battery, so that it does not retain a full capacity charge. As a result, it becomes equivalent to much lower-rated battery. To cure this problem, discharge the battery and short circuit each cell to obtain a cell, obtaining balance at zero potential. This process is known as equalization.

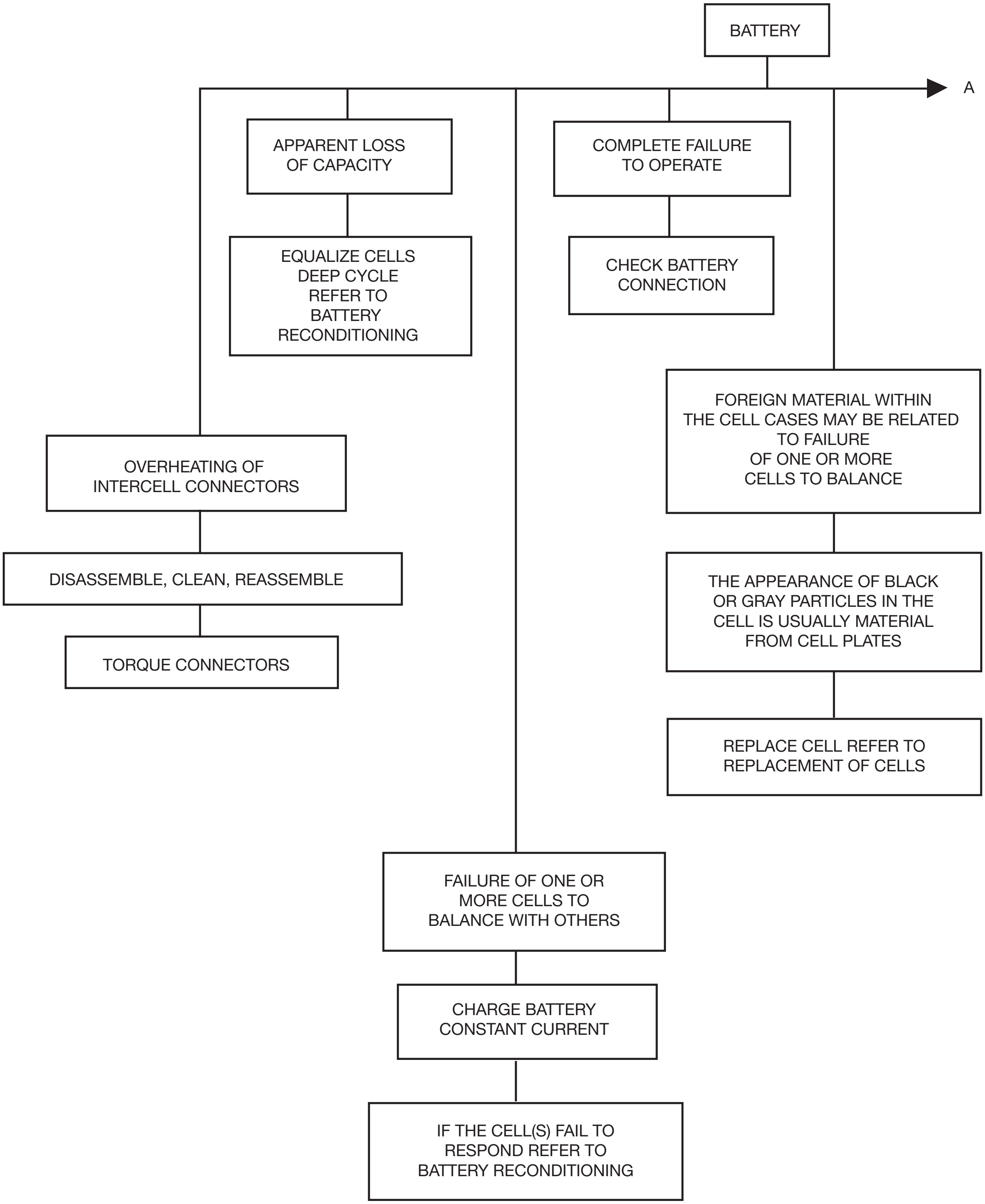


Figure 24-16. Battery Checks Diagram (Sheet 1 of 2)

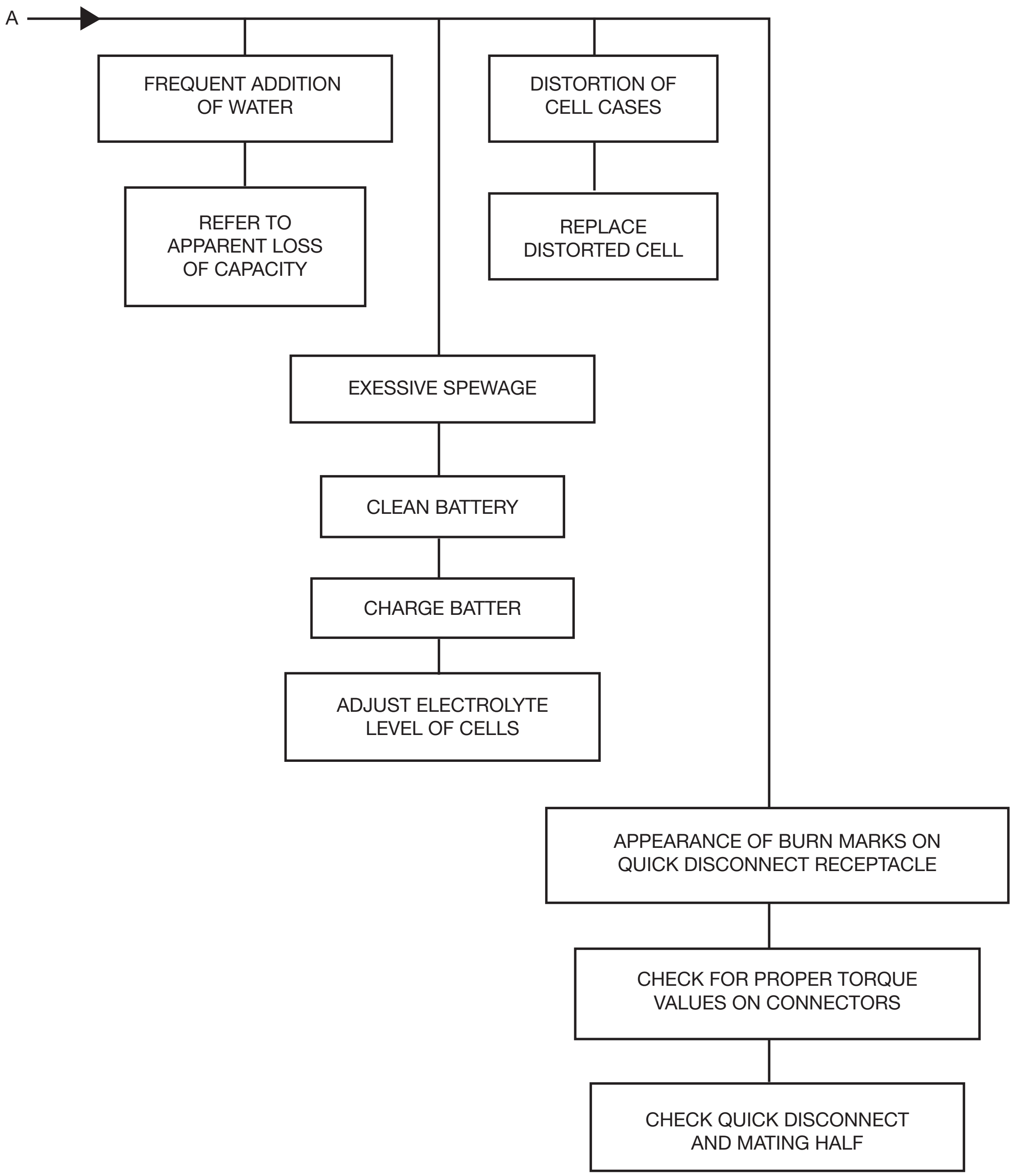


Figure 24-16. Battery Checks Diagram (Sheet 2 of 2)

WARNING

The electrolyte used in nickel-cadmium batteries is a caustic solution of potassium hydroxide. Serious burns result if it comes into contact with any part of the body. Use rubber gloves, rubber apron and protective goggles when handling this solution. If electrolyte gets on the skin, wash the affected areas thoroughly with water, neutralize with 3% acetic acid, vinegar or lemon juice. If electrolyte gets into the eyes, flush with water and get immediate medical attention.

The battery electrolyte is corrosive and should never be serviced inside the aircraft. The battery electrolyte has a high affinity for carbon. Any amount of electrolyte that is expelled reacts with carbon dioxide to form white crystals of potassium carbonate. This substance is noncorrosive, nontoxic, and nonirritating. It can be wiped away with a clean damp cloth.

Sealed Lead Acid Battery Servicing

The lead acid battery in the airplane is rated at 44 amp-hours and is maintenance free.

Storage

The lead acid battery used in the airplane is to be serviced and charged when the new battery is received and must be recharged when in storage (every 90 days).

New batteries should be placed in service from storage within 2 years of the manufacturing date.

Batteries not recharged every 90 days when in storage must be conditioned by charging at the test rate of 35.2 amp for one hour. Charging after conditioning must be at 28.2 ± 0.5 VDC, and 3.5 amp constant current, for approximately 18 hours or until the voltage reaches 30 volts and remains 30 volts for one hour.

Battery Checks

The battery should be recharged when its open-circuit voltage drops below 2.08 volts per cell or the open circuit voltage drops below 25.0 VDC (Figure 24-16).

CAUTION

Never deep cycle the lead acid battery. Whether in storage or in operation, do not allow the lead acid battery voltage to drop below 18 VDC. Even if subsequent recharging restores the battery voltage to an acceptable level (25 VDC minimum), the battery life cycle could be severely degraded. If the lead acid battery open circuit voltage is above 18 VDC but below 22 VDC, battery must be removed and serviced.

Reserve or Emergency Capacity Test

The following tests may be performed if the capacity of the battery is in question:

1. Make sure the battery is fully charged.
2. With the battery temperature above 59°F (15°C), discharge the battery at the rate of 35.2 amp for one hour.
3. Using a voltmeter, check open circuit voltage. Voltage must be 18 VDC or greater.
4. If the battery fails the voltage check, it is no longer considered serviceable and must be replaced.

Battery Charging

The battery must be charged using a constant potential or constant voltage charger regulated at 28.2 ± 0.5 VDC.

The battery must contain a reserve or emergency capacity. The airplane electrical system is capable of charging the battery by placing the battery switch to ON with generators operating or with external power applied, provided the battery voltage is above 22 VDC.

CAUTION

If the battery appears to be dead, do not attempt to charge using the airplane generators or external power.

Always make sure that the battery is disconnected during long periods of maintenance with external power applied.

NOTES

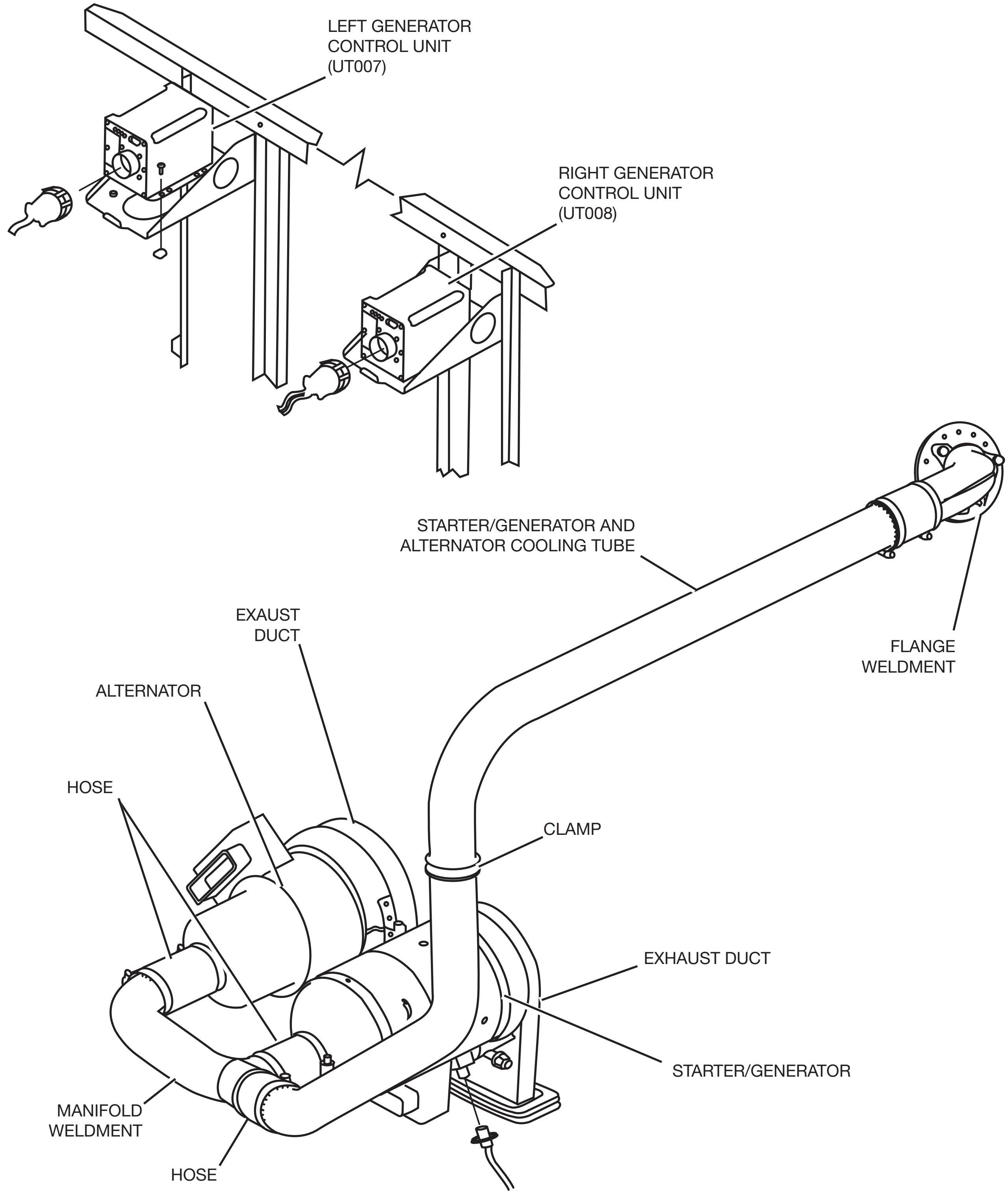


Figure 24-17. DC Generator System

DC POWER GENERATION

DC GENERATOR

Primary electrical power is obtained from two 28.5 VDC, 300-ampere continuous-rating engine-driven generators (starter-generator) (Figure 24-17). These generators are also used as motors for engine starting.

The starter-generator is on the forward center pad of the accessory gear box of each engine. Access to the starter-generator is gained by removing the lower engine cowling.

The DC generator system is the aircraft primary source of 28 volts direct current (VDC) electrical power. The DC generator system is divided into a split bus system: left and right. Each generator system is operated independently, but the distribution systems are in parallel except under fault condition. The generators share loads equally (± 30 amp) under normal operation via an equalizer connection between generator control units (GCUs).

Starter Limitations

Three engine starts in 30 minutes with a 90 second rest between starts.

NOTES

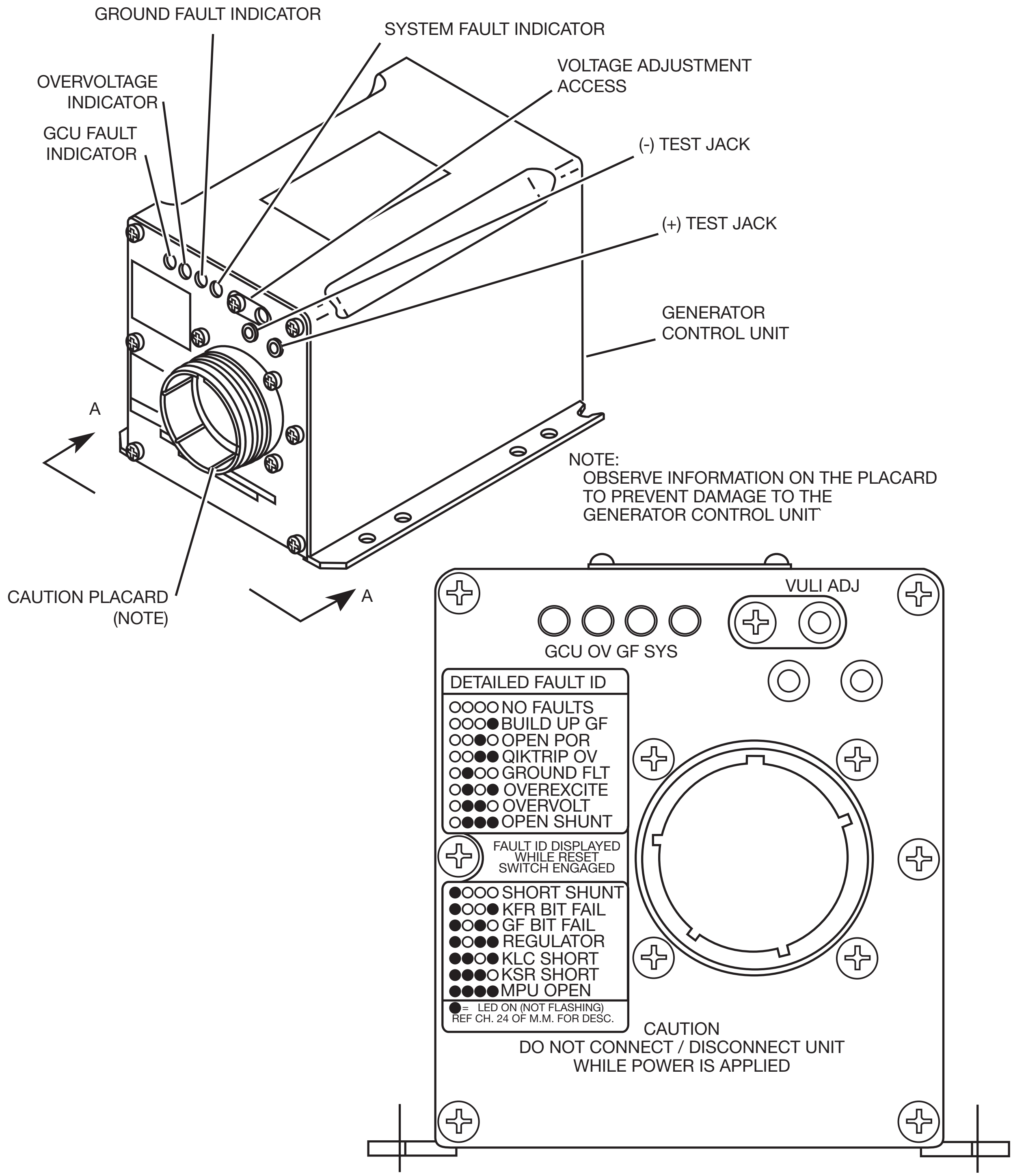


Figure 24-18. Generator Control Unit (GCU)

GENERATOR CONTROL UNIT

Description

There are two GCUs (UT007 left and UT008 right) in the DC generating system: one GCU for each starter-generator (Figure 24-18). They are on the left and right sides of the aircraft in the tail cone aft of FS 473.40. The GCU utilizes solid-state integrated circuits and amplifiers to provide lightweight controls.

The GCU includes voltage regulation with:

- Automatic high-accuracy load division
- Overvoltage monitor system
- Overexcitation protection
- Automatic line contactor control
- Reverse current protection
- Starter cut-off
- Field weakening
- Ground fault protection

NOTES

Table 24-3. INDICATIONS OF GCUs LED DISPLAY

GCU LED FAULT DISPLAY	GCU FAULT ID DISPLAYED	GCU FAULT INDICATIONS
OOOO	NO FAULTS	No faults system functioning properly
OOOX	BUILD UP GF	Build up ground fault - generator current >200A while power relay (KLC) was off.
OOXO	OPEN POR	POR (pin B) sensing wire open
OOXX	QIKTRIP OV	Quick Overvoltage trip (GCU failure)
OXOO	GROUND FAULT	Ground Fault according to CT signal - pin M to m.
OXOX	OVEREXCITE	Overexcitation—check pins W and p.
OXXO	OVERVOLT	Overvolt detected.
OXXX	OPEN SHUNT	Open shunt engine start—check connections to generator—pins (AA, DD, y, B).
XOOO	SHORT SHUNT	Field current exceeded 20A on pin AA.
XOOX	KFR BIT FAIL	Failed KFR (GCU failure) field relay inside GCU.
XOXO	GF BIT FAIL	Failed GF test (GCU failure)
XOXX	REGULATOR	Regulator failed to energize during engine start (GCU failure).
XXOX	KLC SHORT	Line contactor driver (pin L to J) overloaded or failed to turn ON.
XXXO	KSR SHORT	Start contactor driver (pin L or J) overloaded or failed to turn ON.
XXXX	MPU OPEN	No speed pickup (pin X to Y) signal.

The GCU is controlled by a microprocessor and utilizes BIT and nonvolatile memory for fault detection and isolation, during start-up and in the running mode. The GCU has the capability of recording and displaying a no-fault code or 14 possible fault codes (Table 24-3) when the generator switch is pressed and held in the reset position. The four LEDs on the front of the GCU are used to indicate faults.

GCU FUNCTIONS

Voltage Regulator

By using an integrated circuit comparator amplifier with a regulated reference voltage, any difference between the reference voltage and the generator output (as seen by the sensing line) is amplified. It is then supplied to the comparator circuit which controls the shunt field excitation of the generator. Additional safety is built-in, preventing generator build-up with an open-field relay, until the pilot has placed the generator control switch in the RESET position. The field relay is automatically reset and the reset circuit is isolated so that cycling does not take place in the event that the system is reset into a fault.

Generator Paralleling

The control utilizes an integrated circuit through which the difference between the interpole voltage of the generator and the equalizer bus is amplified, inverted, and filtered. The resulting difference voltage is then coupled to the summing function of the overvoltage circuit. When this voltage change is fed to the regulator it causes a shift in the regulator output.

The equalizing circuit is always trying to sum the difference to zero: between the voltage across the interpole of the local generator and the equalizer bus to zero. The equalizer relay circuit works in conjunction with the control relay. Whenever the control relay is deenergized, the equalizing circuit is also disconnected, resulting in complete isolation of the tripped generator.

NOTE

The generators do not come online if the voltage is 0.3 volts below the bus voltage. Parallel the generators if this condition is noted.

Generator Protection

Differential Voltage/Reverse Current

Before a generator is connected to the load bus, differential voltage sensing allows closure to occur, only if that generator is within 3/10 volt of the load bus voltage. It does not necessarily have to be above the load bus to allow the power relay to close. After the generator has been connected to the bus, reverse current sensing automatically takes place. The same circuit which evaluated the differential voltage is now automatically converted to reverse current sensing.

NOTES

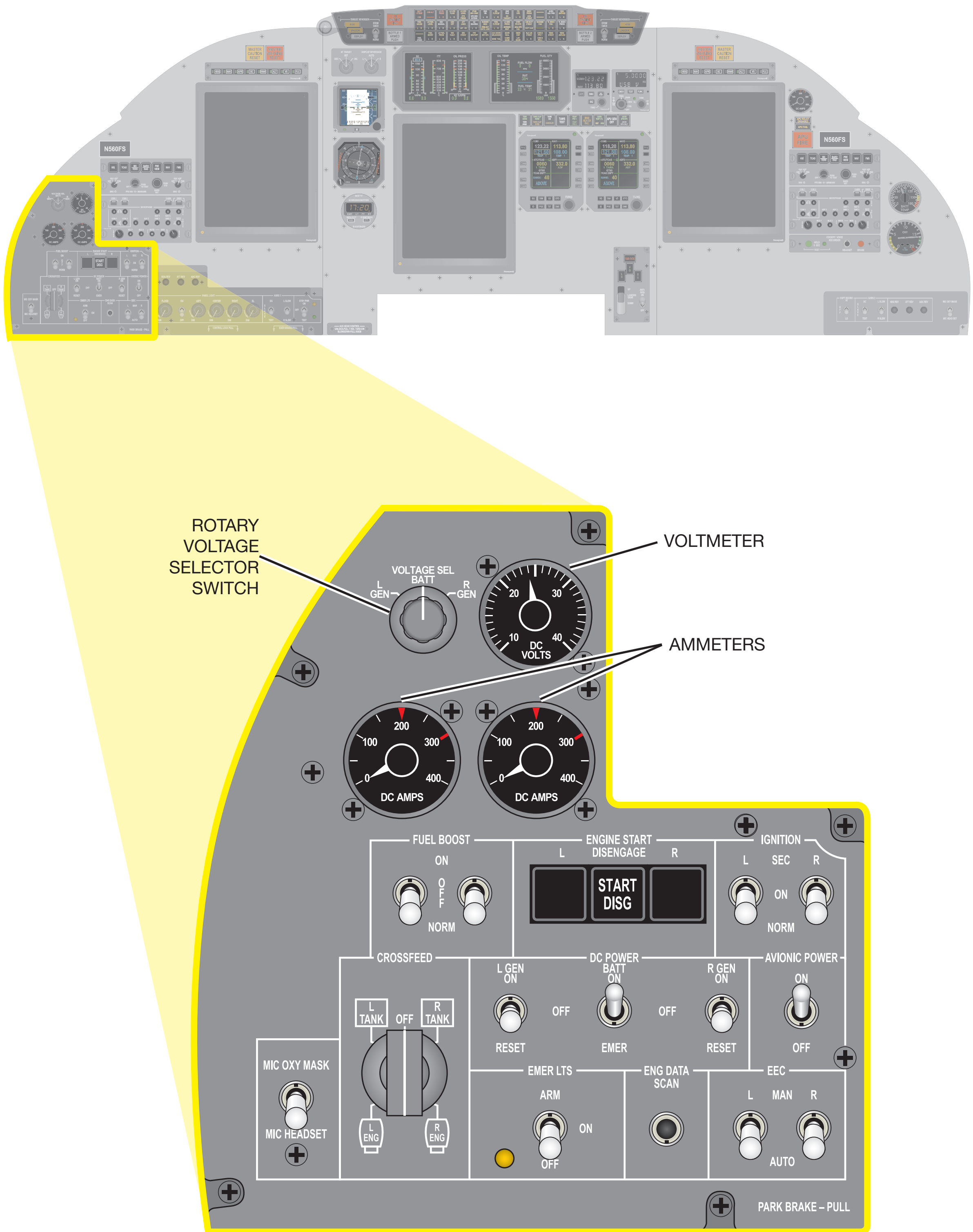


Figure 24-19. DC Power Switches and Annunciators (XL/XLS)

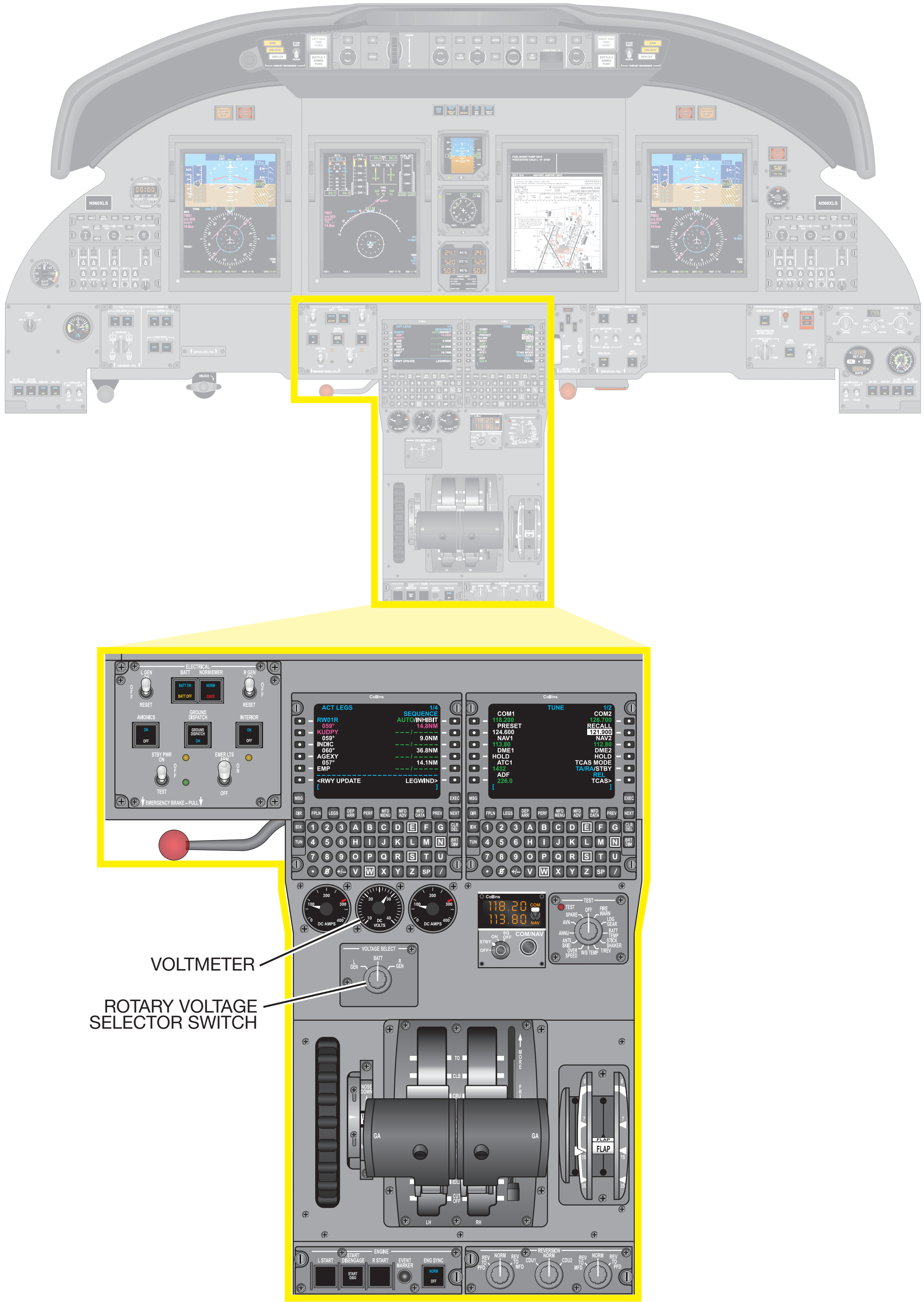


Figure 24-20. DC Power Switches and Annunciators (XLS+)

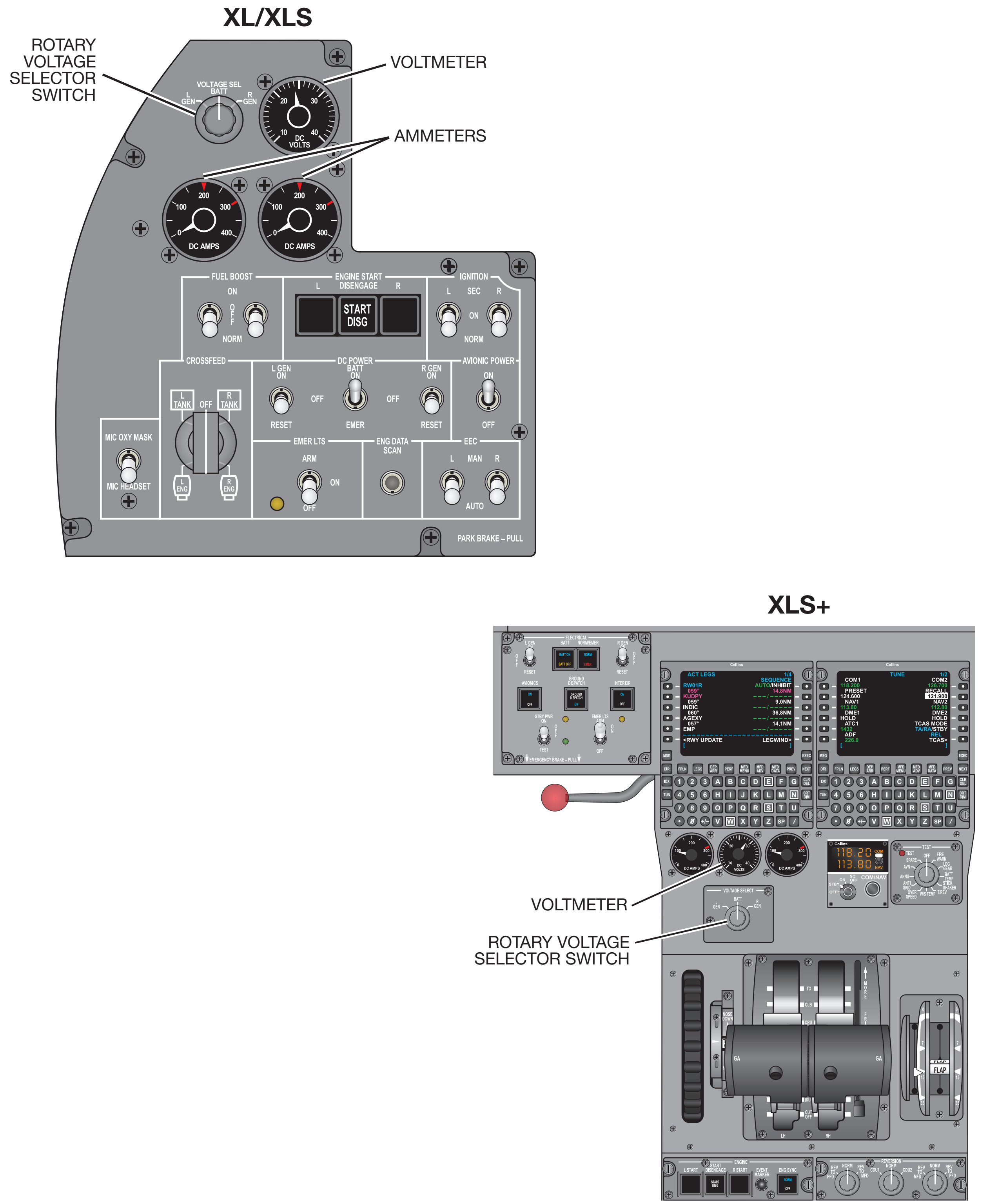


Figure 24-21. DC Power Switches and Annunciators (XL/XLS and XLS+)

If a generator is unable to maintain voltage and draw reverse current (becoming a load on the remaining generator) it is removed from the line (when 10% or more of its load rating is present in the interpole winding). Once the generator has been dropped from the bus due to reverse current, the control does not permit the generator to come back online until the generator's output voltage reaches a proper level, ensuring forward current to the bus.

Field Weakening Circuit

The control incorporates an optimum field-weakening feature. This circuit goes into a current regulation mode during engine startup. That is, it senses the current in the interpole windings of the starter-generator, which in turn controls the field-weakening circuitry in order to hold a fixed amount of current in the generator windings (by virtue of controlling shunt field excitation). Until a certain value current is reached, a full field-condition exists. When the current drops below this value, regulation continues until the start circuit is deenergized. This occurs at the starter cut-off speed point. During the engine start mode, all other protection functions of the control panel are disabled, eliminating any possibility of nuisance trips.

CONTROL SWITCHES AND INDICATOR LIGHTS

The battery switch (SI022) has three positions marked ON-OFF-EMER. With the BATT switch in the ON position, the battery (or external power) is connected to the battery bus and emergency bus. In the OFF position, the battery (or external power) is isolated from all loads except those on the hot battery bus. The EMER position on the battery switch connects the battery (or external power) to the emergency bus. Since the battery relay is disengaged, only systems receiving DC power from the hot battery bus and emergency bus are active.

NOTE

If the generator is online, but the battery switch is set to OFF or EMER, the battery is not being charged. If in the EMER mode, the battery does, in fact, discharge.

The left and right generator switches have three positions marked GEN, OFF and RESET. With the switch placed in the GEN position, the generator control for the regulation, protection and load bus connection is an automatic function. The generator is connected to its load bus when the correct voltage output and generator speed has been obtained.

The generator de-excites and is disconnected from the bus as a result of:

- An overvoltage
- Feeder fault or
- Engine fire switch actuation

Placing the switch to the OFF position also isolates the generator from its respective load bus without de-exciting the generator. The switch RESET position is momentary and provides a means of resetting a generator that has tripped as a result of an overvoltage, feeder fault or engine fire switch actuation. RESET is sometimes necessary following a windmilling airstart of an engine.

Two ammeters (EI011 left and EI010 right) installed on the left switch panel, display a visual indication of the load current supplied by the respective starter-generator. The two ammeters are identical and have a red triangle at 200 amp to indicate an on-ground continuous max amperage load per starter-generator. The ammeters also have a red line at 300 amp for an in flight max continuous rating

The voltmeter (EI009) installed on the left switch panel indicates the voltage supplied by the power source. The voltmeter has a scale range from 10 to 40 volts.

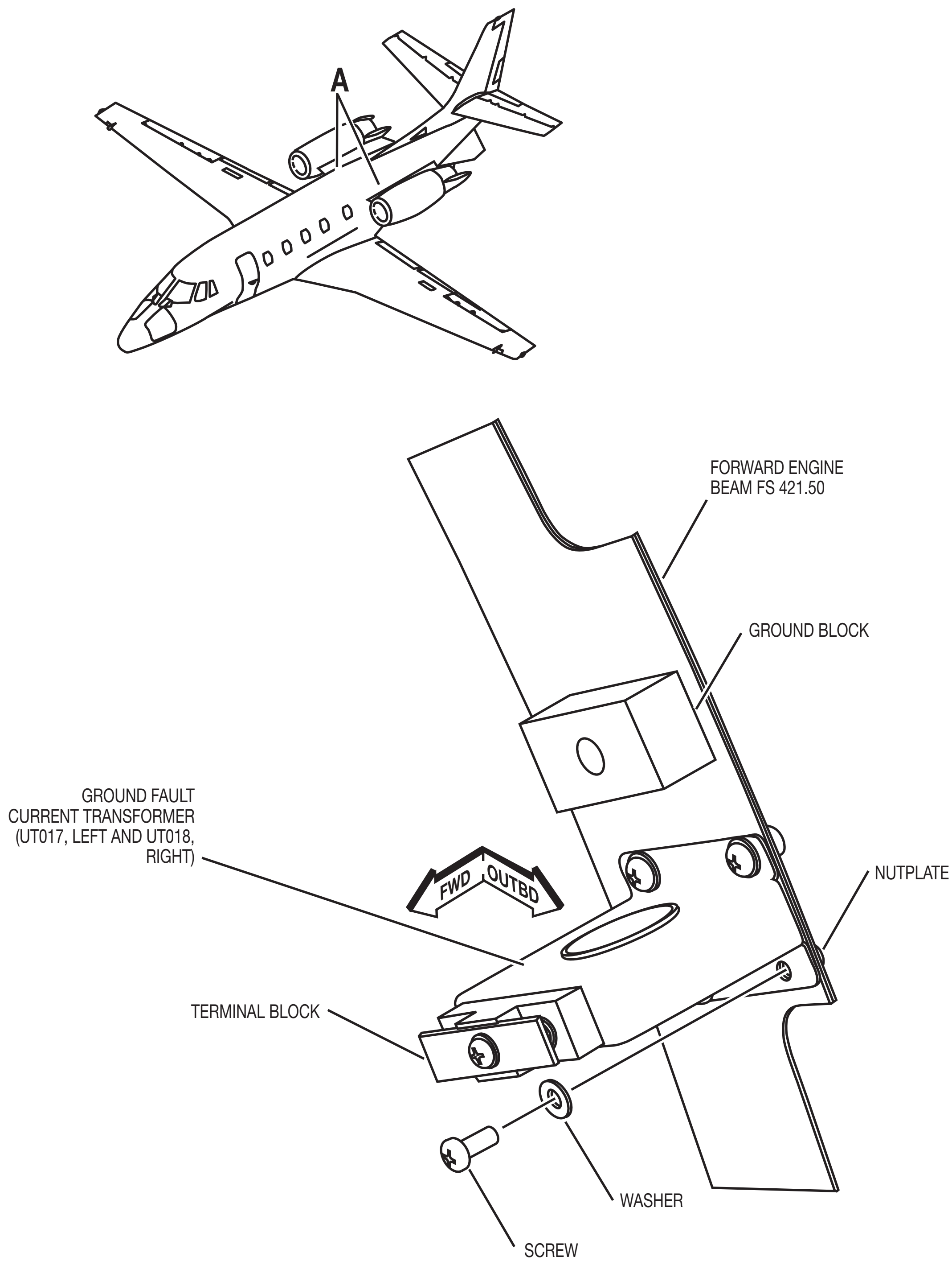
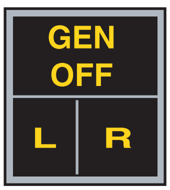


Figure 24-22. Current Transformer (CT)

The rotary voltage selector switch (SI023) is used to transfer the voltmeter to the desired DC voltage supply source. The VOLTAGE SEL switch has three marked positions: L GEN, BATT and R GEN.

The amber L–R GEN OFF annunciator illuminates when the respective power relay is open (Figure 24-23). This isolates the generator from its respective load bus. As each power relay opens, the master caution switchlights also illuminate. Should both left and right GEN OFF annunciators illuminate at the same time, the MASTER WARNING RESET switchlight also illuminates.



L/R GEN OFF
Annunciator flashes to indicate the respective generator relay is open and the generator is off line. Activates MASTER CAUTION lights. Both GEN OFF L/R flashing simultaneously will activate both MASTER WARNING and MASTER CAUTION lights.

XL/XLS ANNUNCATOR

DC GENERATOR OFF L-R			
Color	**Inhibited By**		**Debounce**
Red	LOPI	TOPI	Standard
Amber	*ESDI	SIPI	*1.0 Seconds
This message is displayed when the respective generator contactor is open. Refer to red EICAS message for details.			

XLS+ CAS MESSAGES

Figure 24-23. Generator Off Indications

Ground Fault Running

After initial closing of the power relay, ground fault sensing is accomplished by use of two current transformers (CT) (Figure 24-22). One is placed at the negative terminal of the generator, and the other is placed as far down the positive feeder cable as possible, before passing the power relay. The ground fault current transformers (UT017, left and UT018) are in the tail cone baggage compartment on the forward engine beam at FS 421.50. The area between these current transformers is known as the protected zone. During all normal load transients, a single pulse is induced into each CT. The relation between the CTs is such that the pulse induced is oppositely polarized and thus opposing. If one of the CTs is bypassed due to the presence of a ground fault, the opposite CT provides a net pulse current through the control panel, actuating the ground fault circuitry and causing a trip of the field relay.

Ground Fault Build-up

As each generator becomes initially excited after the start relay drops out, sensing for feeder-to-ground short (known as ground fault) begins. If a load is carried by the generator equal to or greater than one-half of its rating (before the power relay is initially closed) this is the basis for a ground fault build-up trip-off. The control does not allow a generator to become continuously excited or to close the control relay circuits until the relay has been tripped.

NOTES

Overvoltage/Overexcitation

If an overvoltage situation occurs due to failure of the regulator, an inverse time-curve is generated which causes the overvoltage integrator, to trip the field relay, after a predetermined time. When a generator has been paralleled with other components in the system, overvoltage quite often does not occur due to loading by the remainder of the system. A faulted regulator, however, causes the corresponding generator to attempt to carry more than its share of the load.

The paralleling circuitry within each control evaluates each system's load sharing with respect to the equalizer bus. When a given system fails and attempts to carry more than its share of the load, this alone causes a de-excitation signal to be fed to the system's respective regulator. All other systems cause an excitation signal to occur. The faulted system, being unable to comply with this de-excitation command, does not do so; and a second signal, fed from the paralleling circuit to a special summing point at the overvoltage integrator, trips off this system. All other systems remain active.

Starter Cut-Off

The starter cut-off circuit works from the sensing of a variable-frequency input that is supplied from a monopole internally mounted on the generator. This circuit automatically switches off the start mode. If the generator switch is in the GEN position with the starter switches off, the generator builds up and generates.

Resetting a Dead Bus

The control panel provides a reset feature, which allows the resetting of a field relay from a dead bus with no external power required. If a local system needs to be reset while all other systems remain inoperative, it is necessary only to place the generator switch into the RESET position. If that generator is capable of operation, build-up occurs, allowing the field relay to close and the system to come up to voltage in a normal manner.

DC GENERATOR SYSTEM TROUBLESHOOTING

LEDs

The GCU provides for self diagnostic analysis using internal circuitry in conjunction with four case mounted LEDs. The LED fault indicators indicate why the GCU is or is not working. The four LEDs are labeled as follows:

- GCU LED—Indicates an internal fault.

  If the OV LED is also flashing, the field transistor has shorted. Check the wire on pin DD of the GCU. This is the return for the field suppression diode.

  There is a general GCU internal failure.

  If the LED is on steady, the internal microprocessor has shut itself down and the GCU should be replaced.

- OV (Overvoltage) LED—The GCU has detected an overvoltage condition and has shut itself down.

  If the SYS LED is flashing also, there is a generator overexcitation condition. There may be a short to power on the field wire (pin AA).

  If the GCU LED is also flashing, the field transistor has shorted. Check the wire on pin DD of the GCU. This is the return for the field suppression diode.

- GF (Ground Fault) LED—The GCU has detected a ground fault condition and shut itself down.

  Check the ground fault transformer wiring and the transformer orientation.

  Check the generator power feeders for an actual fault to ground.

  If the start relay is slow to disengage after the start cycle terminates, there is a build-up ground fault trip.

- SYS (System) LED—The GCU has detected a problem with the aircraft wiring and shut itself down after detecting one of the following problems:

  Start relay wire is shorted. Pin J or L has drawn excessive current greater than 8.0 amps.

  Power relay wire is shorted. Pin P or L has drawn excessive current greater than 8.0 amps.

  Generator field is shorted. The field transistor output (pin AA) has drawn excessive current greater than 20.0 amps. Pin AA should measure 2.2 ohms to ground.

  The GCU has sensed a problem with the generator field wiring.

  There are 14 fault indications associated with the GCU LEDs. Refer to the placard on the GCU face for detailed fault identification. Press and hold the generator ON/OFF/RESET switch for detailed information of the fault.

CAUTION

Disconnecting or connecting the generator control unit electrical connector with electrical power applied (or when the generator is rotating) damages the generator control unit. Interchanging generator control units during maintenance is not recommended. A wire fault in one channel could also damage the other control unit.

DC Generator System Operational Test

1. Start the engines.
2. Accelerate the engines to 60 percent turbine speed. Synchronize the engines.
3. Position the L GEN switch (SI019) and R GEN switch (SI020) to RESET and then to OFF. The GEN OFF (L and R) annunciators should illuminate.
4. Position the R GEN switch (SI020) and L GEN switch (SI019) to ON.

NOTE

The total electrical load should be divided ±30 amp as indicated by the L and R AMMETERS (EI011 left and EI010 right). If the load unbalance exceeds 30 amp, adjust the GCUs.

5. Position the R GEN switch (SI020) and L GEN switch (SI019) to OFF.
6 Position L GEN switch (SI019) to ON, then the R GEN switch (SI020) to ON. Make sure that both GEN OFF (L and R) annunciators extinguish.
7. Position L GEN switch (SI019) and R GEN switch (SI020) to OFF, and make sure that both GEN OFF (L and R) annunciators illuminate and that the MASTER WARNING switchlight flashes.
8. Perform steps (6) and (7) in the opposite sequence.
9. Once the DC generator system test procedure is complete, shut down the engines.

DC Generator System Adjustment/Test

1. Before performing GCU adjustment, verify that the starter-generator system maintenance has been performed and that the engines are in operable condition.

NOTE

Adjusting the GCU requires two maintenance persons: one to operate the engines and the other to perform the adjustment.

2. Start the engines. If prior maintenance involved correcting a GCU, do not use the generator assist start.
3. Accelerate the left engine to 60% N_2.
4. Set the L-GEN switch (SI019) to RESET and then to OFF position. The L GEN OFF annunciator should illuminate.
5. Rotate the VOLTAGE SEL switch (SI023) to the L GEN position. The VOLTMETER (EI009) should read approximately 28.5 volts direct current (VDC). The meter indication is monitoring the circuit which regulates the left generator (MD001). This is not bus voltage indication.

CAUTION

Use a nonmetallic screwdriver when adjusting the GCU.

6. Connect a precision voltmeter to the test jacks on the left GCU (UT007). Adjust the externally accessible potentiometer on the GCU unit with a screwdriver until the precision voltmeter reads 28.5 VDC, ±0.1 VDC.

NOTE

Record voltage and left engine turbine speed.

7. Return left engine to idle.
8. Accelerate the right engine to 60% N_2.
9. Set the R GEN switch (SI020) to RESET and then to OFF position. The R GEN OFF annunciator should illuminate.
10. Rotate the VOLTAGE SEL switch (SI023) to the R GEN position. The VOLTMETER (EI009) read approximately 28.5 VDC. The meter indication monitors the circuit that regulates the right generator (ME001). This is not bus voltage indication.
11. Connect a precision voltmeter to the test jacks on the right GCU. Adjust the externally accessible potentiometer on the GCU to match the recorded voltage while adjusting the left GCU. Also, verify that the right engine turbine speed matches the left engine turbine speed.

NOTE

By matching the potentiometer voltage settings and engine turbine speed, the electrical load is distributed between the two generator systems.

12. Position the L GEN switch (SI019) to ON.

The GEN OFF L annunciator should extinguish.

The L AMMETER (EI011) shall indicate the generator load current.

The battery voltage as read on the VOLTMETER (EI009) shall be 28.5 ± 0.8 VDC.

NOTE

With the left generator (MD001) on the line, the battery voltage indication on the voltmeter (EI009) is equal to/or slightly less than the voltage indication on the voltmeter on the left generator.

13. Position the L GEN switch (SI019) to OFF.

14. Position the R GEN switch (SI020) to ON.

The GEN OFF R annunciator shall extinguish.

The R AMMETER (EI010) should indicate the generator load current.

The battery voltage as read on the VOLTMETER (EI009) shall be 28.5 ± 0.8 VDC.

NOTE

With the right generator (ME001) on the line, the battery voltage indication on the voltmeter (EI009) is equal to/or slightly less than the voltage indication on the voltmeter of the right generator.

15. Establish an aircraft electrical load.

NOTE

Do not operate the engine anti-ice system or windshield heat.

16. Position the L GEN (SI019) switch to ON.

The total electrical load should be divided ±30 amp as indicated by the L and R AMMETERS (EI011 left and EI010 right).

17. Position L GEN switch (SI019) and R GEN switch (SI020) to OFF.

18. Position L GEN switch (SI019) to ON, then the R GEN switch (SI019) ON. Verify that both GEN OFF (L and R) annunciators extinguish.

19. The generator system adjustment procedure is complete. The adjustment procedures also include generator test procedures. Shut down the engines and remove any test equipment.

NOTES

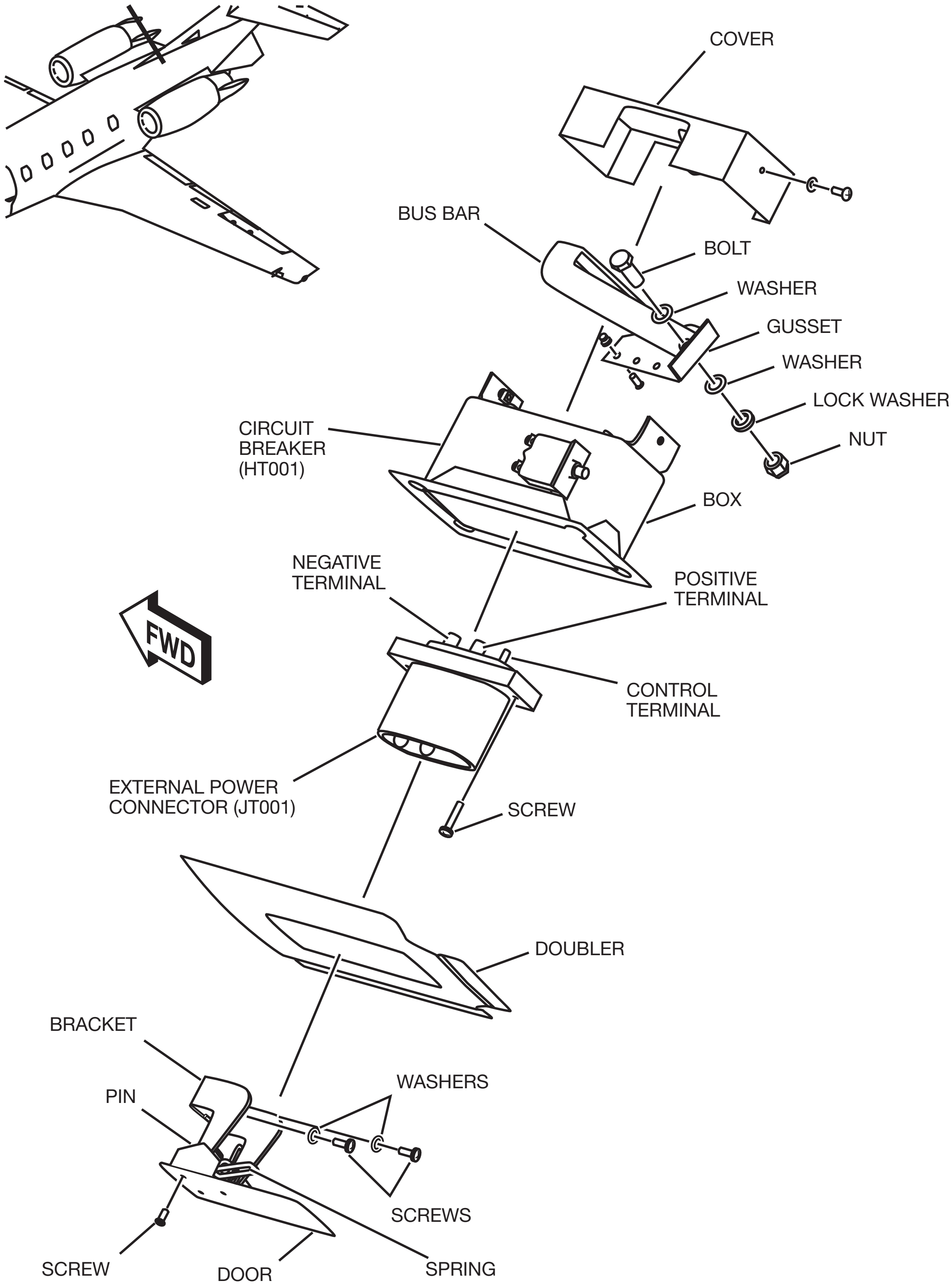


Figure 24-24. External Power

EXTERNAL POWER SYSTEM

DESCRIPTION

The DC external power system consists of: an external power connector (JT001) on the left side of the aircraft at FS 491.50; and an external power contactor (KZ007) in the aft power junction box at FS 473.46. The external power system components provide a means of connecting 28 VDC external power to the aircraft electrical system (provided the battery switch) (SI022) is in the ON position (Figure 24-24).

The external power connector (JT001) is a three-pin receptacle housed in a plastic material. The positive and negative pins are permanently marked on the front and rear of the receptacle.

The external power contactor (KZ007) is a single-pole, single-throw relay, utilized to connect the 28 VDC external power source to the hot battery bus.

OPERATION

Connecting the 28 VDC external power source energizes the external power contactor, connecting the external power source to the hot battery bus. Placing the battery switch (SI022) to ON position energizes the battery relay (KY001) and allows the 28 VDC external power to be connected to:

- The battery bus
- Emergency bus
- Left and right main DC buses

CAUTION

> Limit external power unit output to a maximum of 1,000 amp when connected to the aircraft. Adjust power unit output to 28.5 volts maximum with no load. Minimum required external power unit output is 800 amps.

The external power contactor (KZ007) is deenergized to remove external power from the battery bus when either generator is supplying power to the main bus. This is to prevent the aircraft electrical system from supplying electrical power to the external source.

The external power contactor is also deenergized if the overvoltage protection system senses that the external power source is greater than 32.6 VDC.

Some ground power units do not have reverse current protection. If the unit is turned off while connected to the aircraft, rapid discharge and damage to the battery can result. Always disconnect the ground power unit from the aircraft when the ground power unit is turned off.

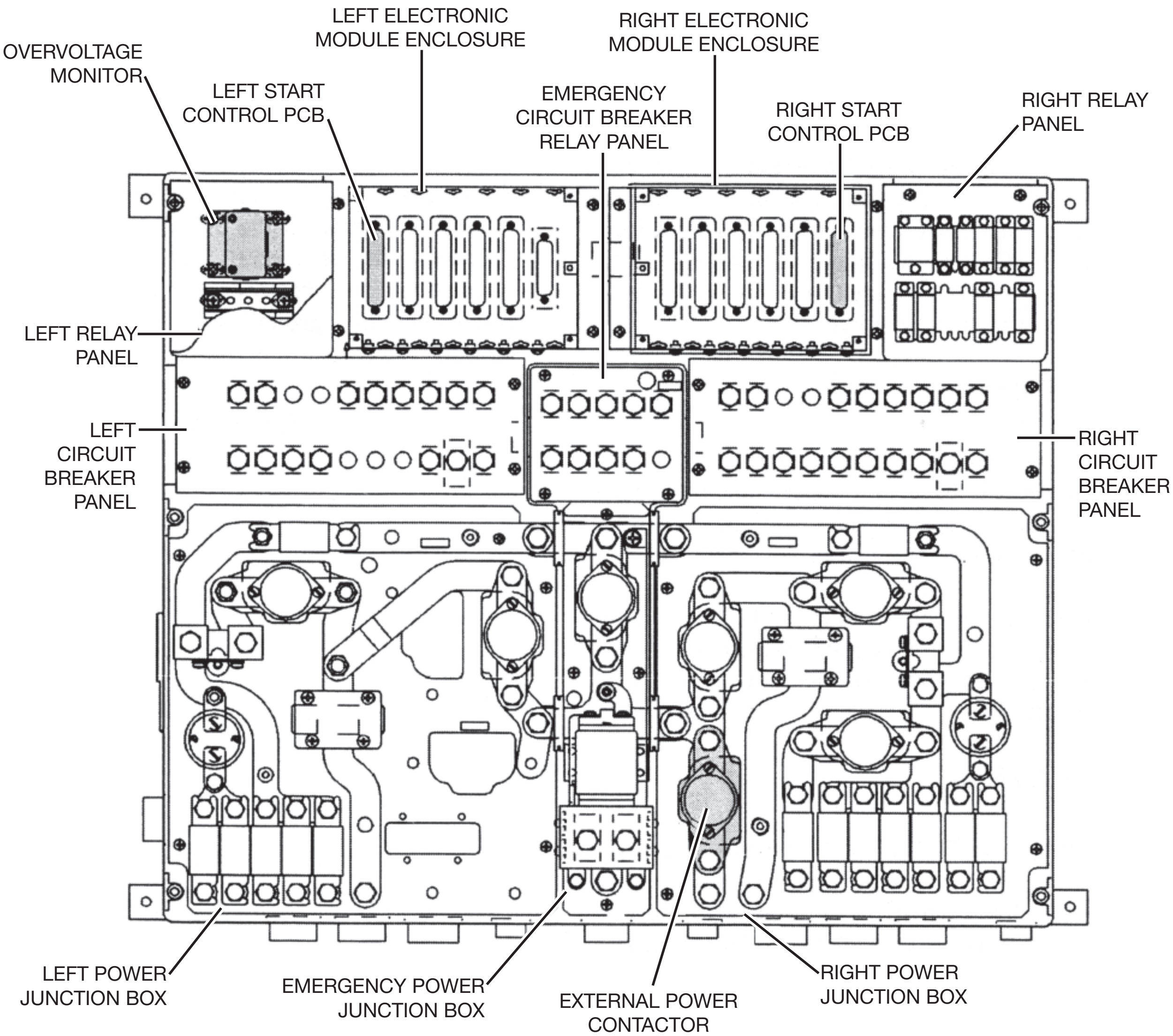


Figure 24-25. External Power Overvoltage Protection Components

CONTROLS AND INDICATIONS

Overvoltage Protection

Overvoltage protection is provided to ensure that excessive external power, applied to the aircraft, does not damage the wiring or systems (Figure 24-25). The overvoltage protection disconnects external power from the aircraft electrical system when input voltage exceeds 32.6 VDC. If the external power unit is disconnected, it must be cycled off and on (to reestablish power to the aircraft).

The system consists of:

- Battery disconnect relay
- Overvoltage monitor
- External power contactor
- Overvoltage protection circuit

The overvoltage protection circuits consists of a left and right start logic control modules printed circuit boards (PCB).

The battery disconnect relay (KY001) is in the battery compartment behind the battery between FS 405.50 and FS 424.50. The battery disconnect relay is utilized to disconnect the battery from the aircraft electrical system during engine start when an external power unit is supplying electrical power to the aircraft. The overvoltage monitor (UZ003) is in the upper left corner of the aft left power junction box behind the left relay panel. It monitors voltage and activates the left or right overvoltage protection circuit when voltage exceeds 32.6 VDC.

The overvoltage protection circuit is provided in the left and right power junction boxes. The left and right start logic control modules (NZ013 left and NZ012 right) are in the left and right enclosure electronic modules. The left and right enclosure electronic modules are in the upper center section of the aft power junction box. The left or right start logic control module (overvoltage protection circuit) is activated by the overvoltage monitor and is electrically connected to the external power contactor.

NOTES

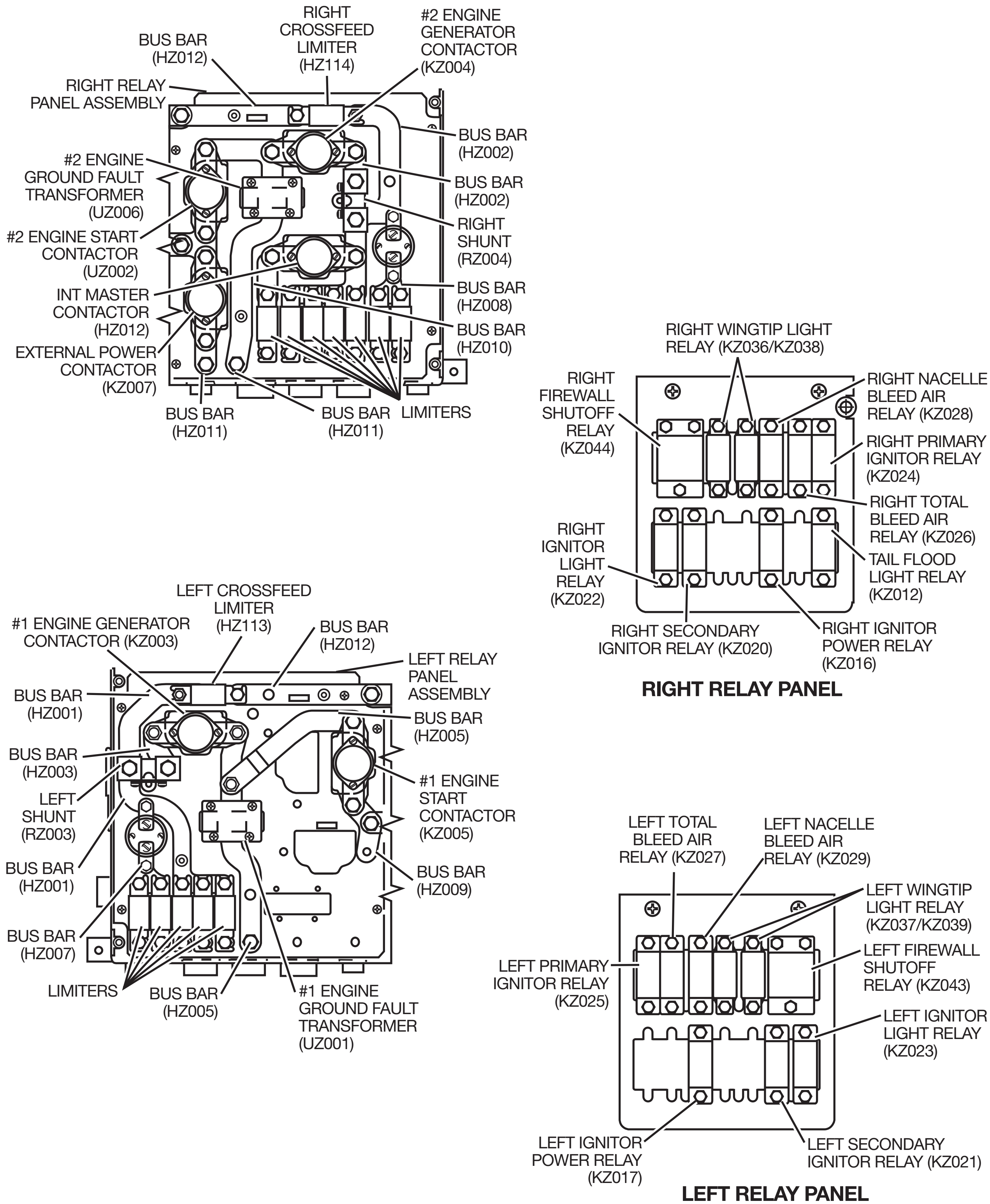


Figure 24-26. Left and Right Power Junction Boxes

DC POWER DISTRIBUTION

DESCRIPTION

DC electrical power is distributed through a system of buses and relays in the left and right power junction boxes, through current limiting fuses, to the main CB panels (Figure 24-26).

The main DC power junction box is aft of the tail cone baggage compartment at FS 473.46 and centerline of aircraft. The power junction box is a single box divided into left and right sides (separated by an emergency junction box). A single cover closes the junction box.

The power J-boxes contain:

- Relays
- Current transformers
- Circuit breakers
- Fuse limiters
- Junction blocks
- Printed circuit boards
- Shunts
- Terminal boards

CB panels in the cockpit provide 28 VDC distribution to various systems. All circuit breakers required for safe flight are on these CB panels.

Feeder cables are routed independently of the main aircraft wire bundles from each side of the power junction box to the respective circuit breaker panel. The feeder cables are protected at both ends: in the junction box by individual 60-ampere fuse limiters and in the CB panel by 50-ampere circuit breakers.

NOTES

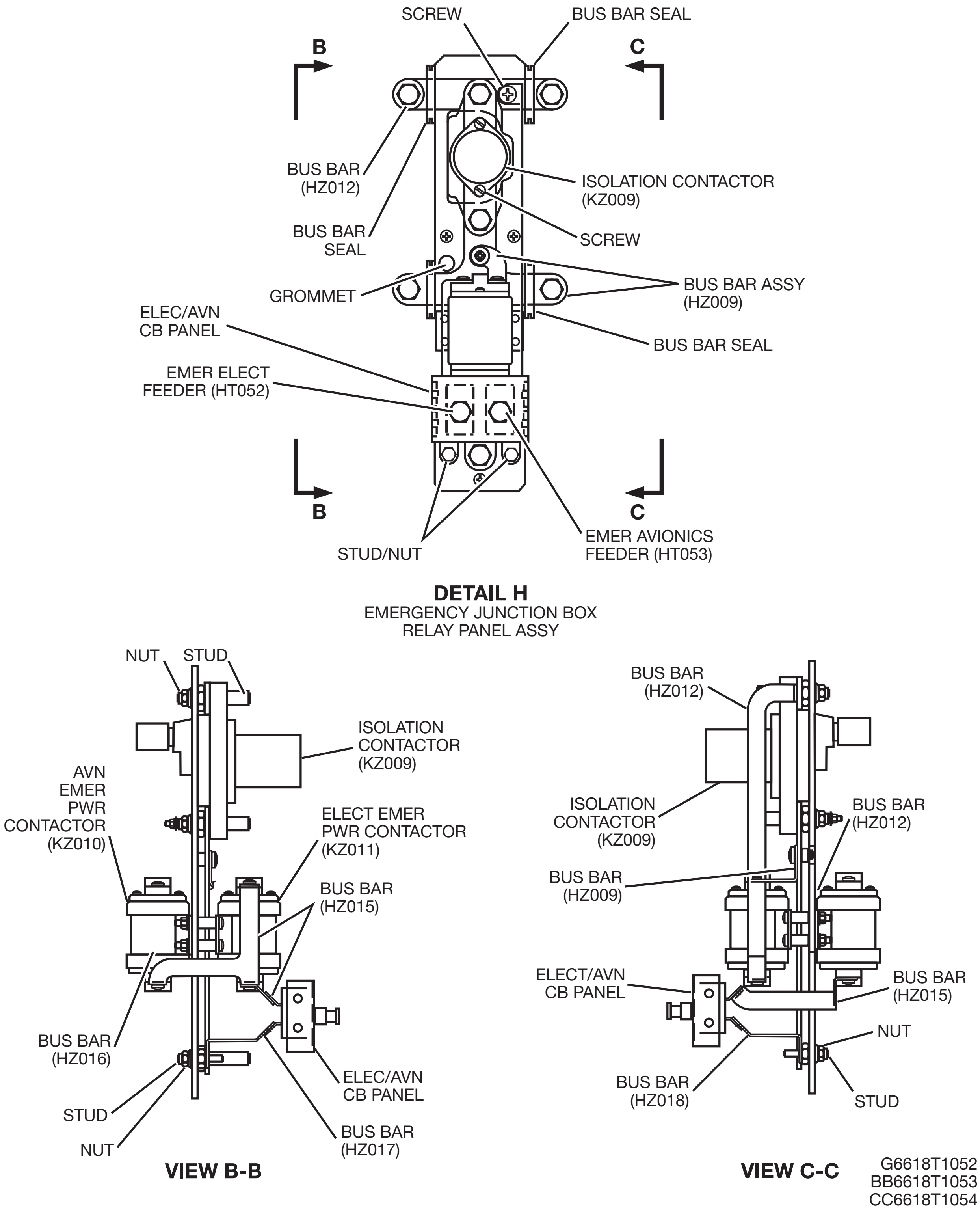


Figure 24-27. Emergency Junction Box Components

The emergency bus (Figure 24-27) is powered from a common crossfeed bus, through isolation bus relays or from the APU power relay.

The CB panels are on the left and right sides of the flight compartment, under the side windows. Each CB panel incorporates various electrical system circuit breakers with the majority of the avionics circuit breakers in the right CB panel. Internal bus bars interconnect groups of circuit breakers. At the back of each CB panel, shields made of fire resistant material are formed and bonded to the aircraft structure. The circuit breakers are identified by silkscreen lettering, illuminated with an electroluminescent light (EL) panel. On the right CB panel is the flight hour meter and a map light control rheostat. There is also a map light control rheostat on the left CB panel, as well as the battery disconnect switch and the interior master switch.

Unless the DC POWER BATT switch is in the ON position no power from the battery bus is applied to the crossfeed or feeder buses. Because the ground external power source is connected to the battery bus, powering any system not directly tied to the battery bus by means of an external source of ground power requires the DC POWER BATT switch to be in the ON position.

The APU DC generator is connected to the crossfeed buses by actuating the APU power relay.

NOTES

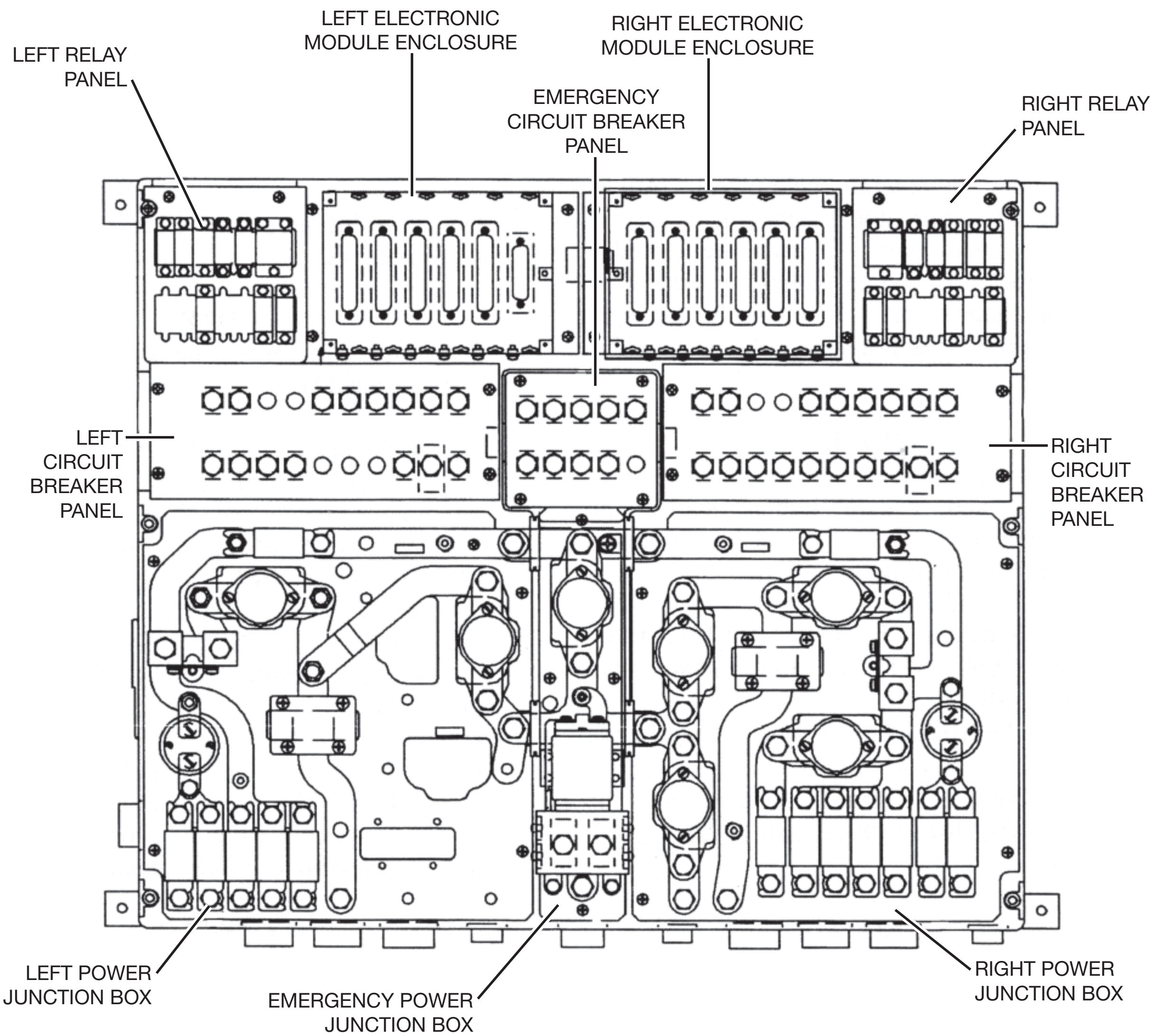


Figure 24-28. Main DC Power Junction Box

COMPONENTS

Power Junction Box

The power junction box contains (Figure 24-28):

- Contactors
- Power relays
- System relays
- Current limiters
- Circuit breakers
- Printed circuit boards (PCBs)
- Terminal strips
- Bus bars
- Other small electrical components

The power junction box may have additional circuit breakers and wiring installed to support specific aircraft configurations. Refer to the *560XL/XLS/XLS+ Wiring Diagram Manual* and the power junction box placard on the power junction box for any changes which may have been made to the power junction box.

The power junction box can be accessed through the right forward tail cone access door. The power junction box incorporates components for the emergency and left and right electrical systems.

NOTES

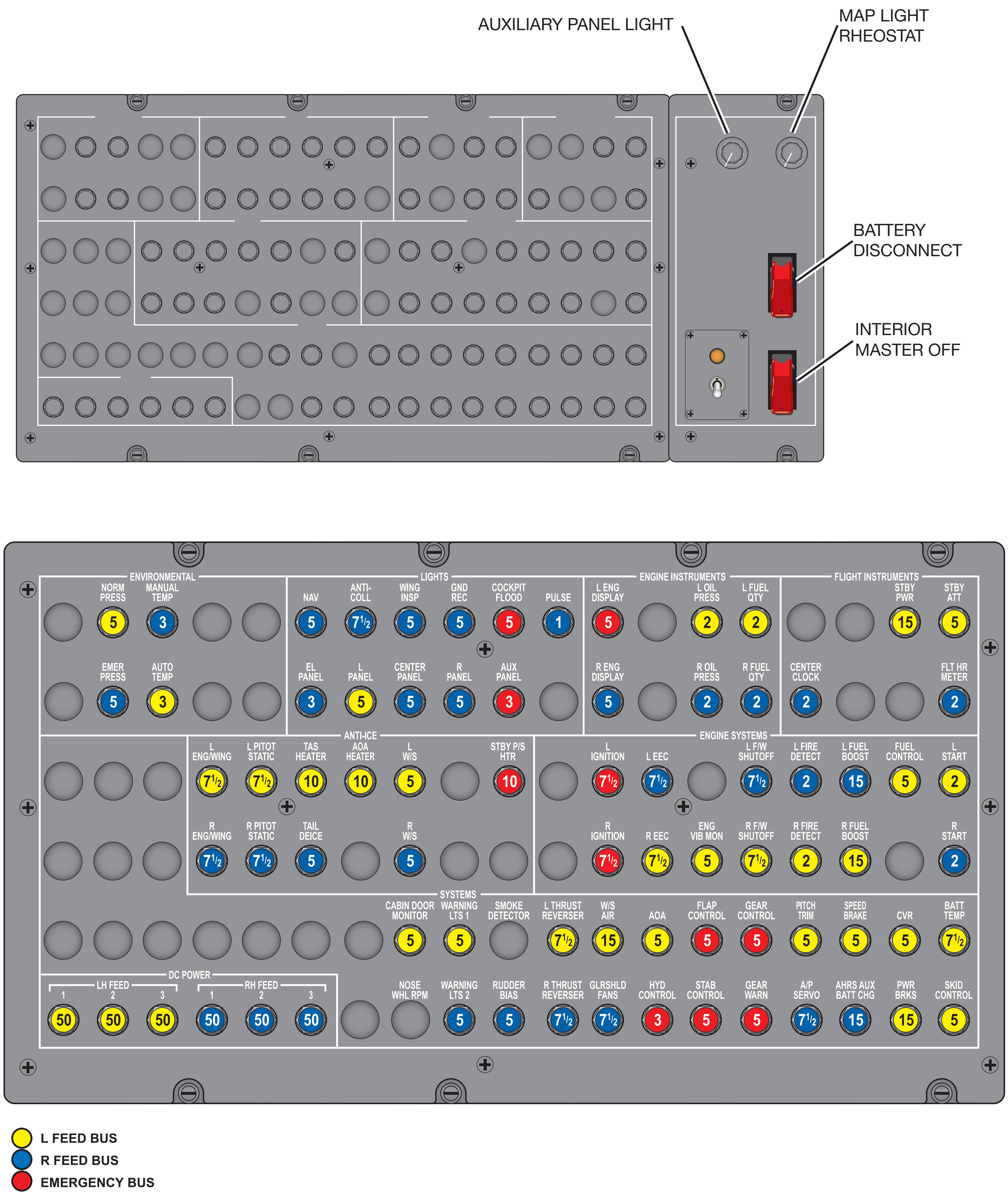


Figure 24-29. Left Circuit Breaker Panel

Main CB Panels

The main circuit breaker panels are on the left and right sides of the flight compartment, under the side windows (Figures 24-29 and 24-30).

The CB panels are rectangular shaped panels with circuit breakers extending through the face. They are secured by jamnuts, lock washers, and a tabbed keyway washer. The entire face of the CB panel is covered by electroluminescent panels, which have holes for the individual circuit breakers to extend through. The electroluminescent panels are secured to the CB panel by screws. Bus bars and jumper wires are secured to the circuit breakers with vendor supplied hardware. Bus bars made of soft copper are installed to interconnect groups of circuit breakers, which receive power from a common power source. Bus bars may be horizontal, connecting groups of circuit breakers in a row; or vertical, connecting circuit breakers in more than one row. Vertical bus bars are variously named crosstie, vertical and main and are coated with an insulating blue fusion bonding epoxy except on the connecting tabs.

The CB panels have electrical disconnect connectors on the aft end of each panel. Feeder wiring and bus connect wiring feed power into the panels. Power out is distributed through the electrical disconnect connectors. In addition to the circuit breakers, there are relays installed in the right CB panel.

NOTES

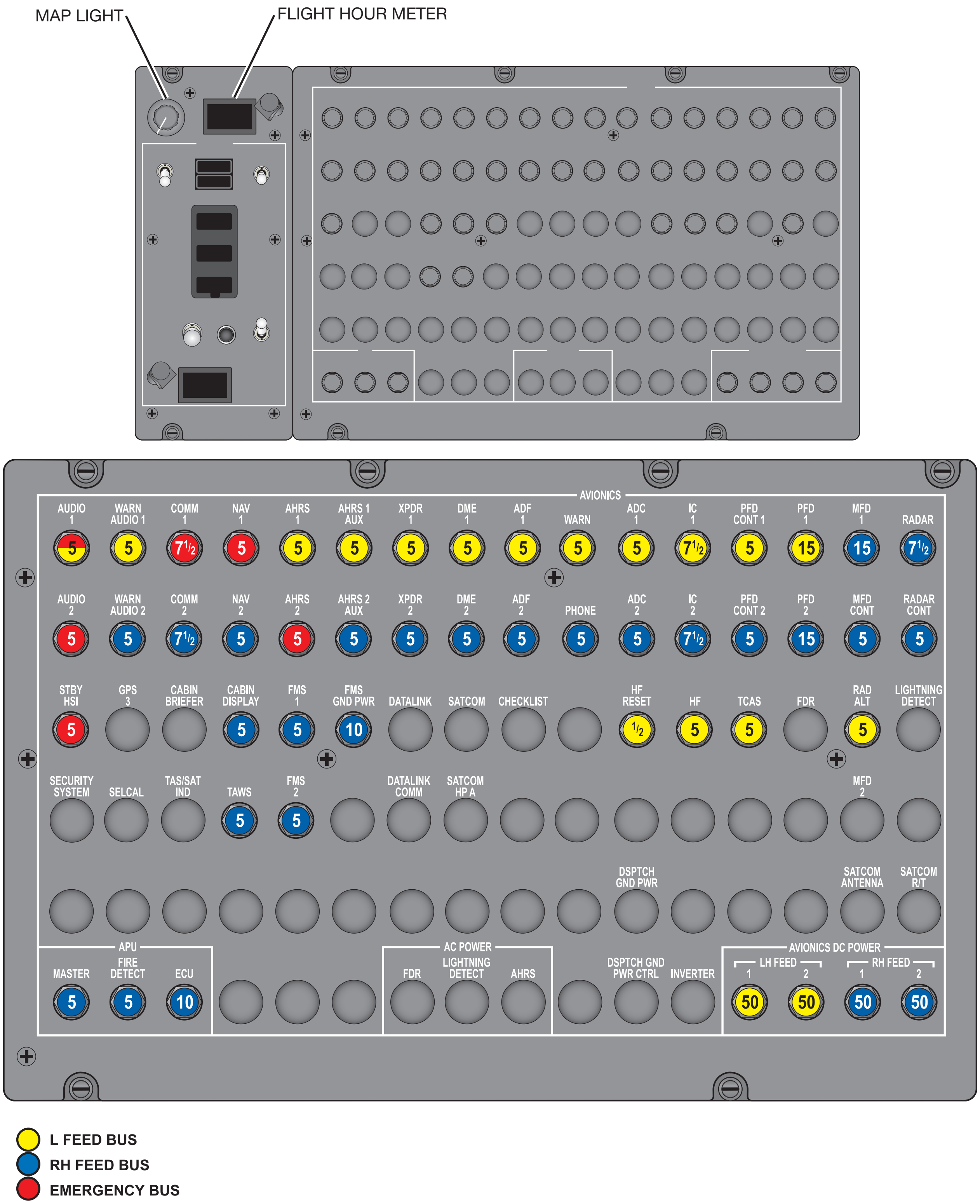


Figure 24-30. Right Circuit Breaker Panel

INTENTIONALLY LEFT BLANK

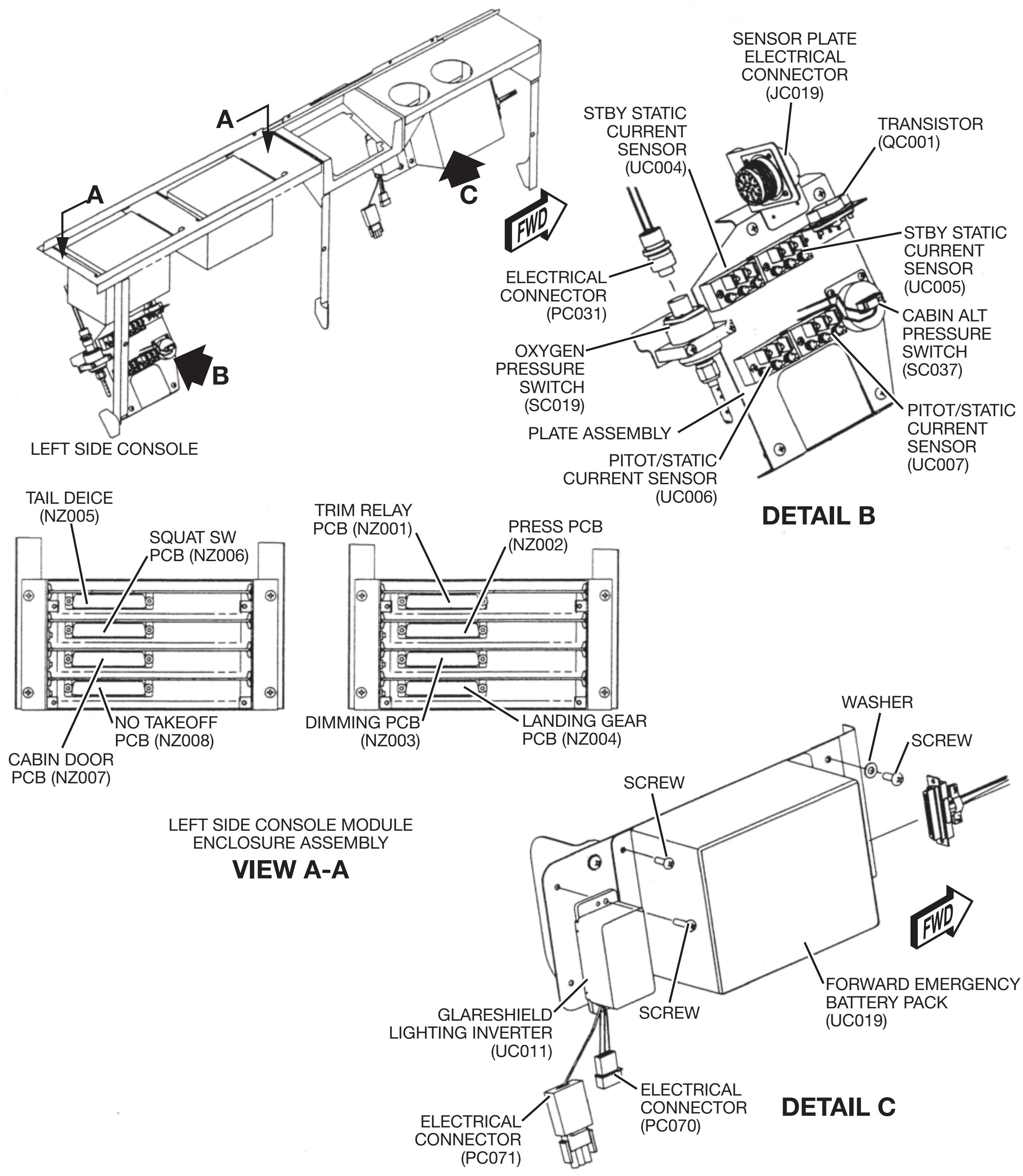


Figure 24-31. Left Side Console Components

Left Side Console Equipment

The left side console equipment components contain (Figure 24-31):

- Relays
- Diodes
- Resistors
- Current sensors
- Transistors
- Pressure switches
- Printed circuit boards for various systems

The side console components are secured to the panel with screws and nutplates.

The electronic module enclosure assemblies contain eight printed circuit boards (PCBs) identified by their pin connector numbers (Table 24-4).

Table 24-4. ELECTRONIC MODULE ENCLOSURE PCBs

PCB	FUNCTION
NZ001	Trim relay
NZ002	Pressurization
NZ003	Dimming
NZ004	Landing gear
NZ005	Tail deice
NZ006	Squat switch
NZ007	Cabin door
NZ008	No takeoff
NZ009	Gear Control

NOTE

NZ005 tail deice PCB moved to avionics bay in left nose and relabeled NZ 031 deice control and NZ 033 deice monitor.

CAUTION

PCBs are extremely sensitive to static discharge damage. Such damage cannot be detected by visual inspection, and may cause premature failure of the PCBs.

NOTES

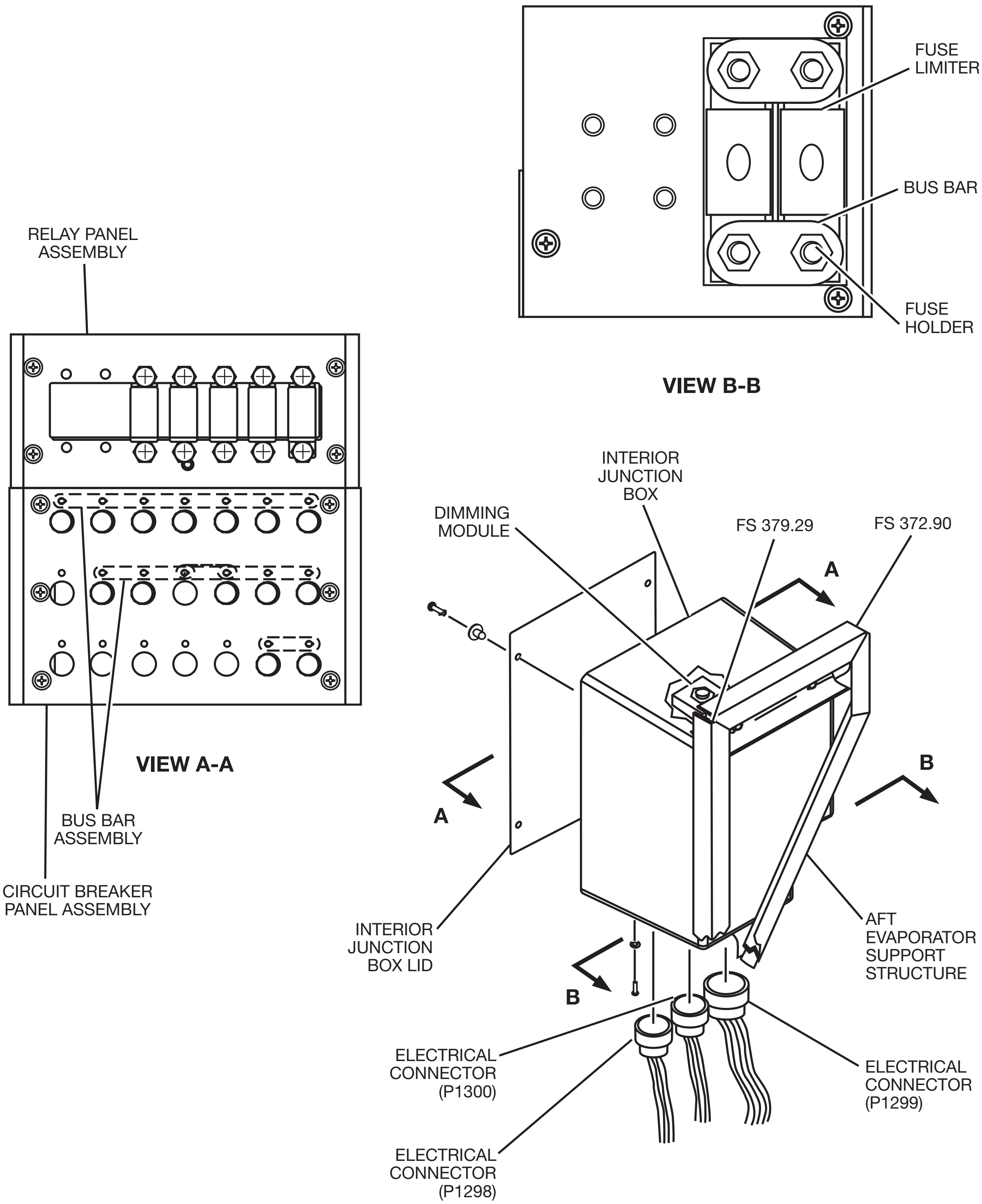


Figure 24-32. Interior Junction Box

Interior Junction Box

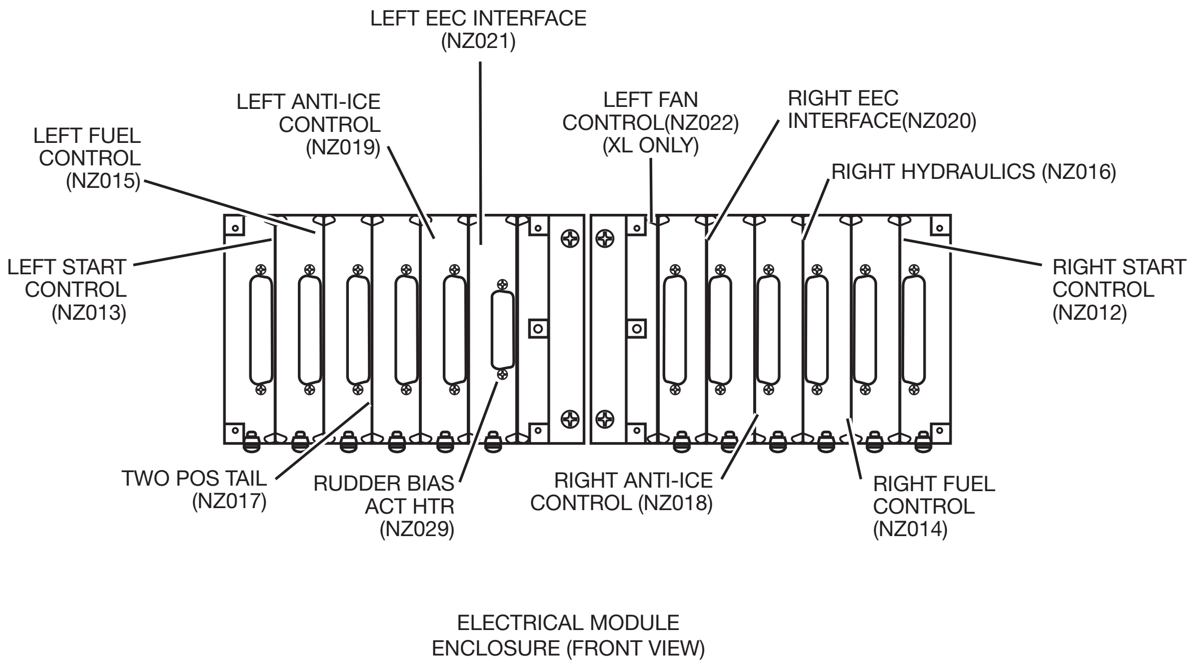
The interior junction box provides DC power distribution to various cabin indirect light systems (Figure 24-32). The interior junction box contains:

- Relays
- Circuit breakers fuse limiters
- Bus bars
- Resistors
- Diodes
- A dimming control module

A silkscreen placard inside the interior junction box lid identifies the various junction box components.

The interior junction box is in the aft cabin between FS 372.90 and FS 379.29 attached to the aft evaporator support structure.

NOTES



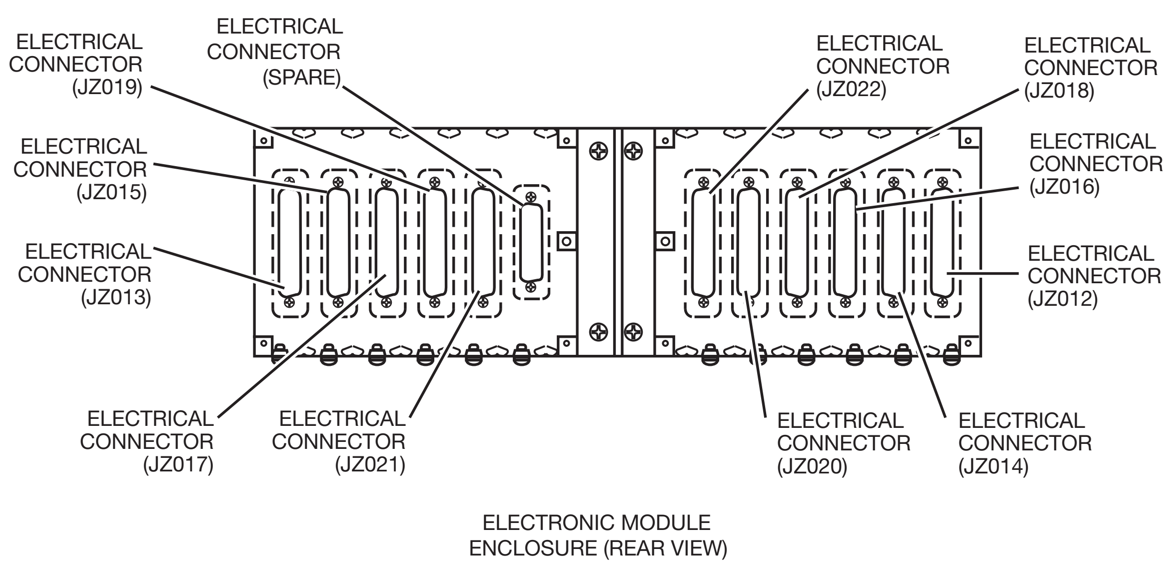


Figure 24-33. Main Power Junction Box PCBs

Power Junction Box Printed Circuit Boards

The printed circuit boards (PCBs) in the electronic module enclosure (Table 24-5), in the left and right power junction boxes, contain relays, diodes, resistors and capacitors (Figure 24-33). Components of each printed circuit board are identified by a silkscreen on the component side of the printed circuit board. Electrical connections are made through an electrical connector.

Table 24-5. MAIN PCB FUNCTIONS

PCB	FUNCTION
NZ012	Right start control
NZ013	Left start control
NZ014	Right fuel control
NZ015	Left fuel control
NZ016	Hydraulic Control
NZ017	Two position tail
NZ018	Right anti-ice control
NZ019	Left anti-ice control
NZ020	Right EEC interface
NZ021	Left EEC interface
NZ022	Fan control
NZ029	Rudder Bias Act. Heater

NOTE

On XLS the NZ022 fan control PCB has been removed.

NOTES

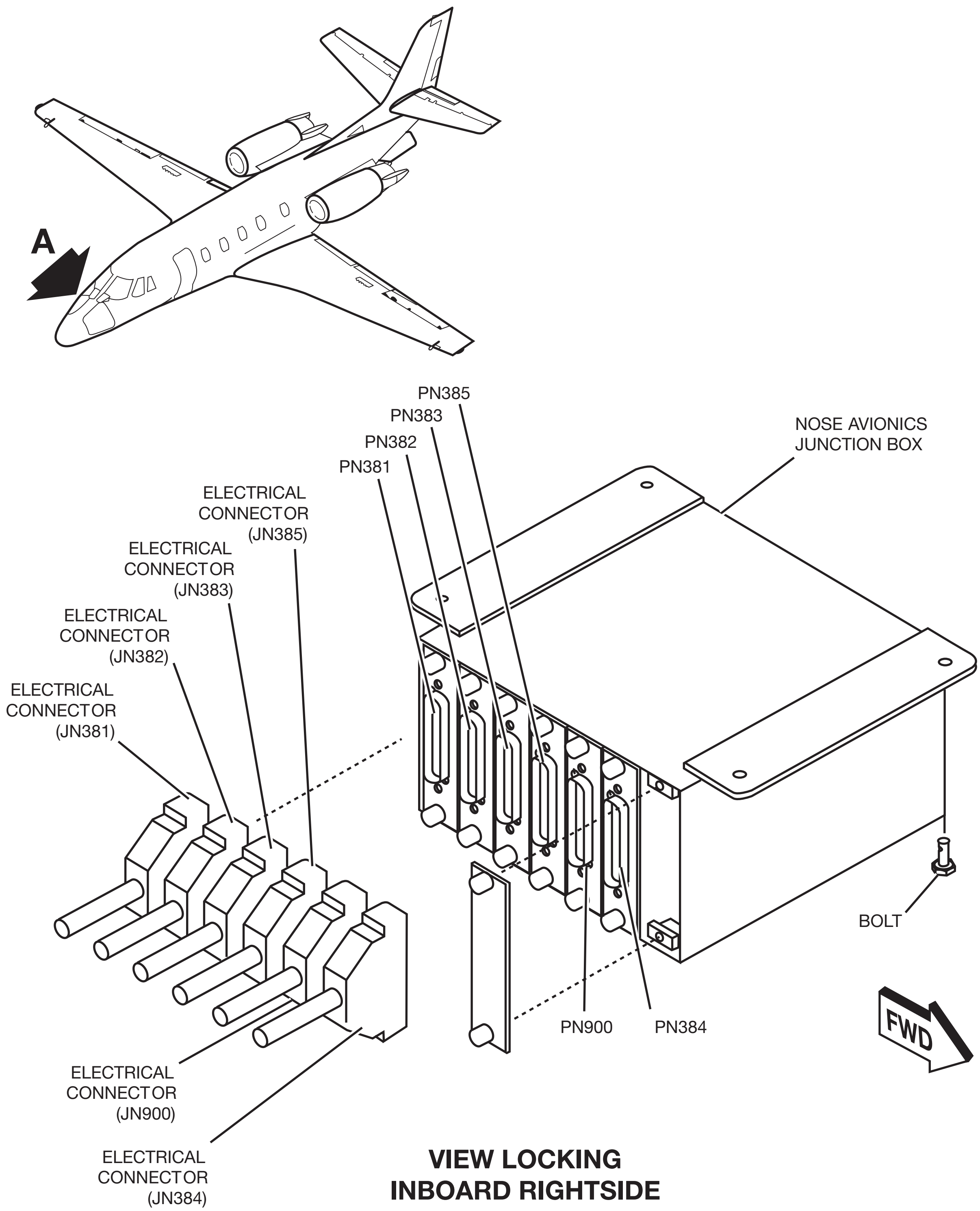


Figure 24-34. Nose Avionics Junction Box

Nose Avionics Junction Box

The nose avionics junction box is in the nose compartment on the right side of the aircraft (Figure 24-34). There is access to the avionics junction box through the right nose door.

The location of each printed circuit board is labeled on the outside of the junction box. The pin connector numbers are also called out on the junction box.

The nose avionics junction box contains six printed circuit boards identified by their pin connector numbers. The following boards and their primary functions are listed in Table 24-6. For a more complete description, refer to the Model 560XL Wiring Diagram Manual and the Avionics Wiring Diagram Manual provided with each aircraft.

Table 24-6. NOSE AVIONICS JUNCTION BOX PCBs

PCB	FUNCTION
PN381	Switching/Jumper Board
PN382	Switching/Jumper Board
PN383	DG/HSI Valid
PN385	DG/HSI Valid
PN900	Lighting/Dim/Test
PN384	DME Switching

NOTES

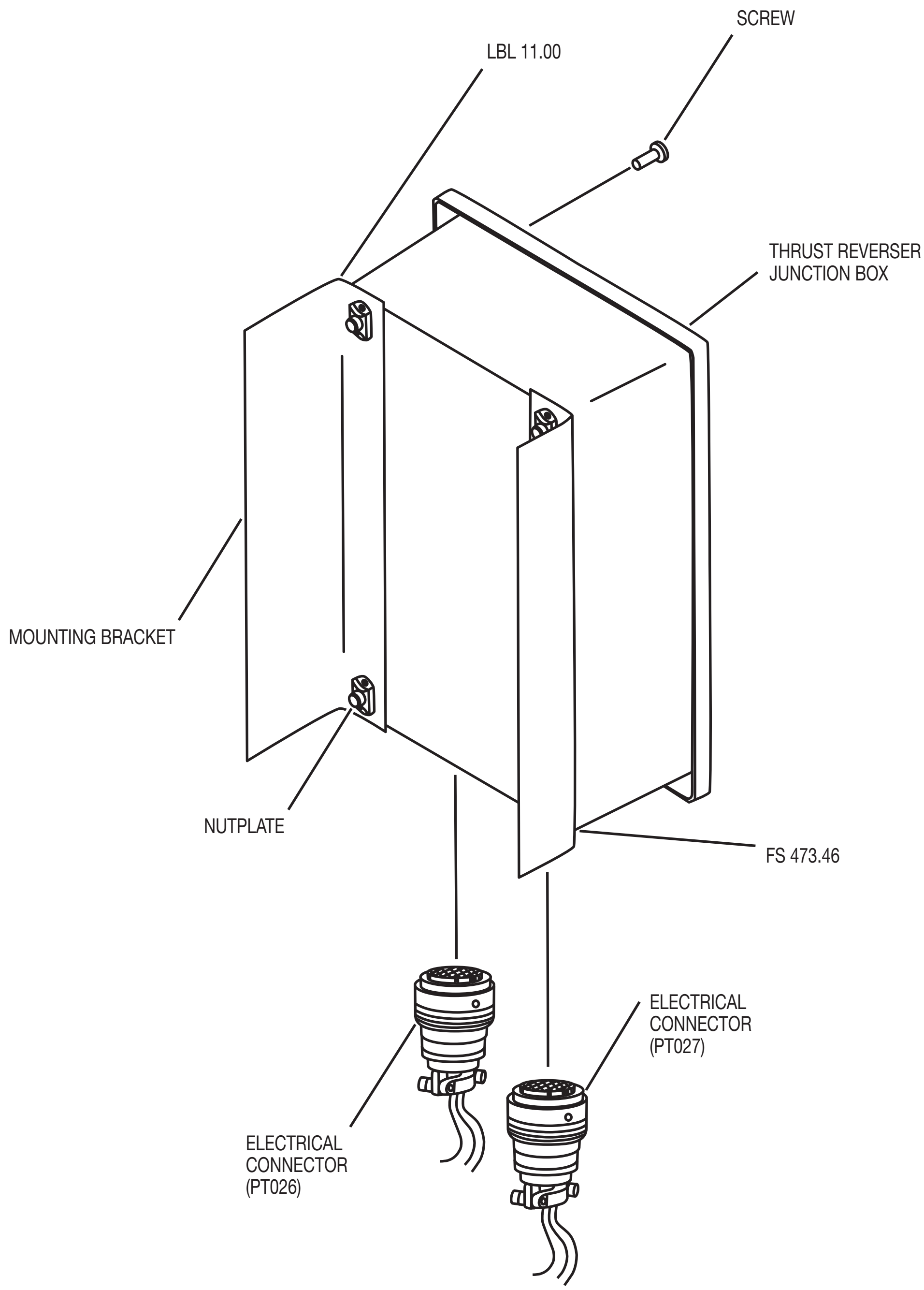


Figure 24-35. Thrust Reverser Junction Box

Thrust Reverser Junction Box

The thrust reverser junction box is in the tail cone compartment on the left side of the power junction box at FS 473.46 (Figure 24-35).

The thrust reverser junction box contains:

- Diodes
- Resistors
- A printed circuit board (PCB4)

The thrust reverser printed circuit board is installed in the thrust reverser junction box with screws, washers and nuts. The printed circuit board contains: transistors, capacitors and diodes. Components of the printed circuit board are identified by a silkscreen, on the component side of the printed circuit board.

NOTES

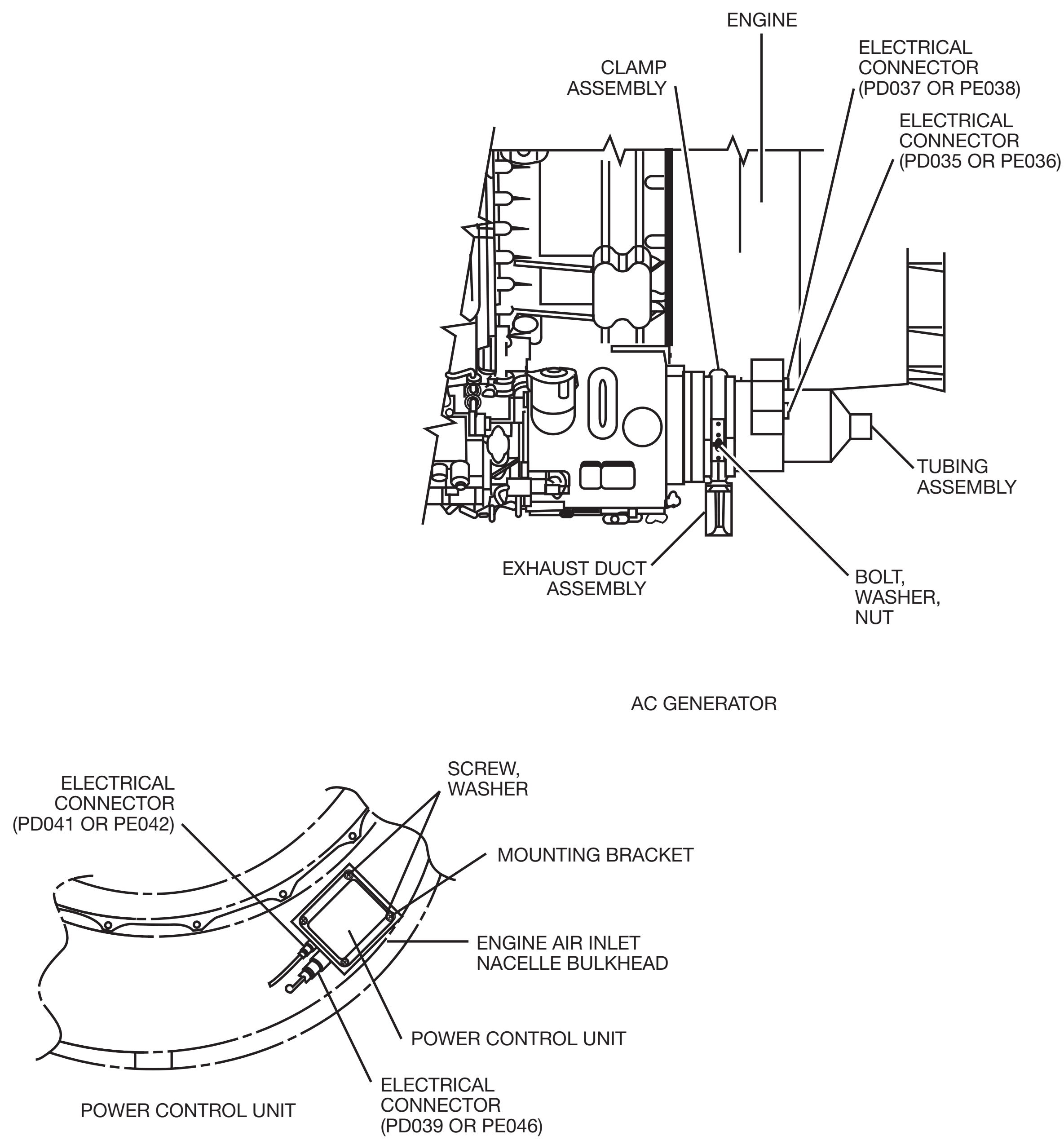


Figure 24-36. AC Alternator System Components

AC POWER GENERATION

DESCRIPTION

This section describes the maintenance of the AC electrical components which generate, regulate and control the AC power for the windshield anti-ice system.

The AC generation system consists of (Figure 24-36):

- Two AC alternator and power control units (PCU) (UD015 left and UE016 right)
- Two engine driven alternators (UD017 left and UE018 right)

Each AC alternator is on the engine accessory drive pad. Each alternator is rated at 3 KVA and 115/200 VAC and operates at a variable frequency of 200 to 400 Hz, depending on engine speeds.

Each AC alternator is regulated for voltage by a PCU, on the aft side of each engine nacelle inlet and forward of the alternator on engine inlet flange “A” at ES 81.46.

NOTES

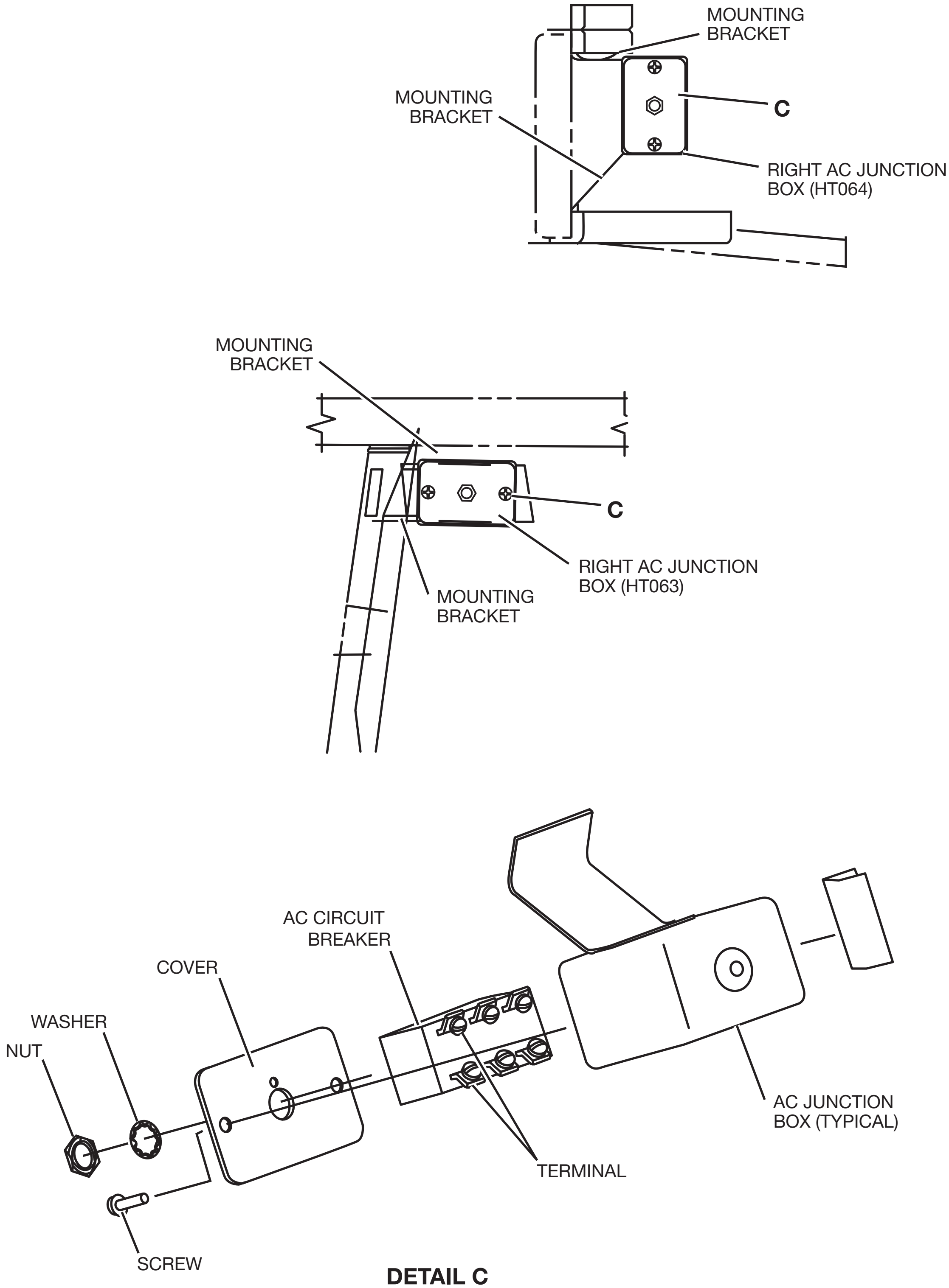


Figure 24-37. AC Junction Box

CONTROLS AND INDICATIONS

AC Junction Box (Circuit Breaker)

The left and right AC junction boxes (circuit breakers) are in the tail cone baggage compartment directly above and across from the tail cone baggage compartment door at FS 433.92 and FS 425.79. Each AC junction box provides three-phase AC power distribution for the windshield anti-ice system (Figure 24-37). Each AC junction box consists of:

- A three-ganged circuit breaker
- Mounting plate
- Junction box cover assembly.

OPERATION

When an engine reaches approximately 6,000 RPM (at idle) the AC alternator is capable of producing 115 VAC. An external turn on signal, from the windshield anti-ice switch, allows the PCU to regulate the output voltage of the alternator. Then, power is supplied to the windshield anti-ice system. The frequency of the output voltage is not critical; however, it ranges from 200 Hz to 400 Hz, depending on engine speed.

NOTES

DIAGNOSTICS

AC System Troubleshooting

The AC generation system is a fully self-contained system. The only function of the generator and power control unit (PCU) is to provide power to the windshield anti-ice system.

A fault of either the AC alternator or the PCU causes the W/S FAULT L or R annunciator to illuminate. Using a voltmeter, check for 115 VAC on each AC circuit breaker (HT063, left and HT064, right) in the tail cone baggage compartment. If 115 VAC is not present at the AC circuit breakers, substitution of a known good PCU may help isolate the problem.

Each AC alternator is also equipped with two switches, at each bearing location. If a bearing should fail, a secondary bearing assumes that the load, and switch for a bearing is grounded causing a bearing indication. Indication of an AC alternator bearing failure illuminates the L–R AC BEARING annunciator. If the L–R AC BEARING annunciator illuminates, the affected AC alternator is about to fail and must be replaced.

NOTE

Each alternator contains an anti-rotation pin at the end plate. When installing an alternator, ensure this pin is properly aligned with engine accessory drive pad.

NOTES

QUESTIONS

1. A fully charged battery should supply power to the BATTERY bus and the EMERGENCY buses for approximately:
   A. 2 hours
   B. 1 hour
   C. 30 minutes
   D. 10 minutes

2. If either red BATTERY OVERTEMP CAS message appears, the BATTERY switchlight should be initially placed to ______ to isolate the battery from the generators and obtain a voltage reading.
   A. OFF
   B. EMER
   C. Either A or B
   D. None of the above

3. With generators online, BATT switch in BATT ON, and the VOLTMETER SELECT switch remaining in BATT, the voltmeter gauge indicates:
   A. Generator system voltage, 28.5 V, from the crossfeed bus
   B. Generator system voltage, 28.5 V, from the battery bus
   C. Battery voltage, 24–25 V, from the battery bus
   D. Battery voltage, 24–25 V, from the crossfeed bus

4. If the amber DC GENERATOR OFF L CAS message appears:
   A. Right generator ammeter gauge should indicate double the previous load
   B. Left generator amperage should drop to zero
   C. Voltmeter should register zero with the VOLTMETER SELECT remaining in the BATT position
   D. Both A and B

5. If the amber J-BOX-LIMITER CAS message appears:
   A. Aft J-BOX 60 amps feed bus current limiter is open
   B. Aft J-BOX 225 amps feed bus current limiter is open
   C. Generators should be selected OFF one at a time to determine which limiter is open
   D. Aircraft should be landed as soon as possible

6. Select the correct statement concerning the use of a ground power unit:
   A. Never connect the power cord to or remove it from the aircraft with power applied.
   B. The battery does not receive a charge if the BATT switchlight is in BATT ON.
   C. The generator switches must be OFF for the engine start when using the GPU.
   D. The GPU ground unit must be regulated at 24 volts and 800/1,000 amps.

7. If the battery voltage indicates 24 volts prior to engine start:
   A. Battery is low and must be charged to 28 volts
   B. GPU must be used for starting
   C. 24 volts is the minimum voltage required
   D. Voltage is excessive and could damage the starter

8. When selecting an external power unit to be used for ground power starts, the unit should be limited to:
   A. 1,000 amps, 24 volts
   B. 1,200 amps, 24 volts
   C. 1,000 amps, 28.5 volts
   D. 1,200 amps, 28.5 volts

9. Concerning starting limitations, which is the correct statement?

A. Three battery starts per hour are allowed.
B. A generator assisted start counts as one battery count.
C. If four or more battery starts are performed in one hour, the battery must be allowed to cool for 30 minutes.
D. There are no starter limitations when using a GPU.

10. External power overvoltage protection is provided by the:

A. Battery disconnect relay
B. External power control relay
C. 225 amp current limiter(s)
D. Overvoltage monitor, located in the main J-box.

11. With external power applied to the aircraft, the battery receives a charge:

A. When external power control relay is energized
B. Except during an engine start
C. Only with the battery switch in the BATT position
D. All of the above

12. What would indicate an open start CB?

A. AFT-JBOX CB illuminates
B. GEN OFF illuminates
C. AFT-JBOX LMT illuminates
D. No annunciator illuminates

1**3**.Starter/generator overhaul is required:

A. At engine overhaul
B. During each phase 5 inspection
C. When the brushes are changed
D. Each 1,000 hours of operation

14. With the battery switch in BATT:

A. The emergency bus is deenergized and all other busses are energized.
B. Voltage on the voltmeter is read with the battery switch in EMER only.
C. The battery isolation relay is energized and emergency power relay remains deenergized to supply DC power to the aircraft.
D. The left ammeter indicates less than 30 amps if battery needs servicing.

15. The RH GEN OFF light illuminates; this indicates that the:

A. Field relay has opened
B. Generator relay has opened
C. Right generator is supplying 30 amps more than the left generator
D. Voltage selector switch is stuck in the LH GEN or RH GEN position

16. The battery electrolyte level should be checked only when:

A. Battery is fully charged
B. Battery is fully discharged
C. Replacement of a cell is necessary
D. There is a possibility of reverse polarity

17. Which of the following conditions cause the field relay in the GCU to open?

A. Secondary overvoltage of 40 volts or more, firewall shutoff depressed or ground fault being sensed
B. Overvoltage, firewall shutoff being depressed or ground fault being sensed
C. Power in on pin A, start relay control inoperative, or bus sense on pin R more than 0.3 volts
D. Loss of power in on pin D, generator switch in OFF, or power in on pins X or Y

18. Using a GCU breakout box, you observe no voltage on pin L with the right start button pushed. Why?

A. No voltage is available on pin A of the right GCU
B. Wire to pin X on right GCU is open
C. No ground on pin FF
D. 24 volts is available on pin *S of right GCU

19. With external power applied, the right engine running and the right generator on, left engine start is accomplished with:

A. External power only
B. Battery only
C. Battery and right generator assist
D. Battery and external power unit power

20. During engine start, the ground fault system is disabled by a relay on the start PCB:

A. To prevent nuisance tripping of the start cycle
B. Preventing damage to the 225 amp current limiters
C. Preventing damage to the aircraft electrical circuit in case the external power overvoltage monitor malfunctions
D. Ensuring that the start relay closes before the power relay closes

21. Which busses would lose electrical power if the right 225 amp current limiter is open (engine not operating)?

A. Emergency, left crossover and right main extension busses
B. All left DC busses
C. All right DC busses
D. Left main extension and right crossover busses

22. If the first engine is started by the aircraft battery and the second engine is started by generator assist, which of the following switch positions is correct?

A. Battery switch OFF
B. Battery switch OFF, generator switches OFF
C. Battery switch in BATT or EMER, generator switches in GEN
D. Battery switch in BATT, generator switches in GEN

23. With both engines operating and both generators on, the equalizer circuit allows the generators to share the load within:

A. ±100 amps of each other
B. ±60 amps of each other
C. ±30 amps of each other
D. ±20 amps of each other

24. The splines on the start/generator drive shaft are lubricated:

A. By hand each 50 hours, using Mobil 10 lightweight high temperature grease
B. By engine lubricating oil during engine operation
C. By spraying WD-40 into the starter/generator air inlet while motoring the engine (observe starter/generator duty cycle)
D. By hand with DC33 at starter/generator overhaul

CHAPTER 25
EQUIPMENT AND FURNISHINGS

CONTENTS

ILLUSTRATIONS

CHAPTER 25
EQUIPMENT AND FURNISHINGS

INTRODUCTION

This chapter provides information on the equipment and furnishings in the Citation XL/XLS/XLS+ flight compartment, passenger compartment and baggage compartment. Emergency equipment and insulation are also included in this chapter. Special order equipment/furnishings are not defined in this chapter.

GENERAL

This chapter is divided into sections and subsections to assist maintenance personnel in locating specific equipment and furnishings. A brief description of each section herein is as follows.

The Flight Compartment section—Describes the upholstery, trim and equipment in the flight compartment. It includes the headliner, window trim, windshield trim, upholstery, glare shield, pedestal covers, sunvisors, carpet, seats and equipment, such as navigational chart cases, oxygen masks and smoke goggles.

The Passenger Compartment section—Describes equipment and furnishings within the passenger compartment. It includes the headliner, passenger service units (PSU), upholstery, trim, carpet, seats, couch, dividers, forward closet, tables, seat drawers, magazine racks and storage cabinets.

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The Refreshment Center section—Describes maintenance practices for the standard and optional refreshment centers. The storage cabinets are also included in this section.

The Vanity section—Describes the standard and deluxe vanity with sink.

The Baggage Compartment section—Describes the upholstery within the baggage compartment. It also includes loose equipment.

The Emergency Equipment section—Describes the locator transmitter system, life vests and water barrier.

The Insulation section—Describes the insulation and acoustical dampening material that is installed in the aircraft.

FLIGHT COMPARTMENT

This section describes the upholstery, trim and equipment in the flight compartment. The flight compartment is the area from the forward pressure bulkhead to, but not including the forward divider. The headliner, window trim, windshield trim, upholstery, glare shield, pedestal covers, sunvisors, carpet, seats and equipment, such as navigational chart case, oxygen mask and smoke goggles are included in this section.

HEADLINER

The headliner is along the top of the flight compartment. Cutouts and inserts are provided for mounting air outlets (Wemacs), chart lights, warning horns, audio speakers and cockpit floodlights.

WINDOW AND WINDSHIELD TRIM

The window and windshield trim is a decorative molding that fits around the windows and windshields. The trim is held in place with screws and clips.

UPHOLSTERY PANELS

There are five upholstery panels along each side of the flight compartment. They are held in place with quarter-turn fasteners.

GLARE SHIELD

The glare shield is secured to the structure at the top of the instrument panel. There are integral nutplates for installation of the fire tray retaining screws. The glareshield is covered with a black material to reduce glare.

PEDESTAL COVERS

Two covers (access plates) are provided, one on each side of the pedestal for access to wire bundles, throttle switches and cables.

SUNVISOR

The standard sunvisor consists of a telescoping rod and shade assembly. The sunvisors prevent glare from the sun for the pilot and copilot. The sunvisors are adjustable for use on the windshield and side windows.

CARPET

The carpeting in the flight compartment consists of a multiple piece carpet section held in place with Velcro fasteners.

FLIGHT CREW SEATS

Description

This section contains maintenance procedures for removal, installation, and adjustment of the flight crew seats. The maintenance practices for the pilot and copilot seat are typical.

The flight crew seats (pilot and copilot) are mounted on tracks and are adjustable forward-aft and up-down, and have an adjustable seat back tilt.

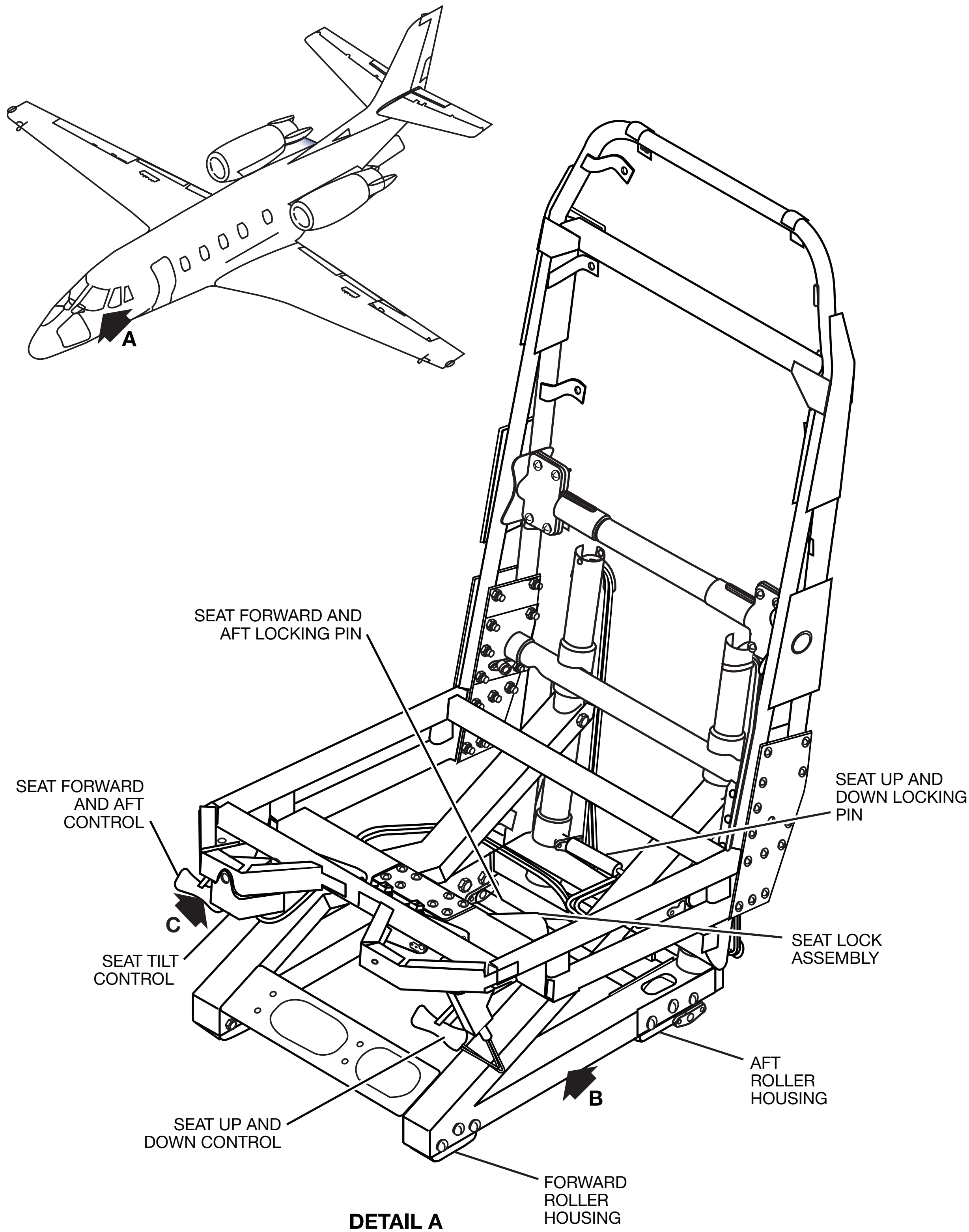


Figure 25-1. Flight Crew Seat Installation

There are three control handles (levers) for adjusting the position of the seat. the outboard lever adjusts the seat vertically. The inboard lever adjusts the seat to the desired reclining position. The center lever adjusts the seat forward and aft. Optional crew seats are equipped with a lumbar support system controlled by a mechanical knob on the side of the seat-back assembly. Pilot and copilot seats are equipped with an in-flight relief tube assembly and overboard drain system. The relief horn and hose assembly are stowed under the seat when not in use. There is a life vest in the back pocket of each seat. The fire extinguisher is below the copilot seat. Each crew seat incorporates a 5-point restraint system with inertia reels.

Holes in the flight crew seat rails mate with seat lock pins to lock the forward and aft seat movement in the desired position. Seat stop bolts, spacers and nuts (installed in the rails) restrict individual seat movement to a specific position on the rails.

For flight crew seat cleaning instructions, refer to Chapter 12—"Interior."

Diagnostics

Remove Flight Crew Seat

Refer to Figure 25-1.

1. Move the seat back straight up and collapse the seat bottom.
2. Release the control lock and move the control column to the extreme forward position.
3. Remove the forward and aft seat stop bolts, spacers and nuts.
4. Lift the forward and aft seat control to release the stop pins. Move the seat forward until front rollers are free from the seat tracks, and the aft rollers align with the seat rail opening.
5. Remove the seat from the seat rails. Exercise care during removal to prevent contact between the seat and the pedestal or instrument panel.

Install Flight Crew Seat

1. Move the front rollers just forward of the seat rails. Align the aft rollers with rail opening. Drop aft rollers onto the seat track and slide the seat onto the seat tracks.
2. Lift the forward and aft control to keep the stop pins retracted, and slide the seat to the aft adjustable position. Release the forward and aft control.

CAUTION

Make sure that forward and aft seat stop bolts, spacers and nuts are installed. Failure to install forward seat stops may permit the seat to roll off forward end of seat track and interfere with control column.

3. Install the seat stop bolts, spacers and nuts in the inboard and outboard seat rails.
4. Set the control lock.
5. Move the seat-back and seat-base to the desired position.

Remove the seat forward and aft control assembly

1. Remove the flight crew seat. Refer to the Remove Flight Crew Seat removal/installation section.
2. Remove the screws that attach the control assembly pin housing to the roller housing.
3. Cut the wire ties, which attach the control cable assembly to the seat.
4. Remove the screws, which attach the control handle housing to the seat assembly.
5. Remove the forward and aft control assembly from the seat assembly.

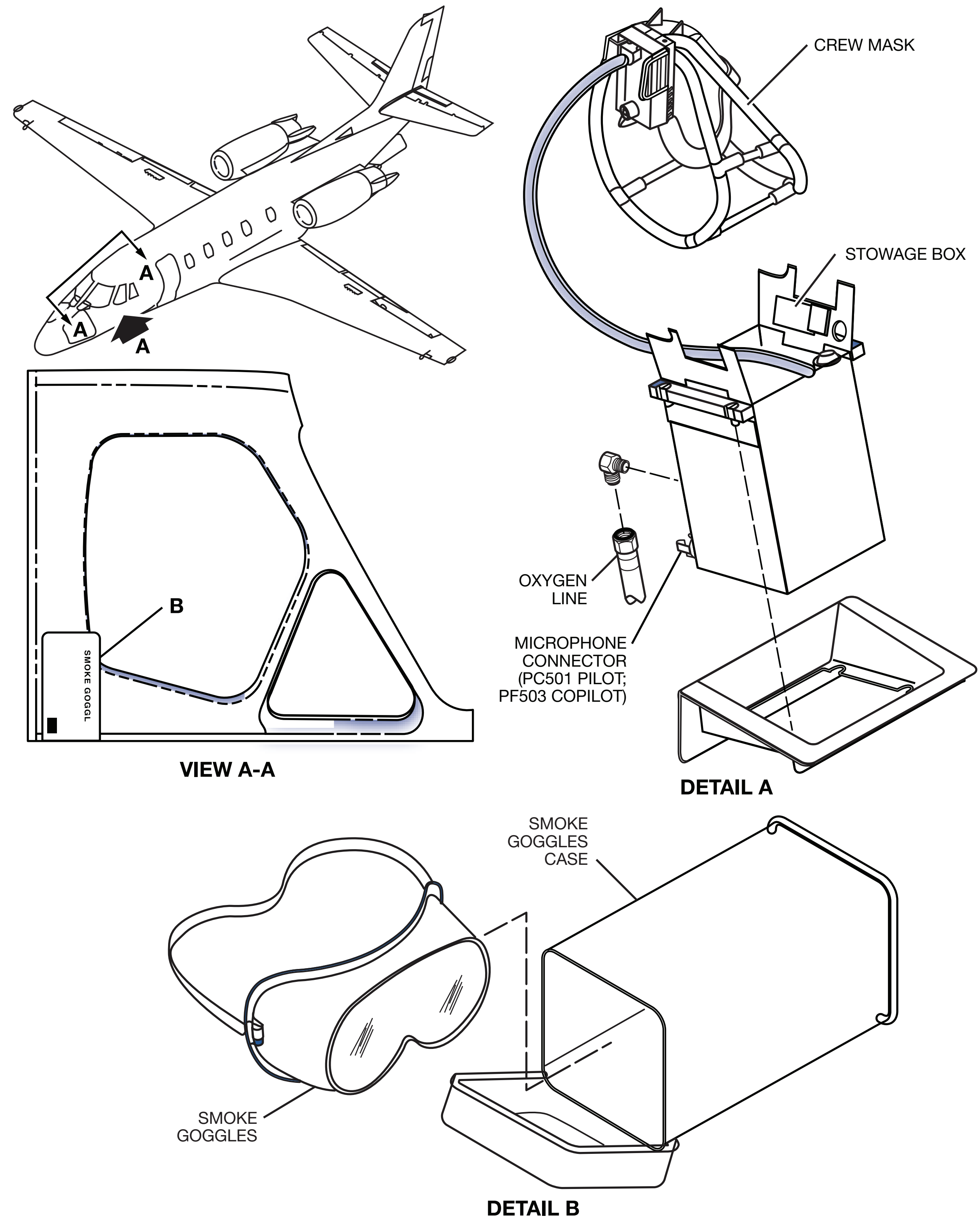


Figure 25-2. Smoke Goggles and Crew Oxygen Mask Storage

Install forward and aft control assembly

1. Align the stop pin with the center hole.

2. Attach the control assembly pin housing to the roller housing.

CREW SHOULDER HARNESS AND SEAT BELT

Description

The shoulder harness holds the upper torso. The harness strap (webbing) reels onto and off of an inertia reel during normal body movements. However, during sudden body movement forward (0.75G to 1.25G) the inertia reel automatically locks the harness strap.

The shoulder harness has an inertia reel attached to the seat base and two harness straps. The harness straps and seat belts are connected by a rotary buckle. This system is known as a four-point restraint system. The harness (webbing) is not replaceable.

The seat belts (lap) and a restraint strap restrain the lower torso. The seat belts have a left half and a right half. The belt halves attach at one end to the seat frame, and buckle to the rotary buckle at the other end.

The seat belt (lap) halves and shoulder harnesses connect to a rotary buckle. One half of the belt is attached to the buckle and does not release. The left and right belt halves have an adjuster to adjust the belt, and to center the rotary buckle.

A restraint strap can be added to the four-point restraint system. The restraint strap is on the bottom seat frame and locks into the bottom of the rotary buckle to form a five-point restraint system.

FLIGHT COMPARTMENT EQUIPMENT

Navigation Chart Case

A three-book navigation chart case may be on the front side of the forward refreshment center (or cabin divider). They are used to store navigational charts and flight manuals. A two-book navigation chart case is optional for the same location.

Oxygen Mask

Each crew oxygen mask is housed in a storage box in the pilot and copilot side consoles (Figure 25-2).

Smoke Goggles

Smoke goggles are provided for the pilot and copilot to prevent eye irritation due to smoke in the event of an emergency. The goggles are in a storage case behind each aft openable window.

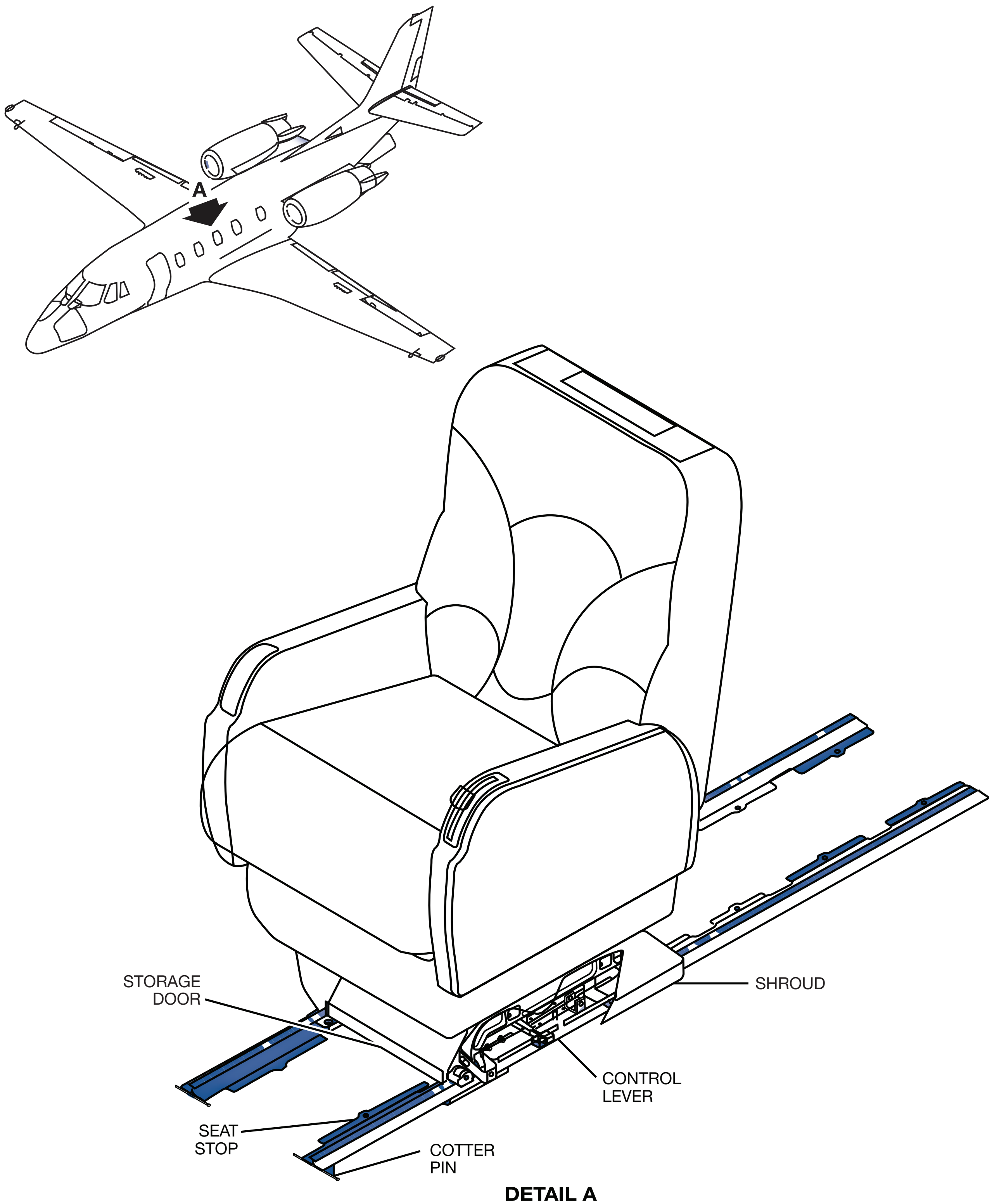


Figure 25-3. Foward and Aft Facing Passenger Seat Installation with Floor Tracking

PASSENGER COMPARTMENT

PASSENGER SEATS

Passenger seating is available in various types or designs depending on interior options. Passenger seating types consist of (Figure 25-3):

- Forward and aft facing seats
- Left forward side facing seat
- Left aft side facing seats
- Right forward two place side facing couch
- Right forward two place side facing couch with armrest

Description

Standard forward facing and aft facing passenger seats are a fixed pedestal design with fore and aft tracking (7 inches) and lateral tracking (4 inches) on the seat base.

Passenger seats recline to an infinite number of positions, with a total of 50° recline. This function is controlled by a lever on the armrest of each seat. All passenger seats are equipped with:

- Seat restraints
- Sliding headrests
- Single armrests
- Swivel capability
- A flotation device stored in the seat base shroud

Passenger seat options consist of:

- Dual armrests
- Seat back pockets
- Alternate tailoring and floor tracking

There are seat stops in locations specified by the weight and balance data shields. The seat stops can track forward and aft, and must be secured according to the methods described in this section.

NOTES

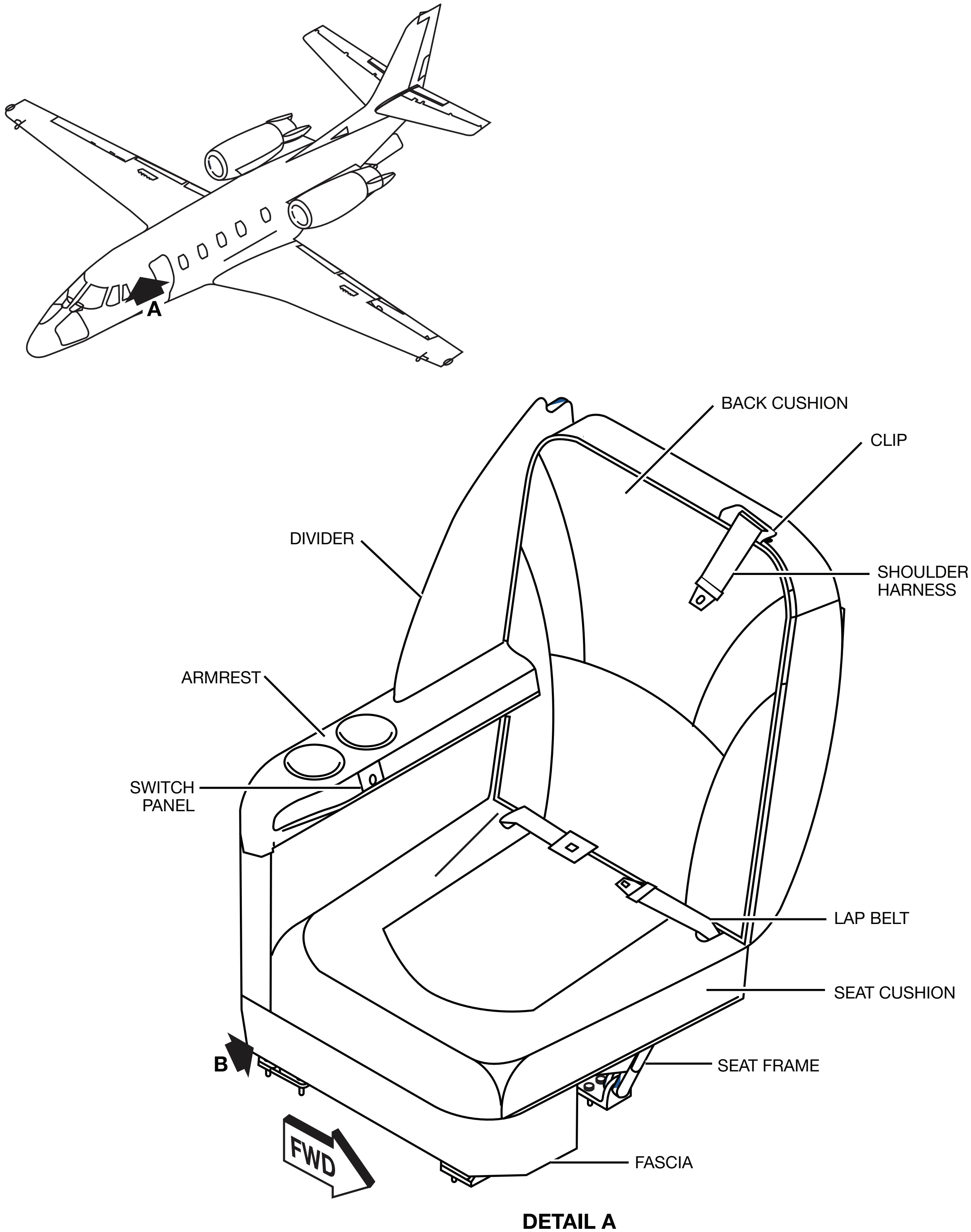


Figure 25-4. Side-Facing Seat Installation

SIDE-FACING SEATS

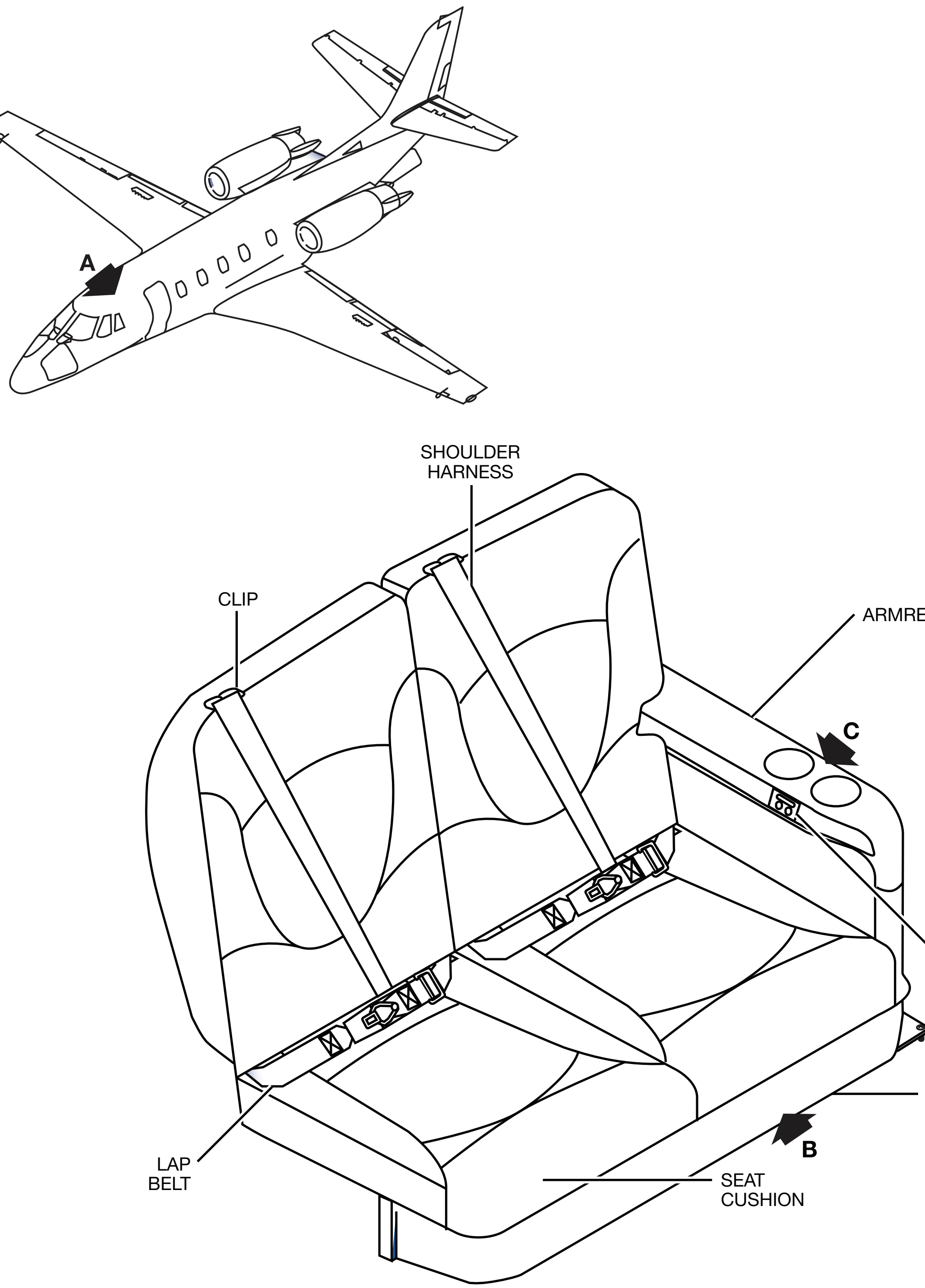
Description

The left-forward side facing seat (Figure 25-4) is forward of the main cabin door if the left-forward refreshment center is not installed. This seat option is installed with an armrest and lexan divider. A life vest is stored beneath the seat cushion and held in place by Velcro.

The left-aft side facing seat is between the aft bulkhead closet and the aft cabin divider if the left belted toilet is not installed. This seat can also be ordered with an optional storage net. The storage net is attached behind the sidewall and can be used with the seat back folded down or with the seat completely removed. The storage net must be neatly stowed behind the sidewall when not in use. A life vest is stored beneath the seat cushion and is held in place by Velcro.

Both forward and aft side-facing seats have a restraint system consisting of a single shoulder harness and lap belt. When the shoulder harness is unbuckled, an inertia reel retracts the harness, allowing freedom of movement.

NOTES



DETAIL A

Figure 25-5. Two-Place Small Couch Installation

COUCHES

Description

The right forward two-place, side-facing couch (Figure 25-5) is aft of the forward cabin divider; and is used in conjunction with the 16.5-inch closet assembly. Life vests are stored beneath the seat cushion, and held in place by Velcro.

The right forward two-place, side-facing couch (with armrest) is aft of the forward cabin divider; and is used in conjunction with the 8-inch closet assembly. The folding armrest is equipped with cup holders. It is stowed in the seat back when not in use. Life vests are stored beneath the seat cushions, and are held in place by Velcro.

Both side-facing couches include a restraint system consisting of two shoulder harnesses and lap belts. When the shoulder harness is unbuckled, an inertia reel retracts the harness, allowing freedom of movement.

NOTES

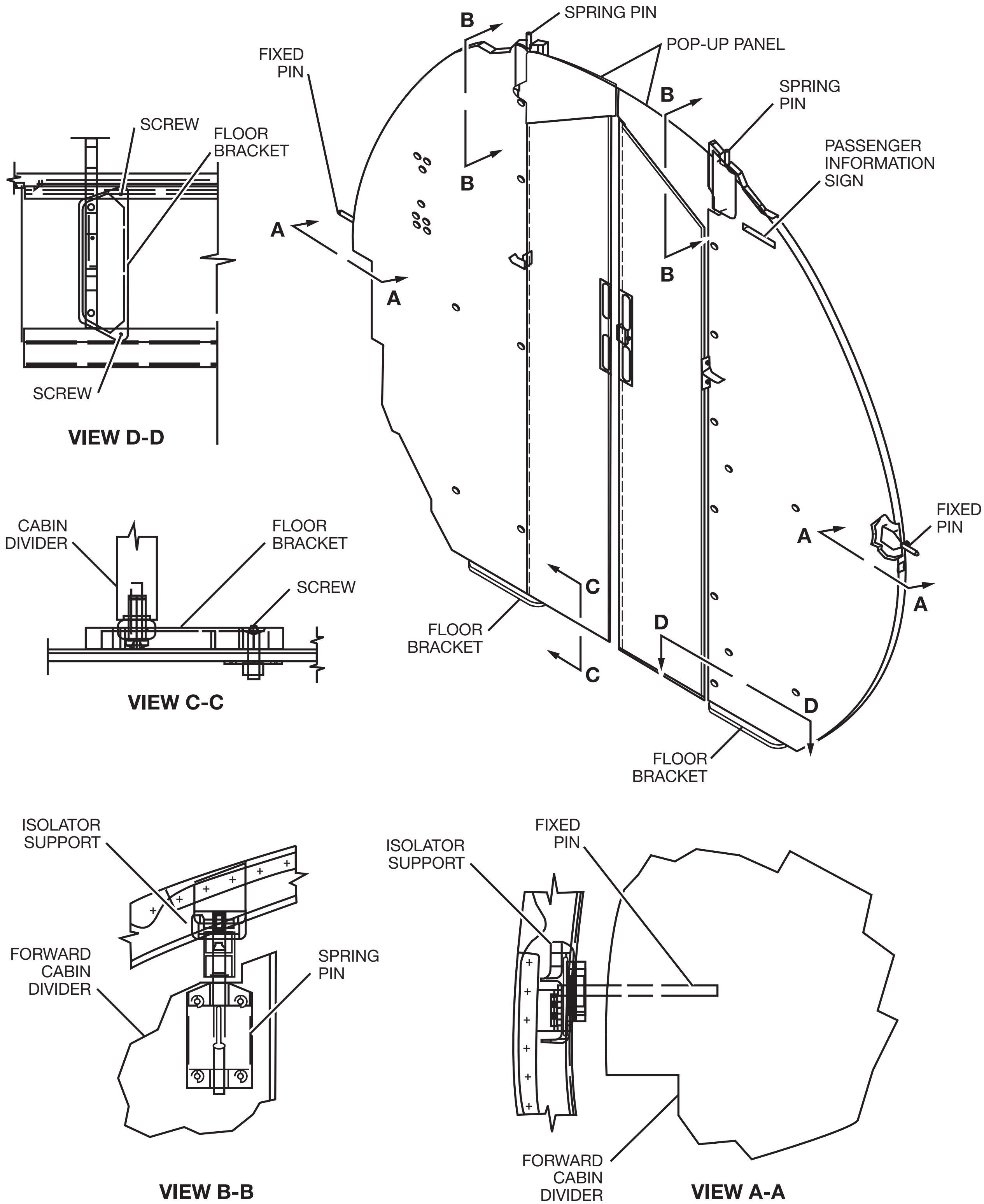


Figure 25-6. Optional Forward Cabin Divider

CABIN DIVIDERS

This section describes removal and installation procedures for both standard and optional cabin dividers (forward and aft).

Description

The standard forward cabin divider (Figure 25-6) consists of a single panel with a sliding curtain which closes the opening to the flight compartment. When the curtain is closed it is secured to the divider with 6 snaps. The curtain is stowed on the right divider (with a tie strap) when is not in use.

The optional forward cabin divider incorporates a sliding door between two panels, to close the opening to the flight compartment. Each door has a pop-up panel, used to close the doorway.

Standard aft cabin dividers incorporate a sliding door with mirror/fabric panels to close the opening to the vanity area. Each door has a pop-up and pop-down panel, used to close the doorway.

Optional aft dividers have fabric covering facing the interior cabin (in place of mirror/fabric panels).

Dividers also provide a place to mount items, like navigation chart cases, the Flightfone and passenger information signs.

The dividers are constructed of a composite honeycomb core; and are available in a variety of finishes.

Aircraft equipped with forward closets or refreshment centers may incorporate the standard curtain or the optional sliding door as part of the assembly. These options are removed or installed as a complete assembly. Refer to the appropriate section within this chapter for information on removing and installing these items.

NOTES

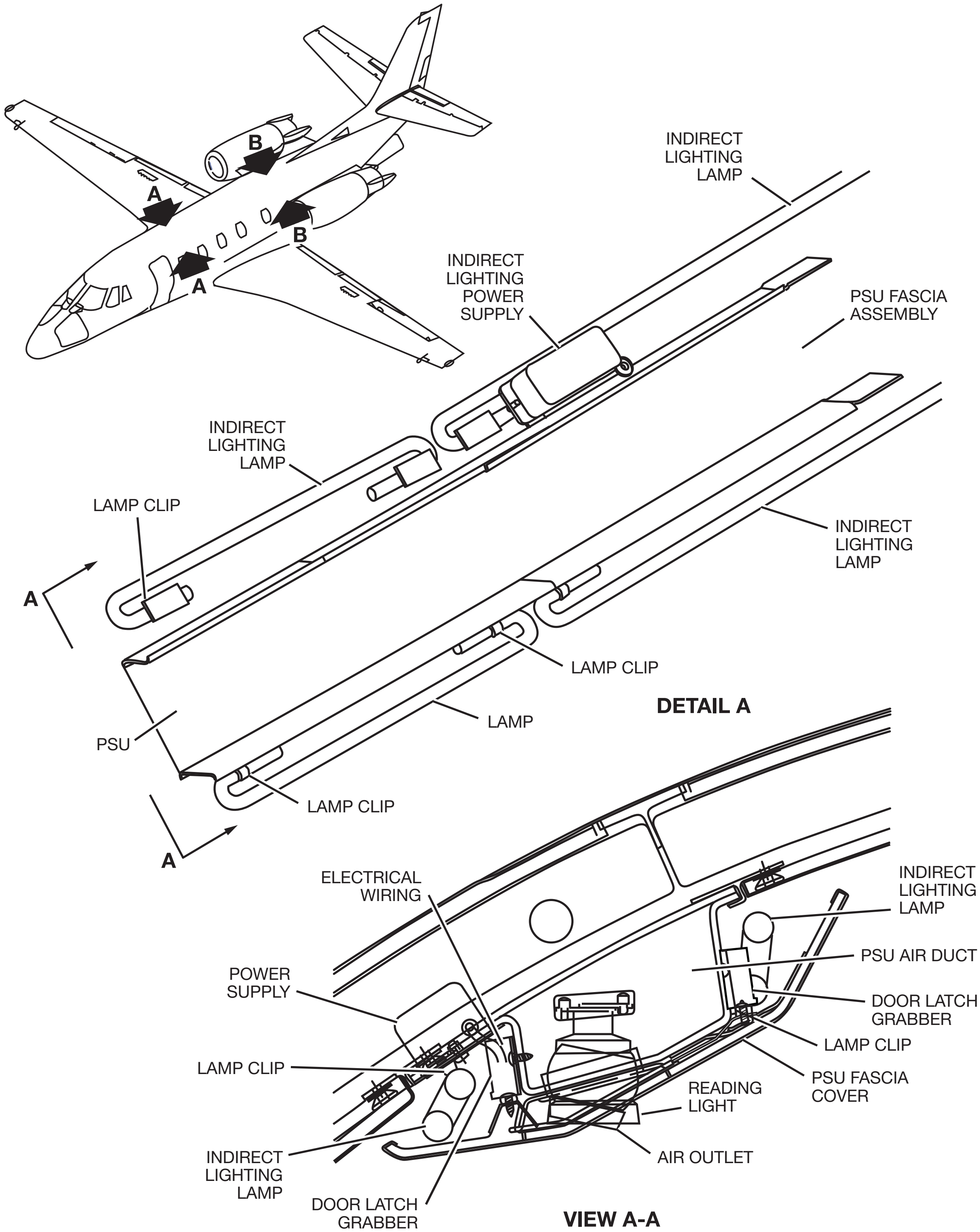


Figure 25-7. Passenger Service Unit Installation

PASSENGER SERVICE UNIT

This section contains removal and installation procedures for the passenger service units (PSU) in the passenger compartment (Figure 25-7).

CAUTION

To prevent damage when installing a PSU, ensure that electrical wiring is properly clamped, and is not chafing or being pinched between the PSU and structure.

The passenger service unit functions to incorporate:

- Indirect lighting
- A reading light
- Provide conditioned air for passenger comfort

Description

PSUs are along each side of the passenger compartment (above the seats). They incorporate:

- Indirect lighting
- Air outlets (Wemac)
- A reading light along the bottom portion

The passenger service unit consists of:

- Left and right main PSU assemblies
- Left and right forward PSU assemblies

Forward PSU assemblies may differ depending on interior cabinet options. When cabinets are under a PSU, both the indirect lighting and air outlets (Wemacs) are removed in that area. If interior cabinet options are changed at a later date, both these functions can be reinstalled. The PSU assemblies are constructed of klegecell core and phenolic skin panels.

NOTES

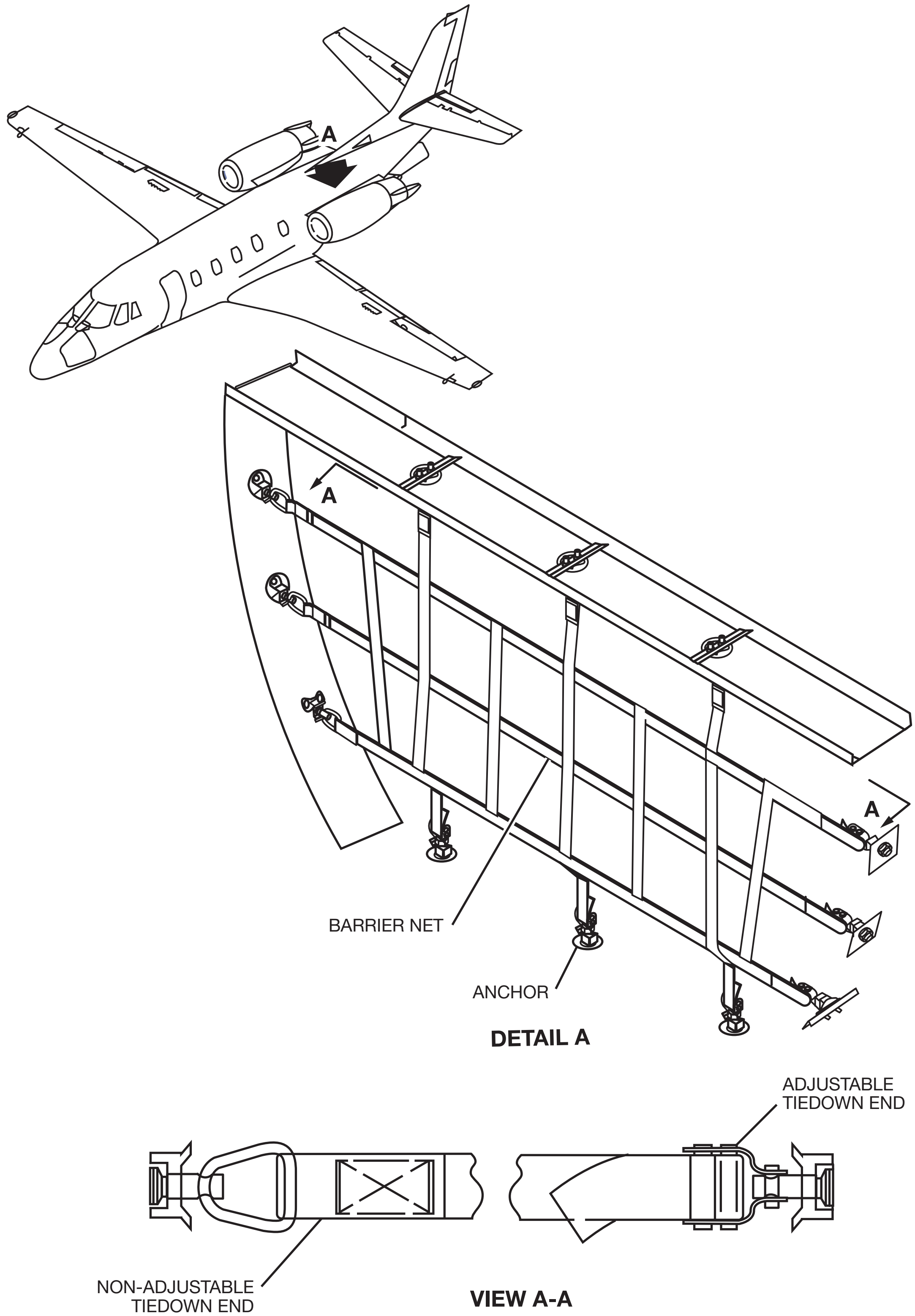


Figure 25-8. Barrier Net Installation

BAGGAGE COMPARTMENT

This section describes upholstery in the baggage compartment.

TAIL CONE BAGGAGE COMPARTMENT

The tail cone baggage compartment is aft of the pressure bulkhead. The tail cone baggage door is below the left engine pylon. Tie-down anchors and a barrier net assembly are provided for securing baggage or cargo.

UPHOLSTERY

The floor upholstery panels consist of padding with a fabric covering. They are held in place with Velcro fasteners and tie-down anchor plates. The overhead panels are painted to match the interior. The side upholstery panels are held in place with Velcro fasteners.

BARRIER NET

A barrier net (Figure 25-8) is provided to secure baggage and other cargo in the baggage compartment. The ten-strap net is constructed of 1-inch wide webbing material. All the tips of the webbing are heat sealed.

Install the barrier net with the adjustable buckle ends positioned left and down for proper orientation. Each strap end assembly is fitted with a track-fitting which connects with tie-down anchor plates found in the baggage compartment.

NOTES

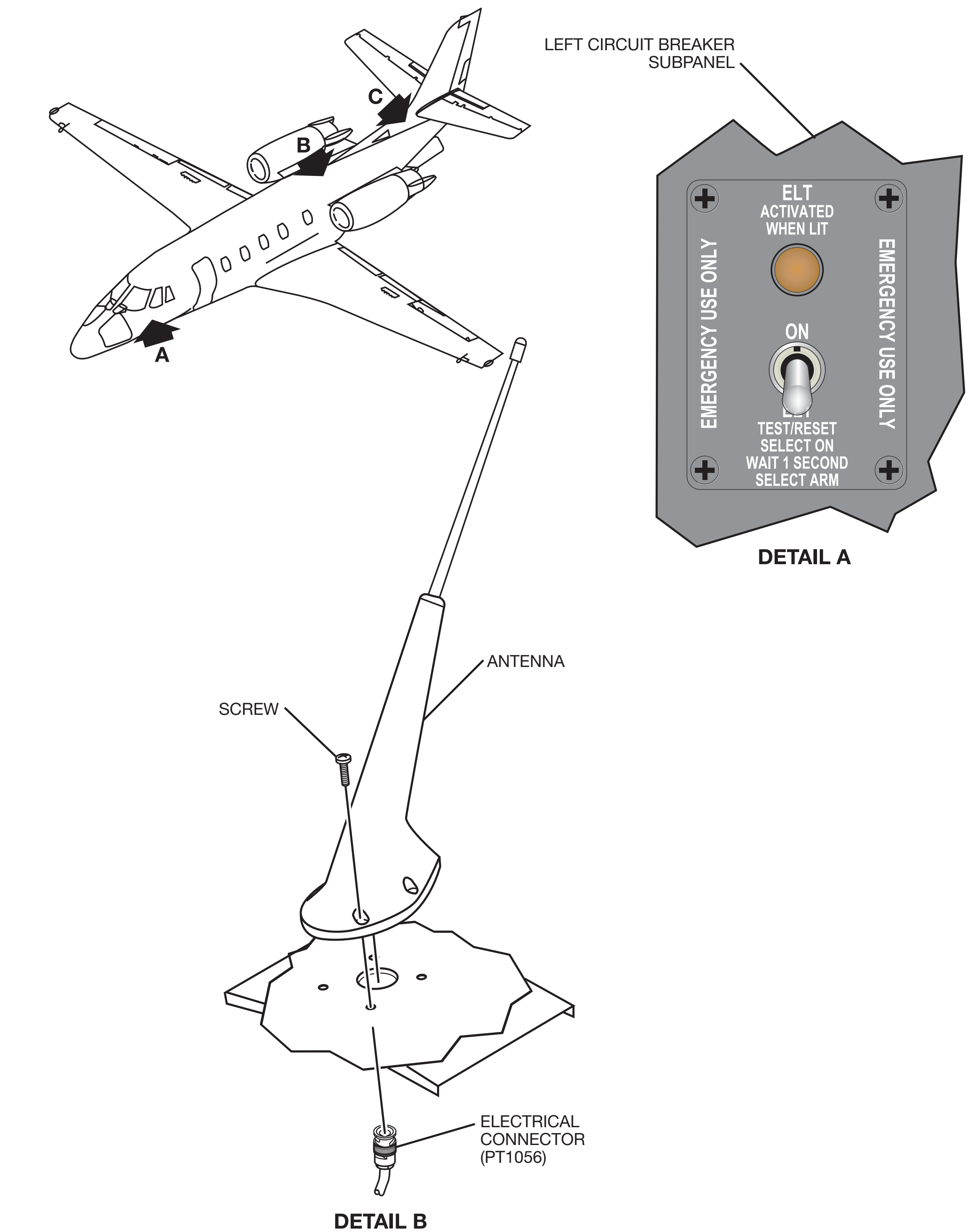


Figure 25-9. Artex ELT 110-4 Locator Beacon System

EMERGENCY EQUIPMENT

This section describes of the emergency equipment used in the aircraft. The emergency equipment covered includes the locator beacon system, life vests, water barrier, and their operation.

LOCATOR BEACON SYSTEM

The locator beacon system is used in emergency conditions. The transmitter is tuned to the VHF emergency channel and transmits a tone modulated signal.

Artex ELT 110-4 Emergency Locator Transmitter System

The Artex ELT 110-4 Emergency Locator Transmitter System (ELT) operates over a wide range of environments to aid rescue teams in locating aircraft in the event of a crash.

Description

The ELT 110-4 emergency locator transmitter system consists of (Figure 25-9):

- A transmitter with integral battery pack
- G-switch
- An antenna
- Remotely mounted cockpit
- Tail cone control switch
- A cable assembly
- An antenna coax cable

The transmitter (with integral battery pack and G-switch) is on a tray-mounted in the dorsal fin at FS 489.75. The system activates automatically in the event of aircraft impact, or manually through one of the remotely-mounted switches. When the aircraft BATT switch is ON, the microprocessor in the transmitter utilizes power from the aircraft electrical system. Power from the transmitter's integral alkaline battery pack is used for the system test sequence and to activate or sustain the system in case of an emergency.

The ELT 110-4 system utilizes an antenna, forward of the dorsal fin (at FS 416.14, RBL 3.68). The antenna is connected to the transmitter with a coaxial cable.

Controlling devices for the system include the G-switch mounted in the transmitter, and a remotely mounted switch. On the left CB subpanel, a two-position guarded switch allows flight crew to activate, reset or test the system.

NOTES

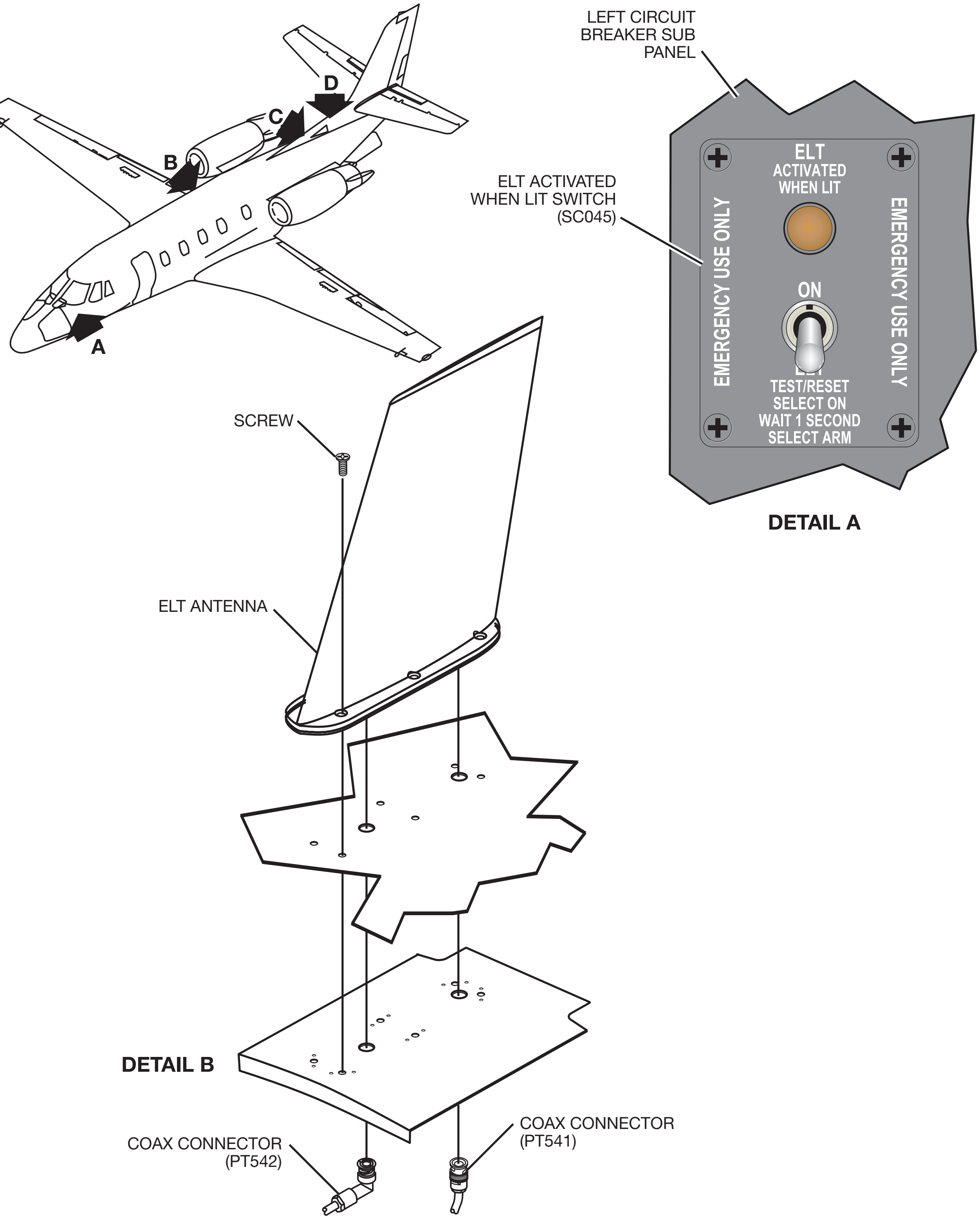


Figure 25-10. Artex 110-406 Emergency Locator Transmitter System Installation

Artex ELT 110-406 Emergency Locator Transmitter System

An Artex 110-406 Emergency Locator Transmitter (ELT) System helps rescue teams locate the aircraft if there is a crash. It operates in a wide range of environmental conditions and is resistant to forces caused by many types of accidents.

Description

The Artex 110-406 Emergency Locator Transmitter (ELT) system has (Figure 25-10):

- A transmitter
- An integral battery pack
- A G-switch
- An ELT antenna
- A remote mounted switch on the left CB subpanel
- A cable assembly
- An antenna coax cable

The transmitter has an integral battery pack and a G-switch installed in a tray. It comes on automatically if the G-switch is actuated or if the cockpit panel switch is ON.

When the aircraft electrical system is on, a microprocessor in the transmitter uses power from the aircraft electrical system. Electrical power from the transmitter's integral alkaline battery pack is used for the system test sequence and keeps the system on in case of an emergency.

The Artex 110-406 system uses an ELT antenna found on the top of the fuselage (FS 414.14 and RBL 3.58). The antenna connects to the transmitter with a coaxial cable.

A G-switch (installed in the transmitter) and a two-position ELT ACTIVATED WHEN LIT switch (on the left CB subpanel) are used to control the transmitter. The ELT ACTIVATED WHEN LIT switch allows the flight crew to activate, reset or test the system. An ON/OFF toggle switch on the transmitter is set to the ON position for normal system operation, and to OFF during maintenance or service.

Operation

The Artex 110-406 Emergency Locator Transmitter (ELT) System can be activated automatically by the G-switch or manually, by either one of the two manual control switches.

The G-switch operates and starts the transmitter due to crash accelerations parallel to the longitudinal axis of the aircraft in a forward direction.

The ELT ACTIVATED WHEN LIT switch (SC045) on the left CB subpanel manually operates the transmitter when the transmitter switch is set to the ON position.

When activated, the ELT transmits on emergency frequencies at 121.50, 243.00 and 406 MHz (at the same time) with a swept tone at three sweeps-per-second.

The 121.50 and 243.00 MHz frequencies send a locator signal that can be followed by those that are receiving it. The 406 MHz frequency activates a satellite tracking system. The Artex 110-406 system is connected to the navigational system of the aircraft as well as the transponder system. When the ELT system is in operation, the location and the tail number of the aircraft is transmitted on the 406 MHz frequency.

The Artex 110-406 system also has a complete self-analysis program with test routines that are transmitted at reduced power over frequencies 121.50, 243.00 and 406 MHz.

The test sequence examines the system microprocessor, antenna and transmitter. The test sequence starts when the remote switch is set to the ON position for one second, then moved to the ARM position switch the system off.

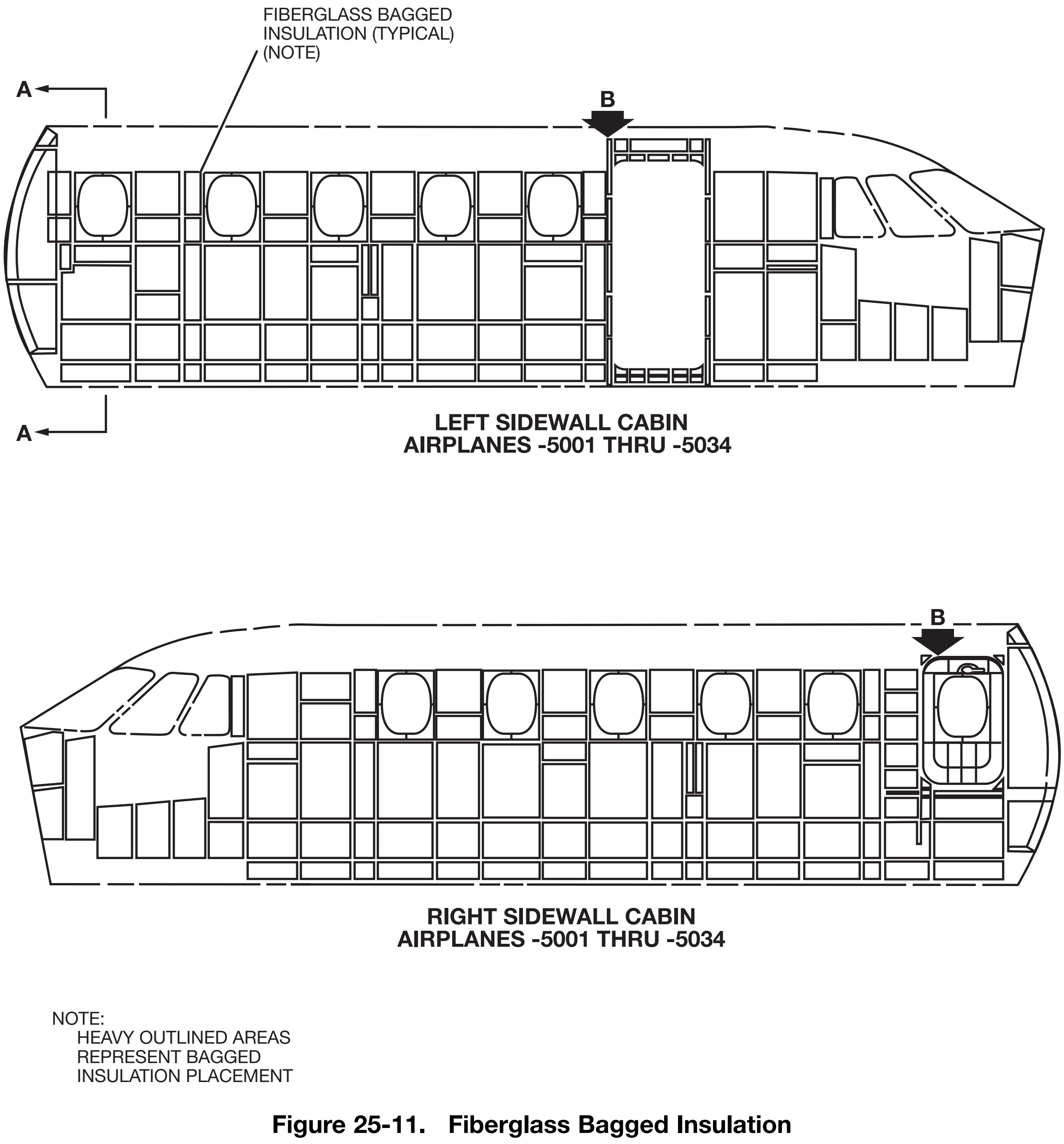


Figure 25-11. Fiberglass Bagged Insulation

LIFE VEST

A life vest is provided for each belted seat location. For standard forward and aft facing seats, the life vests are stored in the passenger seat base shroud assembly. There is also a life vest in a pocket on the back of the pilot and copilot seats. For all side-facing seats and two-place couches there is a life vest under the bottom seat cushion. The life vest for the belted aft carry-out flush toilet is in the aft bulkhead closet. A velcro fastener holds the vest in place.

Life vests are inflated by pulling on the handle, which discharges an air cylinder. The vest can also be inflated manually by blowing into an oral tube.

The life vest is to be stored in its pouch and removed only in an emergency, or for inspection. Instructions for life vest use are provided on a briefing card (for each passenger).

NOTE

The GA-12 Life Vest must be shipped to an approved inspection facility for recertification at intervals specified in Chapter 5.

WATER BARRIER

The water barrier is a short divider installed across the entrance door. It is used to prolong float time if ditching becomes necessary. The water barrier is stowed in the aft bulkhead closet and is secured by a retaining strap. Instructions for use are provided on a placard, and also on the passenger briefing card at each seat.

INSULATION

This section describes the insulation and acoustical dampening material in the aircraft.

The purpose of the insulation is to provide comfort for the passengers and flight crew during extreme changes in temperature. It also helps reduce the noise level. The insulations discussed are the fiberglass bag and nomex blanket-type.

DESCRIPTION

There is fiberglass bag insulation throughout the cabin, overhead and floor compartments. The fiberglass bags within the cabin/passenger area vary in thickness from one to three inches thick (depending on location). Type VIII adhesive secures fiberglass bags to the structure (Figure 25-11).

Nomex blanket insulation is on the back of the acoustic side panels and under the headliner, to provide additional sound dampening.

The baggage compartment is insulated with bagged insulation throughout.

CHAPTER 26
FIRE PROTECTION

CONTENTS

ILLUSTRATIONS

TABLE

CHAPTER 26
FIRE PROTECTION

INTRODUCTION

This chapter presents the fire protection system on the Citation 560XL/XLS/XLS+ aircraft. Included in this chapter is discussion of fire detection and fire-extinguishing systems, along with detailed discussion of the fire detection system control unit. Components and their operation are listed in addition to general maintenance considerations and functional and operational checks. References for this chapter and further specific information can be found in Chapters 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 26—"Fire Protection," of the *Aircraft Maintenance Manual (AMM)*.

GENERAL

Fire protection for the 560XL/XLS aircraft consists of a detection system and extinguishing system. Provisions for fire detection are installed in the left and right engine compartments and consist of a closed-loop sensing system and detector control unit that illuminates the respective red LH–RH ENGINE FIRE switchlights on the cockpit glareshield when a fire or overheat condition is present (Figure 26-1). The warning light, under a transparent, spring-loaded guard, also serves as a firewall shutoff switch. Fire annunciation on the XLS+ will also result in the MASTER WARNING flashing as well as the ENGINE FIRE L–R red CAS message and associated aural annunciation.

The fire-extinguishing system provided for the engine compartments actuates by lifting the guard and depressing the LH–RH ENGINE FIRE switchlights. This simultaneously closes the respective firewall fuel and hydraulic valves, deenergizes the starter-generator, and arms the two extinguishing bottles (Figure 26-2).

ENGINE FIRE L-R			
Color	Inhibited By		Debounce
Red			Standard
			1 Second

This message is displayed when the engine fire detection system has detected a fire.

Figure 26-1. XLS+ Engine Fire Indication

NOTES

XLS+

Figure 26-2. ENGINE FIRE Switchlights

XL/XLS—The firewall shutoff and extinguisher arming are indicated by illumination of the respective L–R LO FUEL PRESS, L–R LO HYD FLOW, F/W SHUTOFF, L–R GEN OFF annunciators, and both white BOTTLE 1–2 ARMED PUSH switchlights (Figure 26-3).

XLS+—The firewall shutoff and extinguisher arming are indicated by illumination of the respective ENGINE FAIL L–R red CAS message, DC GENERATOR FAIL L–R, HYDRAULIC FLOW LOW L–R, WINDSHIELD HEAT INOP L–R amber CAS messages, FIREWALL SHUTOFF L–R white CAS message, and both white BOTTLE 1–2 ARMED PUSH switchlights (Figure 26-3).

Once armed, either bottle can be discharged to the selected engine by pushing the BOTTLE 1 or BOTTLE 2 ARMED PUSH switchlight. The switchlight will extinguish when it is pushed. Both bottles can be directed to the same engine if necessary.

NOTES

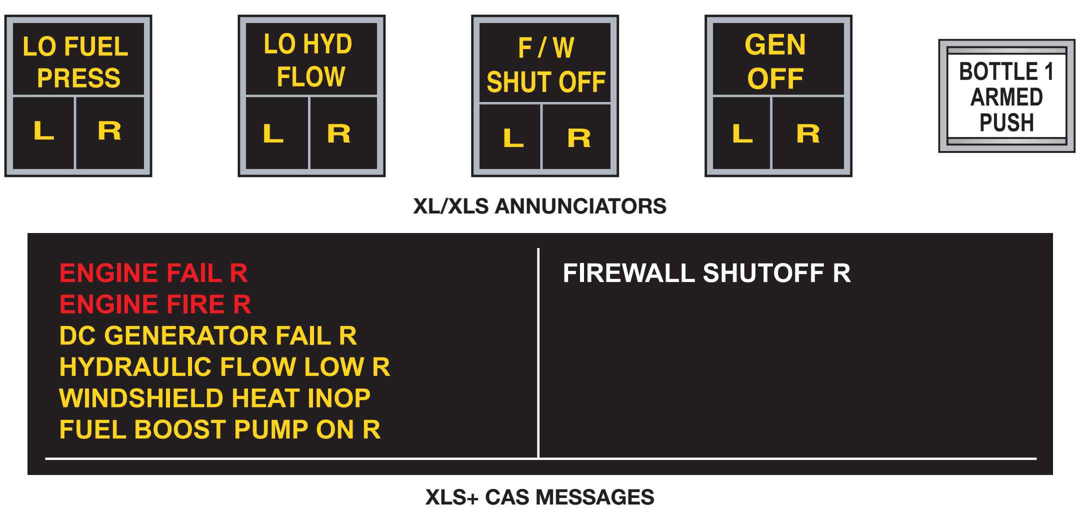


Figure 26-3. Engine Fire Detection Indications

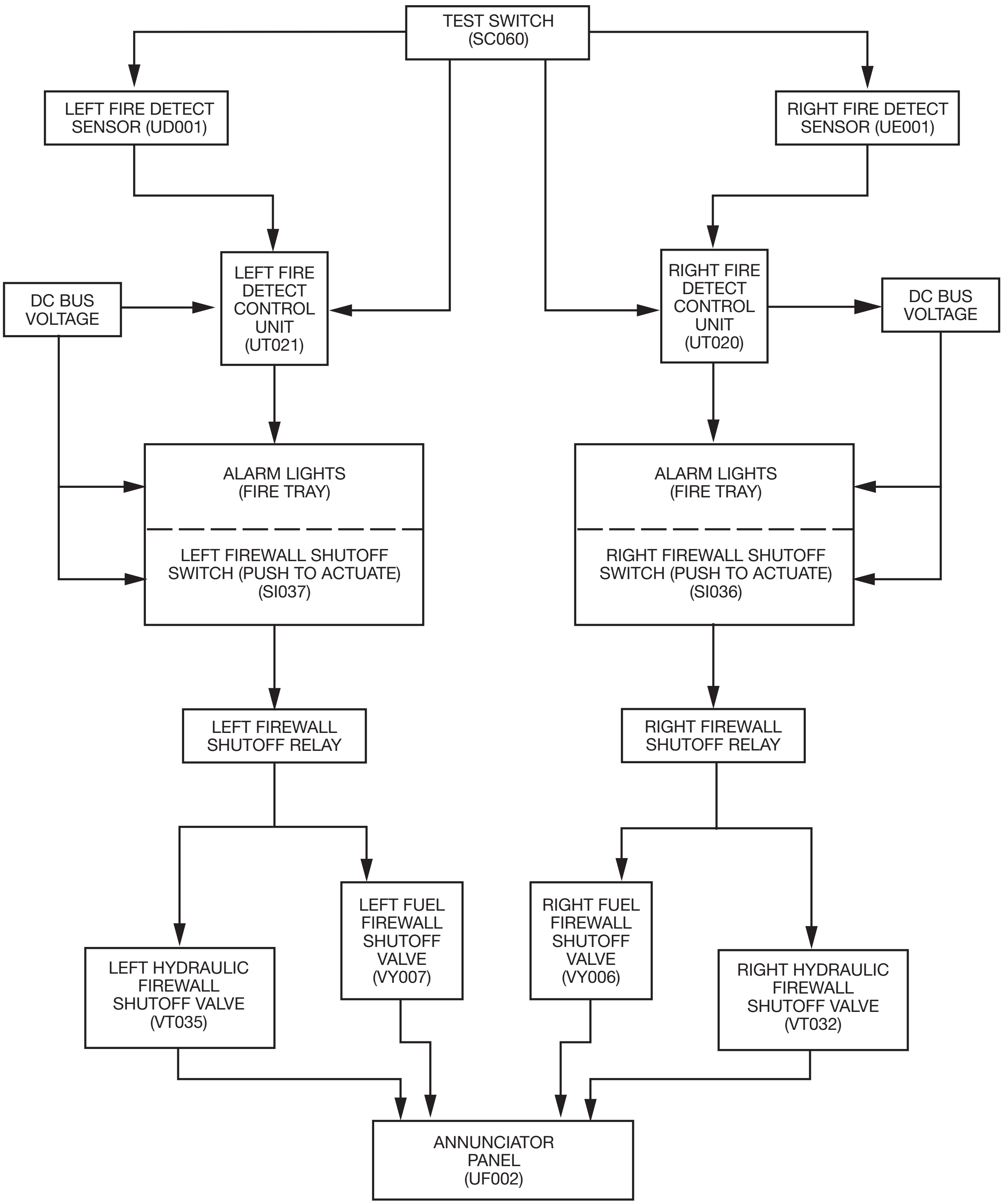


Figure 26-4. Fire Detection Block Diagram

26 FIRE PROTECTION

FIRE DETECTION

DESCRIPTION

This chapter provides maintenance information about the fire detection system, and the fire extinguishing system. The automatic fire detection system detects a fire and provides visual indication to the operator. This detection occurs in the nacelle (Figures 26-4 and 26-5).

The detection system components detect a fire or overheat condition through the following system. The detection system contains a sensing element in the nacelle, an electronic control in the tail cone, and visual indications on the fire tray, (attached to the glareshield).

The fire extinguishing system is a fixed system for the left and right engine compartments and portable hand fire extinguishers. The fixed system is used to extinguish fires in the engine compartments. The system consists of:

- Two fire extinguisher bottles (UT025 and UT026) that store the extinguishing agent
- Deployment tubes and nozzles
- Fire extinguisher discharge controls
- Associated electrical circuits.

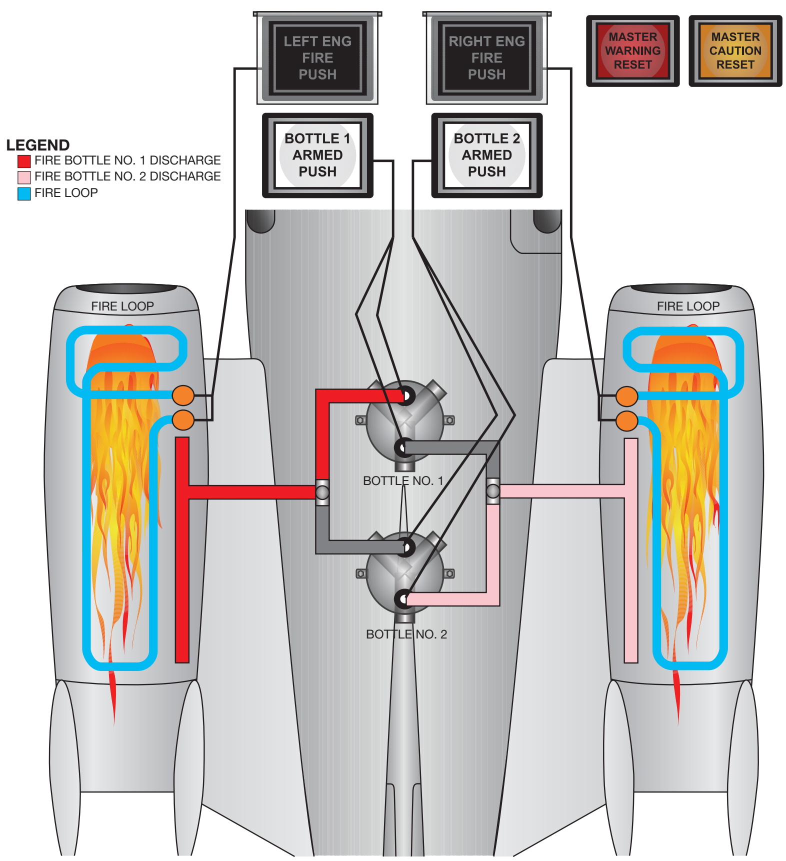


Figure 26-5. Engine Fire Extinguishing System

Figure 26-6. Fire Detection Sensor Cable Installation

COMPONENTS

Fire Detect Sensor Cable

A fire detect sensor cable (UD001 left/UE001 right) is installed on each engine (Figure 26-6). A flexible stainless steel cable is attached to the engine and tubing, using clamps and grommets. The cable clamps attach to the engine with existing engine component fasteners.

The temperature sensitive element of the sensor cable is a semiconductor coaxial cable that has a homogeneous mass. The cable is interconnected to form a closed loop and form one leg of a Wheatstone bridge. With increasing temperature, the resistance of the cable decreases. When the cable passes through null, sufficient current of the proper polarity actuates the null detector (transistor amplifier), which in turn operates a magnetic relay that actuates the fire warning indicator.

The fire detection system detects a fire or overheat condition in the left or right engine compartment. A fire warning indicator (light) alerts the operator when either condition exists.

The aircraft is equipped with a detection system in each engine compartment. The installations are typical. Therefore, the description, operation, troubleshooting and maintenance practices apply to both installations.

Detection Sensor

The sensing cable is a 215.0 inch (5.46 m) flexible stainless steel tube that contains a single wire centered in a highly compacted semi conductor material. The semiconductor material holds the single wire centered, as the cable is bent and looped during installation. The cable is hermetically sealed and has a fireproof connector at each end.

CAUTION

Extreme care must be exercised during maintenance not to twist, kink or dent the sensing loop element.

NOTES

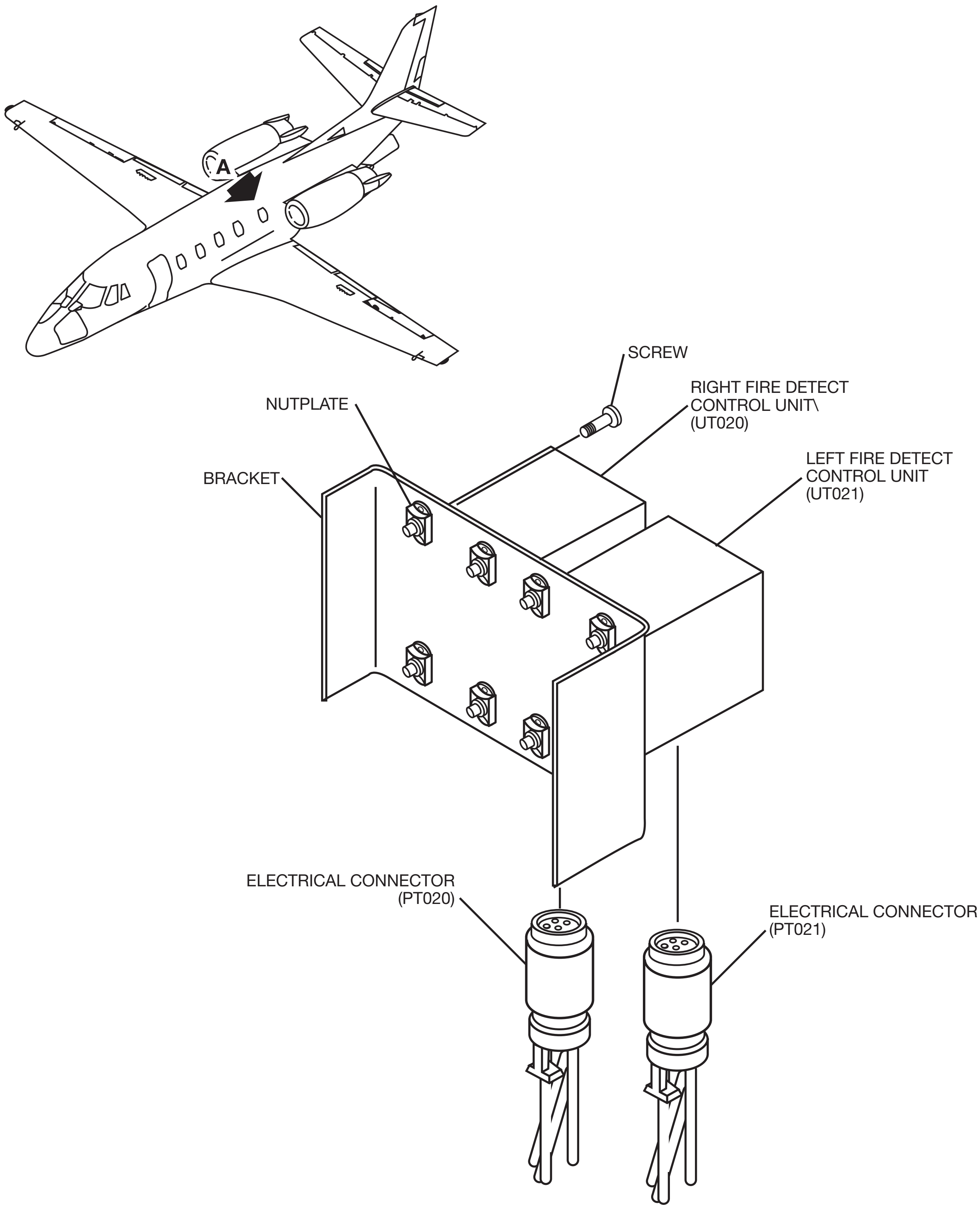


Figure 26-7. Fire Detection Control Unit

Control Unit

There are two fire detection control units (UT021 left/UT020 right) in the tail cone on the aft side of the aft baggage compartment wall, accessible through the forward tail cone access door (Figure 26-7).

Control Assembly

The control assembly incorporates two Wheatstone bridge circuits. The fire warning bridge detects a fire or overheat condition in the engine nacelle. The short discriminating bridge detects a short in the sensor cable. The two bridge circuits share some resistors in the control assembly and they share the sensor cable.

In normal operation, the sensor cable resistance decreases as its temperature rises. As the sensor cable is heated, its resistance falls below the fire alarm point (200 ohms). The control assembly disables the short discriminating lockout circuit by disconnecting its output. It simultaneously energizes the fire warning indicator. If the cable resistance continues to fall, the electronic short discriminating circuit operates, but has no effect on the alarm (since output has been disconnected).

If a fire detection sensor cable center conductor short circuit to ground occurs and the apparent cable resistance falls through the fire (and short discriminating points at the same time) electronic lockout will occur. This disables the fire relay and fire alarm. The fire relay circuit delays deliberately, to provide this lockout feature (for continuous or intermittent response of the cable). Therefore, it does not interfere with normal operation.

The basic control discriminates between a true fire and a short circuit when the control recognizes the manner in which the sensor cable resistance falls. An instantaneous change in cable sensor resistance (to a value below the short discriminator alarm resistance) is rejected as a fire, but is interpreted as a short.

System integrity is verified through use of an internal test resistor. The rotary TEST knob (SC060), when actuated, opens the sensor center wire loop and applies the test resistor to the open end of the sensor cable. The sum of the test resistance/conductor resistance is lower than the fire alarm point. Therefore, it operationally tests sensor cable continuity, the internal circuitry, and the fire warning indicators. The short discriminator alarm point resistance is lower than the sum of the test resistor and sensor cable resistance. Therefore, the short discriminating/disabling circuit is not actuated or tested when the rotary TEST switch is actuated. Because of the disabling action of the short discriminating circuit on the fire relay, a system verification test cannot be accomplished when a short is present.

If the fire detection system fails the system integrity test, an amber FIRE DET SYS L–R cautionary annunciator (XL/XLS) or ENG FIRE DETECT FAIL L–R CAS message (XLS+) illuminates (Figure 26-8).

L/R FIRE DET SYS
Annunciator flashes if the engine fire detect system fails, activates MASTER CAUTION lights. Engine fire extinguishing system remains operational.

XL/XLS ANNUNCIATOR

ENG FIRE DETECT FAIL L-R			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is posted when one of the engine fire detectors has failed. When a failure is detected, the fire detection controller sends a ground to the EICAS system, which displays the message. When the system is operating normally, the controller sends an open, which causes the EICAS to remove the message.

XLS+ CAS MESSAGE

Figure 26-8. Fire Detect Fail Indications

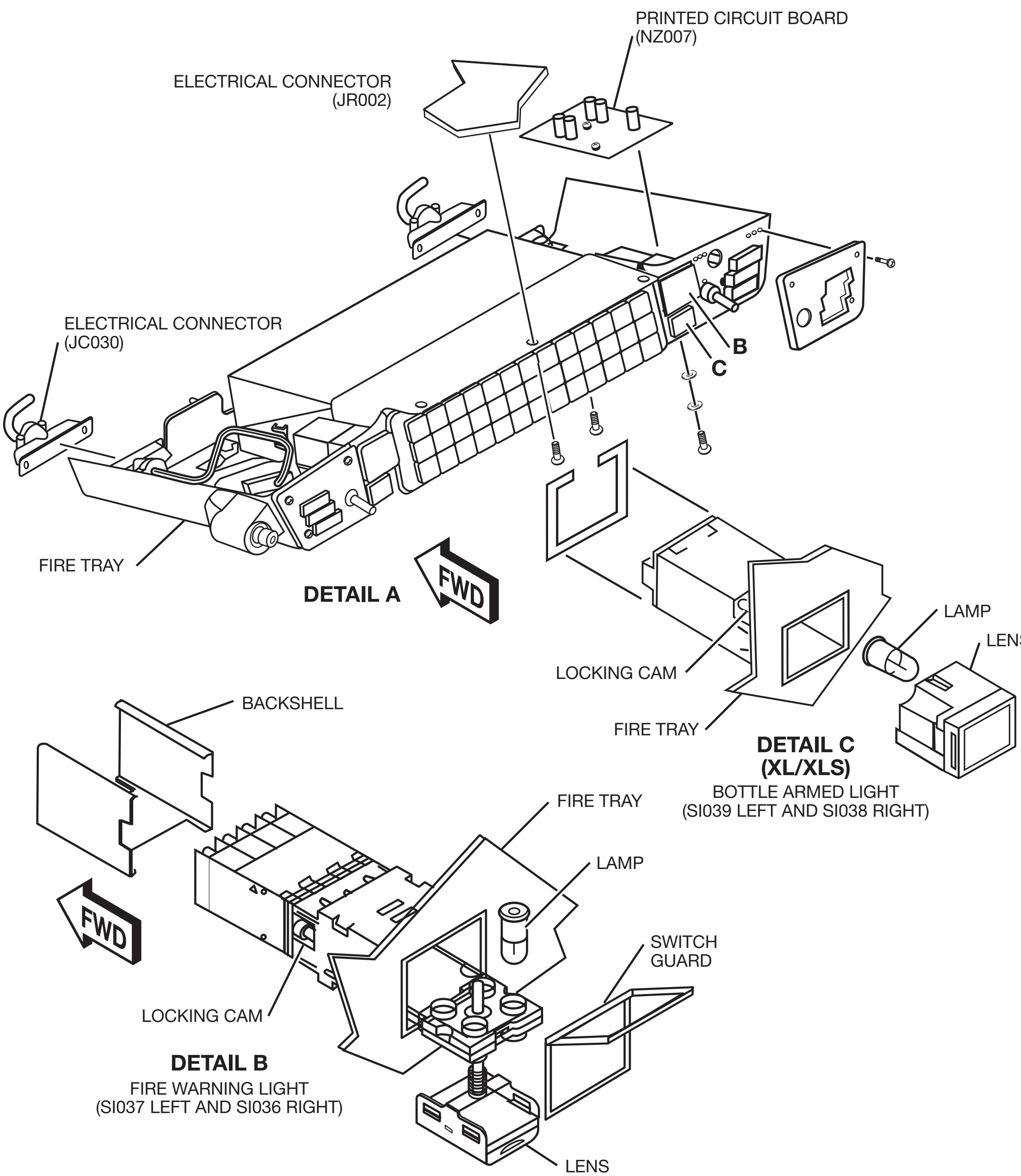


Figure 26-9. Indicating Lights Installation

CONTROLS AND INDICATIONS

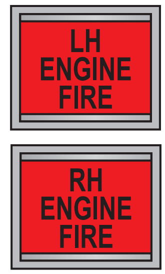
Indicating Lights

Indicating switchlights warn the pilot if there is a fire in an engine compartment; and indicate when the fire extinguishing bottles are armed (Figure 26-9).

The indicator switchlight assemblies have dual functions. The fire warning light assembly contains a switch which arms the fire extinguisher switch, and closes the firewall fuel shutoff (VY007 left/VY006 right) and hydraulic shutoff (VT035 left/V032 right) valves. The bottle-armed light assembly contains a switch that discharges the fire extinguishing container.

The right and left indicator switchlights are identical. Removal procedures are typical for right and left light assemblies.

Lamp replacement can be accomplished without removing light assemblies (Figure 26-10).



ENGINE FIRE
Illumination indicated high temperature is detected in the engine necelle.
Pressing the switchlight:
1. Closes the field F/W shutoff valve.
2. Closes the hydraulic F/W shutoff valve.
3. Deactivates the engine generator (open the field relay).
4. Disarms the thrust reverser.
5. Arms the engine fire bottles.

XL/XLS

BOTTLE ARMED 1/2
Illumination of the white light indicates the repective engine fire bottle is armed. When pressed, the bottle discharges. The red ENGINE FIRE switchlight must be pressed to illuminate the BOTTLE ARMED lights.

XL/XLS/XLS+

ENGINE FIRE L-R			
Color	Inhibited By		Debounce
Red			Standard
			1 Second
This message is displayed when the engine fire detection system has detected a fire.			

XLS+ CAS MESSAGE

Figure 26-10. Warning Indications

DIAGNOSTICS

The control assembly (UT021 left/UT020 right) and the fire detection sensor cable are hermetically sealed, and do not require adjustment. This section provides applicable continuity and resistance checks which may be performed to verify system integrity. A system self-test and cleaning instructions (for the fire detection sensor cable) are also included.

Fire Warning Sensor Cable

1. Inspect the center pins and contacts of each cable to see that the pins are centered properly in the cable terminations. Make sure that no foreign material or contamination exists in the recesses surrounding the pins or contacts.
2. Inspect the sensor cable for proper mounting. Adjust the mounting clamps (as required) to prevent the cable assembly from striking or chafing the adjacent structure.
3. Inspect for evidence of engine bleed air leaking onto the sensor cable.
4. Visually inspect the sensor cable for cleanliness, nicks and abrasions.

Control Assembly

1. Inspect the control unit (UT021 left /UT020 right) for security in installation.
2. Check the connector (PT021 left and PT020 right) for damaged pins and foreign material.
3. Inspect the control unit for any evidence of damage.

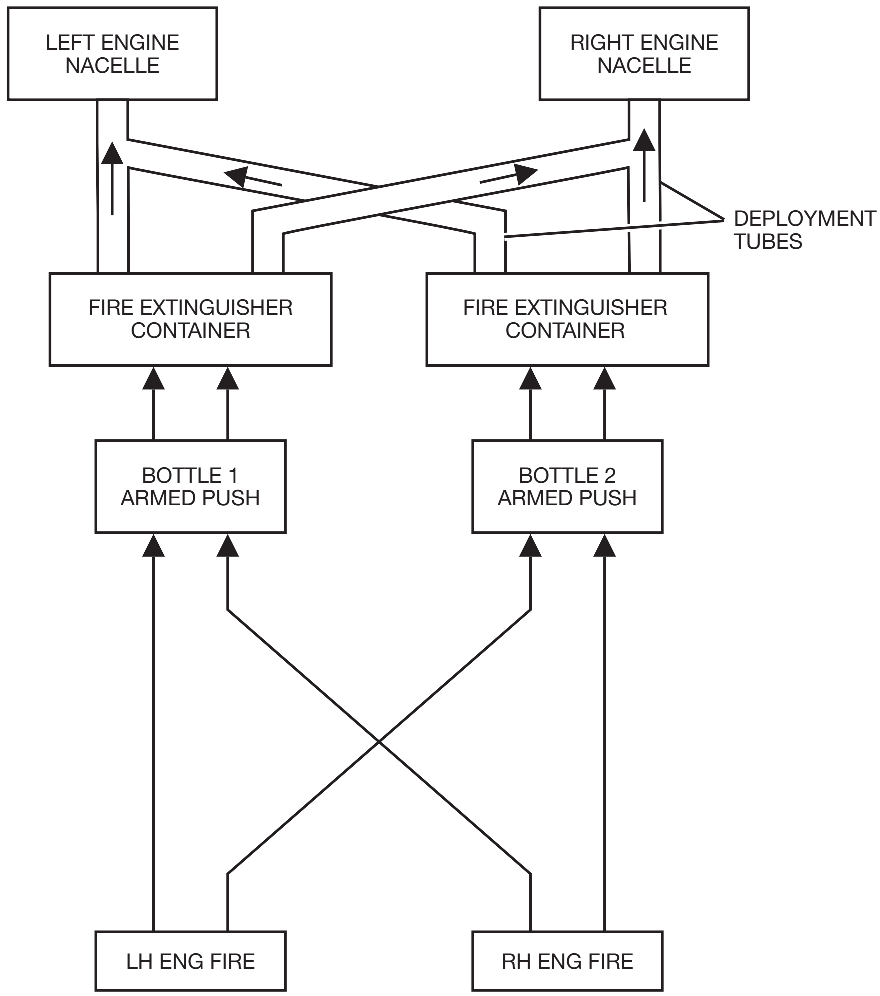


Figure 26-11. Fire Extinguishing Block Diagram

Testing Procedures

Sensor Cable Continuity Check

Connect an ohmmeter from the center pin on one end of the cable to the center pin on the other end to measure the resistance of the center conductor (of the sensor cable). The resistance must not exceed 0.6 ohms per foot of cable length, or 10.75 ohms for total sensor cable length.

Sensor Cable Insulation Resistance Test

The insulation resistance at room temperature depends on the temperature characteristic of the sensor cable. Use a Megohmmeter capable of supplying a test voltage of 100 volts.

CAUTION

If the sensor cable is installed, disconnect sensor cable at both connectors. As a precaution, a fire extinguisher must be in the immediate vicinity.

Measure the direct current resistance from cable center termination to outer sheath, (center termination negative). Read the instrument within five seconds of the first application of voltage. Note sudden or momentary shifts in reading indicative of breakdown.

The sensor cable is acceptable if there is no indication of breakdown, and if the insulation resistance is not less than the pertinent value tabulated in the *AMM*.

If the sensor cable insulation resistance does not meet minimum requirements, and there is no apparent physical damage, clean the sensor cable end fittings in accordance with (Fire Detection Sensor Cable Cleaning Procedure).

Sensor Cable Resistance

Table 26-1 refers to samples of sensor cable resistance.vv

Table 26-1. SAMPLE COPY OF SENSOR CABLE RESISTANCE

Resistance in Megohms For Total Length of Cable	Ambient Temp °F
0.3255	68
0.2680	72
0.2234	76
0.1851	80

FIRE EXTINGUISHING SYSTEM

DESCRIPTION

This section provides maintenance information on fire extinguishing. Nacelle fire extinguishing is the main subject covered. Portable fire extinguishers, associated with the cabin interior, are also described in this section.

The fire extinguishing system consists of a fixed fire extinguishing system (for the left and right engine compartments) and portable hand fire extinguishers (Figure 26-11). The fixed system is used to extinguish fires in the engine compartments.

The system includes:

- Two fire extinguisher bottles (UT025 and UT026)
- Storing extinguishing agent
- Deployment tubes and nozzles
- Fire extinguisher discharge controls
- Associated electrical circuits

The fire extinguisher bottles incorporate fill and pressure relief valves, temperature compensating pressure switches and explosive cartridge operated discharge valves. In addition, a baffle attached to the engine fan duct assembly is an integral part of the system and prevents the fire extinguishing agent from escaping from the aft end of the engine compartment.

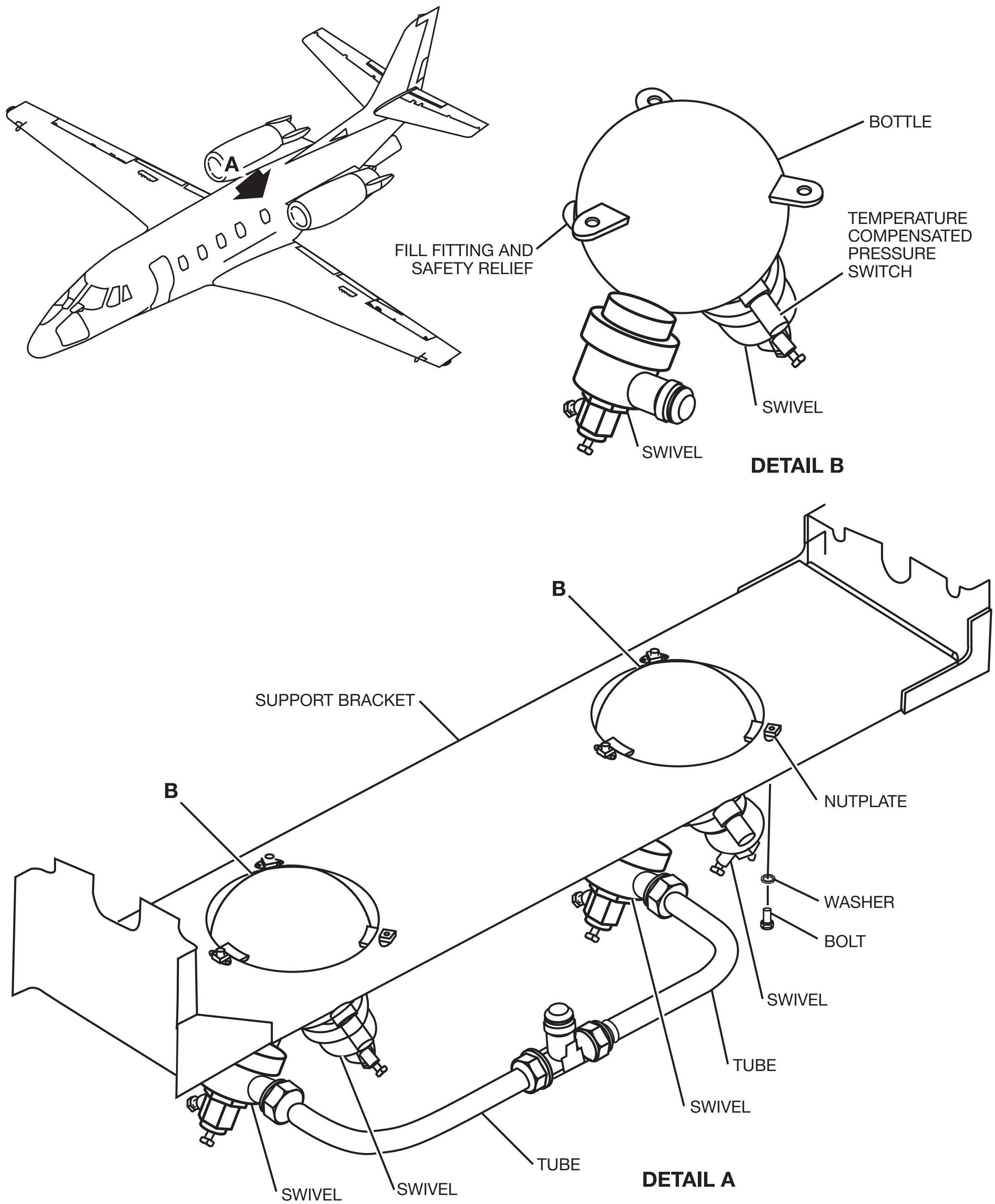


Figure 26-12. Fire Extinguisher Bottles Installation

COMPONENTS

Fire Extinguishing Bottles

Two fixed fire extinguishing bottles are installed in the tail cone (Figure 26-12). Each bottle has an extinguishing agent (deployment tube system) that supplies the extinguishing agent to the left engine or right engine. The bottles store the extinguishing agent under pressure until released by fire extinguishing discharge action. Each bottle provides one extinguishing shot. The bottles are identical and consist of:

- 86-cubic inch spherical steel container with a temperature compensated pressure switch
- Combined safety outlet and fill port and two discharge valves and outlets

Either one or both bottles may be fired into the left or the right nacelle.

The extinguisher bottle is a vessel for containing fire-extinguishing agent (monobromo-trifluoromethane). The bottles are super-pressurized at room temperature with dry nitrogen. A pyrotechnic cartridge in the discharge valve actuates the extinguisher.

The temperature compensating pressure switch indicates a decrease in container pressure. When container pressure drops below 500 ± 30 psig at 70°F (21°C) the switch closes and the FIRE EXT BOTTLE LOW light illuminates on the annunciator panel (XL/XLS) or ENG FIRE BOTTLE LOW 1–2 CAS message (XLS+) displays (Figure 26-13).

The extinguisher utilizes a combination fill fitting and safety relief assembly. If the ambient temperature rises abnormally, a fusible check valve melts within the fill fitting. This relieves the contents of the container through the fill fitting.

The extinguishing agent is non-corrosive and has no damaging effects on engine compartment components. No engine components require replacement as a result of the extinguishing agent entering the nacelle.

FIRE EXT BOTL LOW
Annunciator flashes if one or both engine fire extinguisher bottles have low pressure. Activates MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

ENG FIRE BOTTLE LOW 1-2			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when an engine fire bottle is low, as measured by a pressure switch on the bottle. When the bottle is low, it sends a ground signal to the EICAS system, which posts the message. When the bottle is filled, it sends an open signal which removes the message.

XLS+ CAS MESSAGE

Figure 26-13. Fire Bottle Indications

Controls for releasing the extinguishing agent are in the fire tray attached to the glareshield.

Fire detection indicators illuminate, alerting the operator of the condition in the nacelle. Extinguishing controls also illuminate to alert the operator, who releases the extinguishing agent.

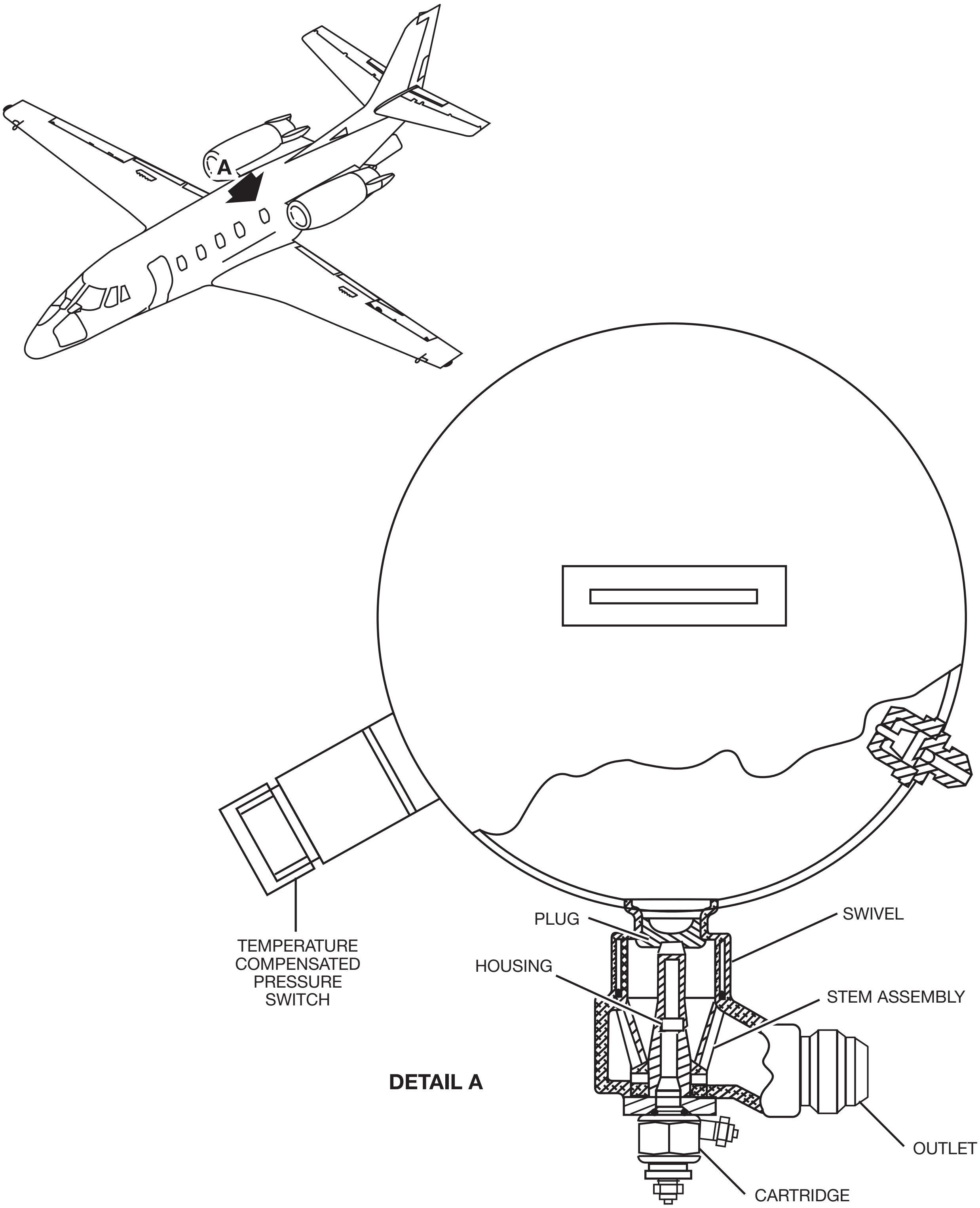


Figure 26-14. Fire Extinguisher Bottle Components

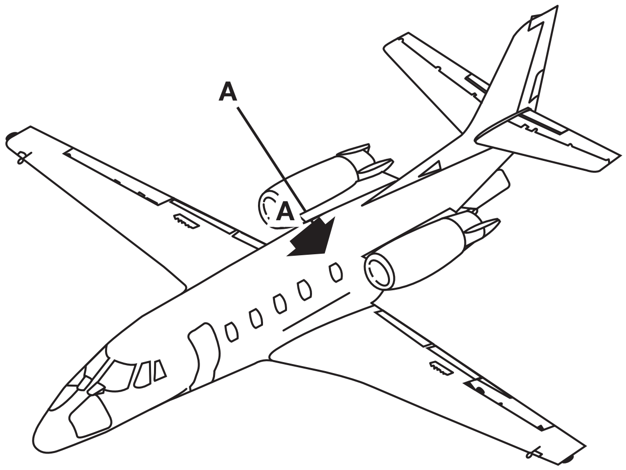
Fire Extinguisher Explosive Cartridges

The fire extinguisher explosive cartridges are electrically fired and provide a means for controlling the release of the fire extinguishing agent (Figure 26-14). When actuated, the cartridge produces high pressure that ruptures the housing assembly, removing the restraining force from the valve plug. The pressurized agent unseats the plug, releasing the agent through the deployment tubes, to the engine compartment. The plug and housing assembly parts collect in a strainer basket.

CAUTION

Do not over torque the terminal screws on the fire bottles. Over tightening of the screws will cause the housing to break.

NOTES



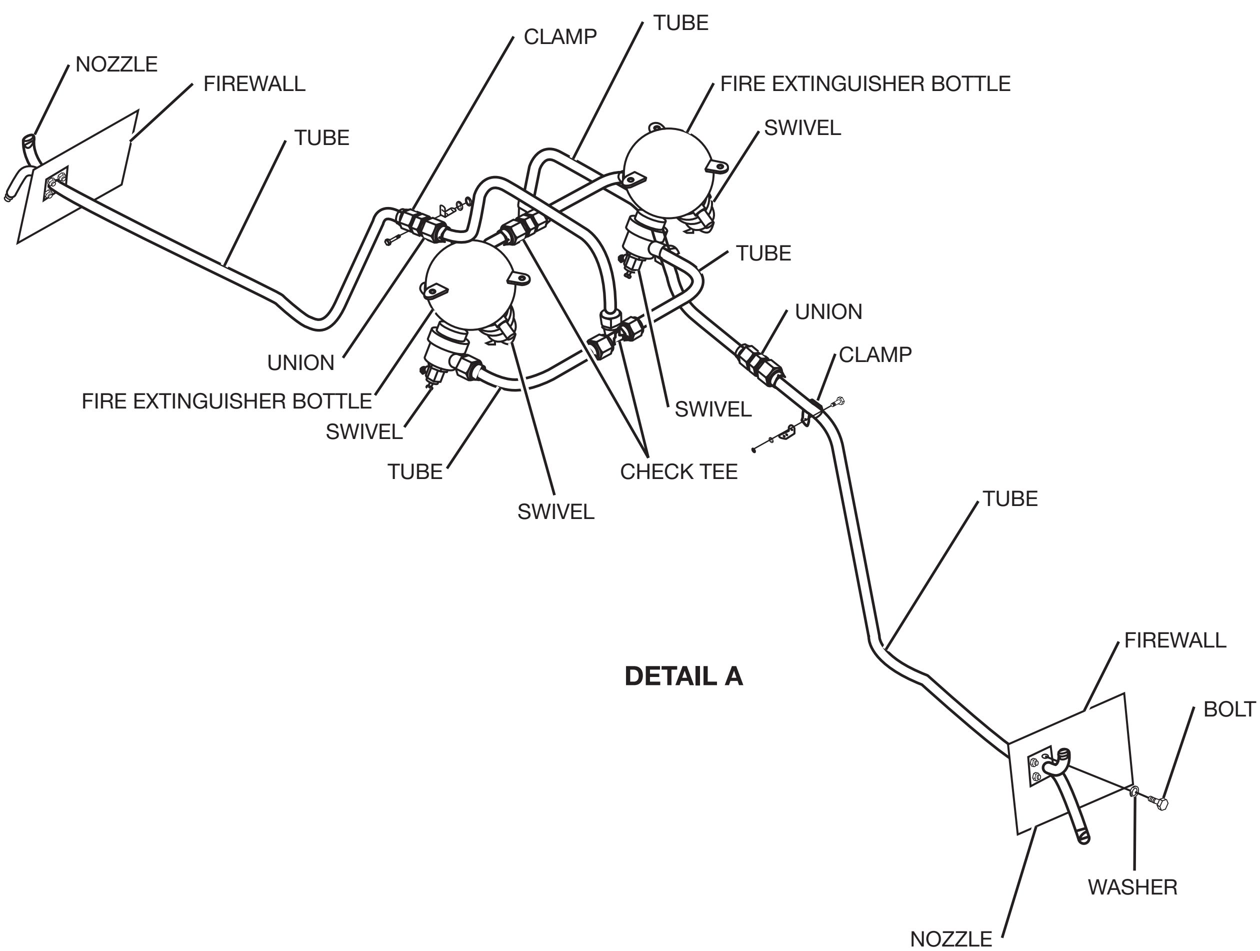


Figure 26-15. Fire Extinguisher Deployment Tubes

Fire Extinguisher Deployment Tubes

Fire extinguisher deployment tubes disperse the extinguishing agent from the bottles to the selected fire area. Each left and right engine compartment area is served by an individual deployment tube system (Figure 26-15).

NOTES

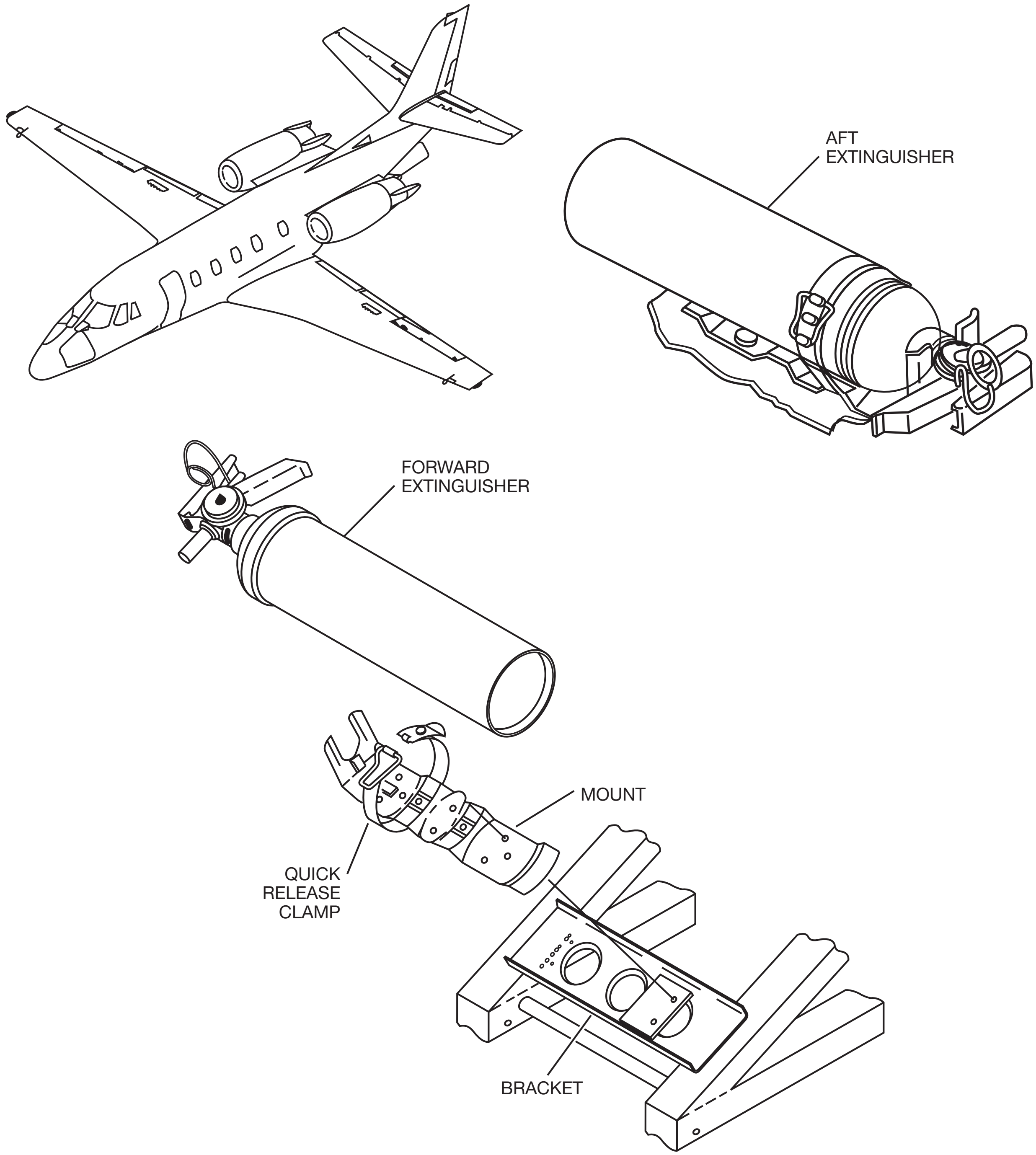


Figure 26-16. Portable Hand Fire Extinguisher

Portable Extinguishing

Two portable hand fire extinguishers are installed in the aircraft. One extinguisher is in the cockpit on the copilot seat (accessible by either the pilot or copilot). The other extinguisher is in the cabin accessible to all the passengers (Figure 26-16).

Both extinguishers are in quick-release mounting brackets (painted red). The bracket assembly is attached by a screw.

The cabin extinguisher is in a bracket on the floor just forward of the divider at FS 342.75.

The portable extinguishers are pressurized bottles containing a Halon Type 1211 extinguishing agent. The bottles have an actuating valve, operated by hand. The extinguishers are rated for Class B and C fires and may be recharged at any locally approved fire equipment service shop.

NOTE

After use, the extinguisher must be charged immediately with Halon 1211. Extinguishers should only be replaced with an identical extinguisher.

To service the extinguishers check the gauge to verify that normal pressure is maintained, and recharge the extinguishers after use (or on the expiration date).

CONTROLS AND INDICATIONS

Fire Extinguishing Discharge Controls

Crew can select and discharge from either fire extinguisher container to either engine compartment using the fire extinguishing controls. The number 1 and number 2 engine fire extinguisher switchlights are on the fire tray. To initiate discharge for either engine compartment after a LH–RH ENGINE FIRE switchlight illuminates, raise the guard over the respective LH or RH ENGINE FIRE switch-light and press the switch light. This action closes the fuel firewall shutoff valve, the hydraulic firewall shutoff valve, illuminate the BOTTLE 1 ARMED PUSH and BOTTLE 2 ARMED PUSH switchlights and provide electrical power to the BOTTLE 1 ARMED PUSH and BOTTLE 2 ARMED PUSH switch-lights.

Upon pressing either BOTTLE 1 or BOTTLE 2 ARMED PUSH switchlight, a voltage of 28 VDC applies to the cartridge that corresponds to the switch. The resulting explosive pressure breaks the end of the housing assembly, removing the mechanical locking force against the valve plug. The fire extinguishing agent discharges through the swivel into the distri-bution network. Once the extinguishing container has been discharged, the respective switchlight extinguishes.

If the fire Warning light stays on, indicating fire is still present, the remaining fire extinguishing switch may be actuated, releasing the fire extinguishing agent from the other extinguishing container to the same fire area.

WARNING

To prevent accidental discharge, make sure all circuits are isolated from the bottle explosive cartridges, when operating fire extinguisher discharge switches for troubleshooting.

DIAGNOSTICS

Bottle Wiring Check

Normal maintenance requires periodic inspec-tion of the Fire Bottle and the Aircraft wiring system.

Refer to the *AMM* for the exact procedures.

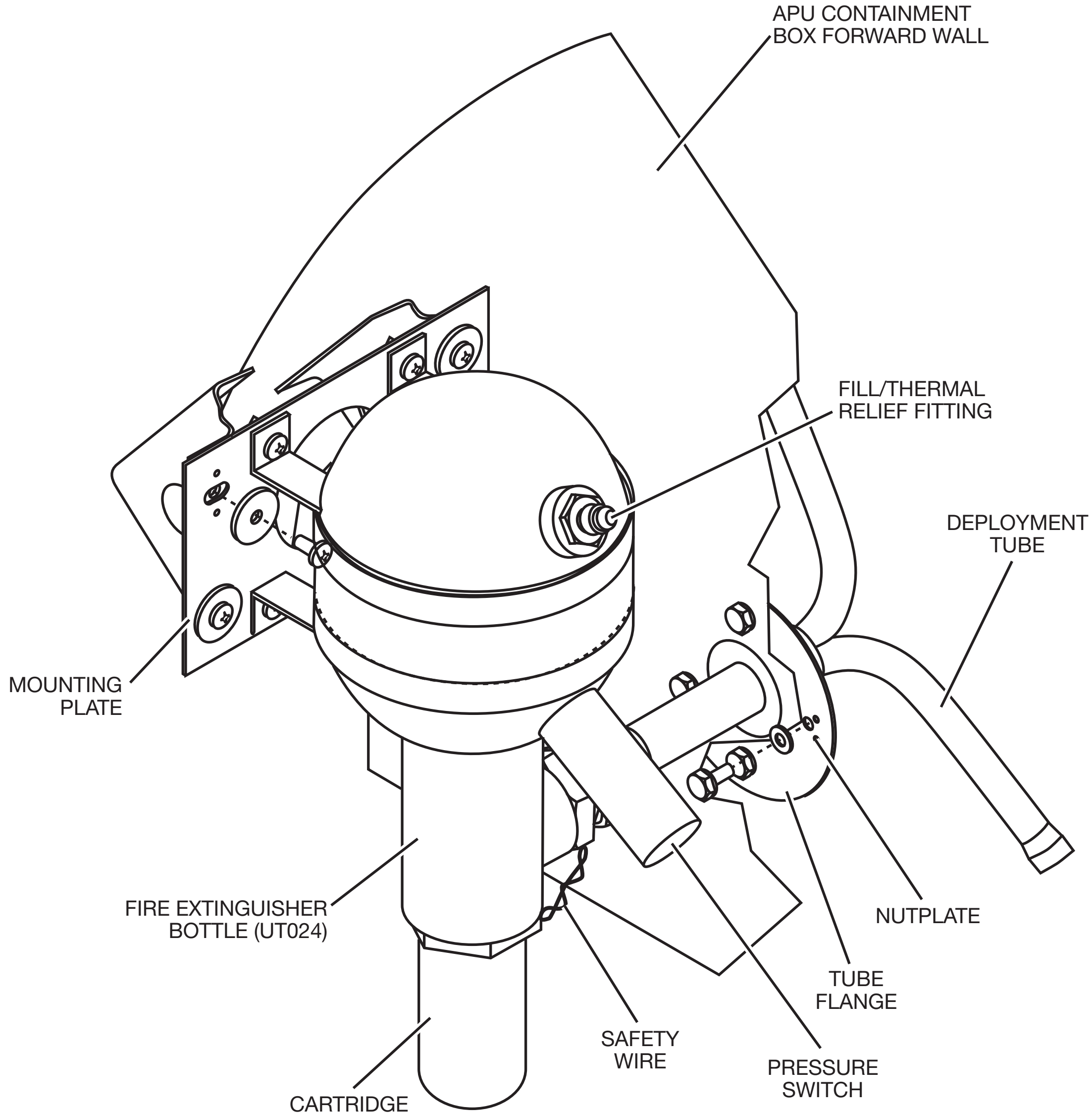


Figure 26-17. APU Fire Bottle

APU FIRE DETECTION

DESCRIPTION

The fire detector assembly is routed around the APU powerplant at strategic points and includes an integral sensor element and a responder unit. The integral sensor element is constructed of stainless steel. The center core is charged with and retains a fixed volume of inert gas. An increase in temperature on any area of the sensor element, (which is routed around the APU) causes the inert gas to expand. The expansion of the gas actuates a switch in the responder unit. The responder unit incorporates two pressure switches that are permanently joined to a common sensor. The switches function as an alarm and integrity responder. When a fire is detected, this responder unit supplies 28 VDC to the ECU and to the APU monitor PC board on pin 22. With this input on pin 22, the APU monitor PC board supplies power out on pin 13 for APU FIRE switchlight illumination.

APU FIRE EXTINGUISHING

DESCRIPTION

The APU is completely enclosed in a fire containment box made of titanium and stainless steel. Access to the APU is gained through a door on the right side of the fuselage. The fire extinguishing system deploys extinguishing agent from a single fire extinguisher bottle into the APU fire containment box, in the event a fire is detected by the associated fire detection system. This bottle is below the firewall fairing and dispenses extinguishing agent via a single deployment tube. The deployment tube is routed through the firewall fairing and terminates at a “T” fitting, which disperses the fire-extinguishing agent within the APU enclosure.

COMPONENTS

Fire Bottle

The fire bottle assembly consists of a steel cylinder, fill/thermal relief port, aluminum discharge outlet, and pressure switch (Figure 26-17). The bottle contains 1.0 pound (0.45kg) of Halon 1301 which is pressurized by dry nitrogen at 600 +25 or –0 psig.

FILL/THERMAL RELIEF—The fill/thermal relief port is on the upper portion of the bottle. This port also incorporates a thermal relief valve which ruptures if internal bottle temperature exceeds between 205°F to 226°F at a pressure of between 1520 and 1710 psi.

NOTES

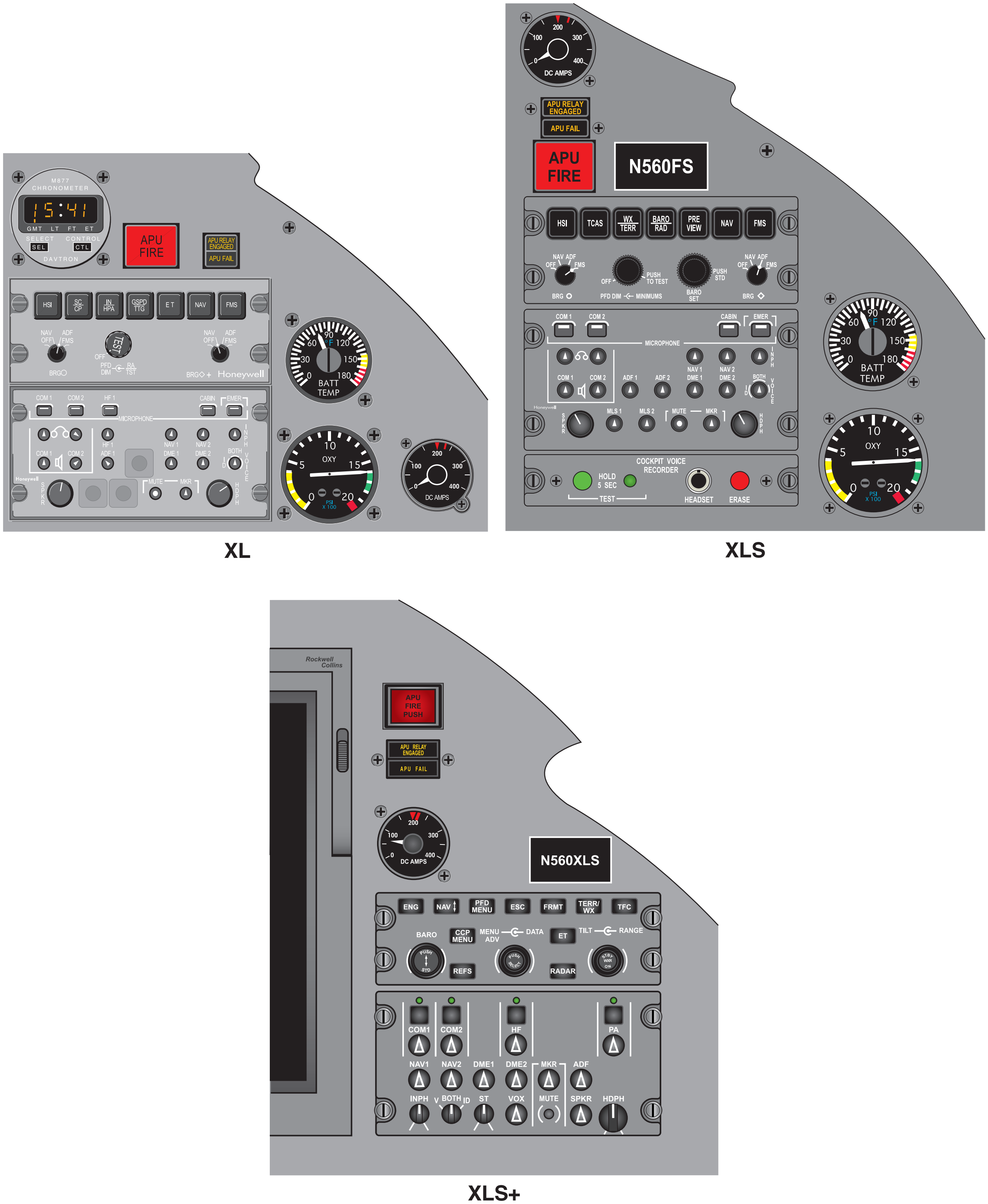


Figure 26-18. APU Controls and Indications

DISCHARGE OUTLET—The discharge outlet is on the bottom of the bottle and contains a pyrotechnic device, which is fired manually by the cockpit APU FIRE switchlight or automatically 8 seconds after a fire is detected (Figure 26-18). The XLS+ also displays an APU FIRE CAS message with associated aural warning (Figure 26-19). A 28VDC signal is sent to the pyrotechnic device when either the APU FIRE switchlight is depressed or by the APU monitor PC board 8 seconds after it receives an input from the fire detection system. The resulting explosion ruptures a diaphragm inside the discharge outlet. This rupture allows rapid expulsion of the pressurized Halon through the discharge outlet and into the discharge tube.

APU FIRE
Illumination indicates high temperature in the APU compartment. The APU automatically shuts down and the APU FAIL light illuminates. Pressing the red switchlight discharges the APU fire bottle. If the switchlight is not pressed, the fire bottle automatically discharges in 8 seconds.

XL/XLS/XLS+ ANNUNCIATOR

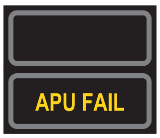
APU FIRE			
Color	**Inhibited By**		**Debounce**
Red	LOPI	TOPI	Standard
This message is displayed when a fire is detected in the APU by a fire loop. 28 Volts on the input to EICAS means a fire has been detected, which causes the message to be displayed. Open circuit means a fire has not been detected, which causes the message to be removed. A voice aural is also triggered with this message.			

XLS+ CAS MESSAGE

Figure 26-19. APU Fire Indications

NOTES

PRESSURE SWITCH—There is a pressure switch on the lower portion of the bottle. This switch is wired into the APU monitor PC board and supplies a signal to the PCB when bottle pressure drops below 500 ± 30 psig at 70°F causing the APU FAIL annunciator (XL/XLS) illuminate (Figure 26-19). The XLS+ also incorporates a white APU FIRE BOTTLE LOW advisory CAS message (Figure 26-20).

	**APU FAIL** Illumination indicated the APU will not start due to a system malfunction (i.e., the APU fire bottle is low or the fire detection system is inoperative). If the APU is operating, the light indicates the APU is shutting down. Reasons for automatic shutdown include fire detected in the APU compartment of the fire bottle is low. Limitation: Stating the APU is prohibited whenever the APU FAIL light is illuminated.

XL/XLS/XLS+ ANNUNCIATOR

APU FIRE BOTTLE LOW			
Color	**Inhibited By**		**Debounce**
White	LOPI	TOPI	Standard
This message is displayed when the APU fire bottle is low, as measured by a pressure switch on the bottle. When the bottle is low, it sends a ground signal to the EICAS system, which posts the message. When the bottle is filled, it sends an open signal which removes the message. The APU FAIL message will be display with this message.			

XLS+ CAS MESSAGE

Figure 26-20. APU Fire Bottle Indications

WARNING

The fire extinguisher bottle cartridge is a pyrotechnic device. Inadvertent detonation can cause personal injury. Always remove electrical power from the airplane, disconnect electrical connector from the cartridge and immediately install shunt plug/wire over cartridge electrical connector pins prior to removing/handling the fire bottle. Also avoid maintaining the fire extinguisher bottle near active radio broadcasting equipment, radar equipment, high voltage lines or during electrical storms.

WARNING

The fire extinguisher bottle discharges 8 seconds after receiving a signal from the fire detection loop.

Diagnostics

The following is verified by depressing the TEST button on the APU control panel:

- The integrity of the entire fire detector assembly
- The condition of the sensor
- Fire extinguisher bottle for adequate extinguishing agent/pressure

Activation of the test circuit illuminates the APU FIRE switchlight.

NOTES

QUESTIONS

1. Depressing an illuminated red ENG FIRE PUSH switchlight:
   A. Fires bottle No. 1 into the nacelle
   B. Fires bottle No. 2 into the nacelle
   C. Fires both bottles into the nacelle
   D. Illuminates both white BOTTLE ARMED PUSH switchlights, arming the bottles

2. After a bottle has been discharged into a nacelle:
   A. No cleaning of the engine and nacelle area is required
   B. A thorough cleaning of the engine and nacelle area is required
   C. An inspection of the engine and nacelle area is required to determine if cleaning is necessary
   D. None of the above

3. When the fire-extinguishing system is armed for operation (red ENG FIRE PUSH switchlight depressed):
   A. The amber FUEL PRESSURE LOW L or R CAS message flashes
   B. The amber HYDRAULIC FLOW LOW L or R CAS message flashes
   C. The amber DC GENERATOR OFF L or R CAS message flashes
   D. All the above

4. If the contents of an armed bottle has been discharged into a nacelle and the red ENG FIRE PUSH switchlight remains on:
   A. The fire has been extinguished.
   B. The other bottle can be discharged into the same nacelle by depressing the other white BOTTLE ARMED PUSH switchlight.
   C. The fire still exists, but no further action can be taken.
   D. The same white BOTTLE ARMED PUSH switchlight can be depressed again, firing a second charge of agent from the same bottle.

5. Depressing the red ENG FIRE PUSH switchlight a second time:
   A. Opens the fuel shutoff valve
   B. Opens the hydraulic shutoff valve
   C. Resets the generator field relay
   D. Both A and B

6. If the amber ENG FIRE DETECT FAIL L/R CAS message displays:
   A. Fire detection system is working properly
   B. Fire detection system is inoperative
   C. Has no effect on the fire extinguishing system
   D. Both B and C

7. If, during flight, EMER on the NORM/EMER switchlight is selected:
   A. Fire detection and extinguishing system is inoperative
   B. There is no effect on the fire system
   C. Fire detection portion of the system is still operable
   D. Fire extinguishing portion of the system is still operable

8. An ENG FIRE switchlight illuminates:
   A. When it is depressed
   B. MASTER WARNING switchlights also illuminate
   C. When temperature in the nacelle area reaches approximately 500°F (XL) or 450°F (XLS)
   D. Electrical resistance of the sensing loop increases due to increasing nacelle temperature

9. Illumination of the FIRE EXT BTL LOW annunciator indicates:
   A. Both fire bottles are low on pressure
   B. Fire warning system is inoperative
   C. Fire detection system is inoperative
   D. Either or both fire bottles have low pressure

10. During rotary test of the fire warning system (XL/XLS):
    A. Both fire warning lights illuminate and the MASTER WARNING switchlights flash
    B. Amber FIRE DET SYS annunciator illuminates
    C. MASTER CAUTION switchlights illuminate
    D. Both ENG FIRE switchlights illuminate

NOTES

CHAPTER 27
FLIGHT CONTROLS

CONTENTS

ILLUSTRATIONS

CHAPTER 27
FLIGHT CONTROLS

INTRODUCTION

This chapter provides a description of the flight control systems used on the 560XL/XLS/XLS+ aircraft, with a description of components and their operation. General maintenance considerations are included, with an introduction to functional and operational checks. References for this chapter and further specific information can be found in Chapter 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," Chapter 20—"Standard Practices-Airframe," and Chapter 27—"Flight Controls," of the *Aircraft Maintenance Manual (AMM)*.

GENERAL

Primary flight controls include elevators, ailerons, and rudder which are mechanically operated and controlled. They control the aircraft movement about the three axes of flight (pitch, roll, and yaw). Trim devices are attached and operated either mechanically or electrically. Flaps that increase lift and drag are actuated hydraulically and controlled mechanically. Speedbrakes that produce drag and slow the aircraft are hydraulically actuated and manually controlled. A pneumatic rudder bias system reduces rudder pedal force to achieve directional control during single engine operations. Warning and indicating systems are also provided.

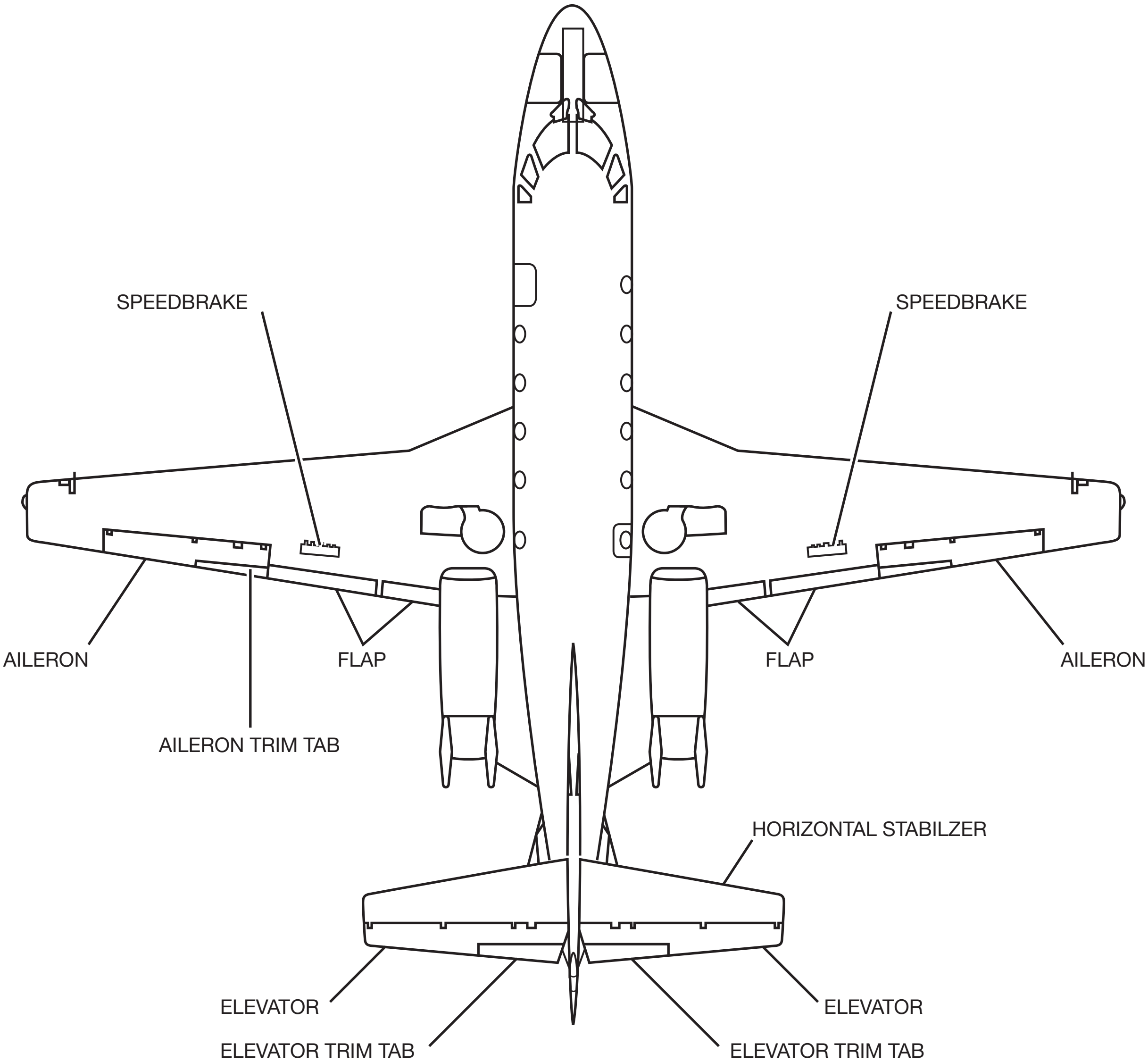


Figure 27-1. Flight Controls Overview

DESCRIPTION

Control Surfaces

The ailerons provide lateral control of the aircraft and operate mechanically by control wheel movement (Figure 27-1). A trim tab control mechanically operates a trim tab, attached to the trailing edge of the left aileron, which provides aerodynamic movement of the aileron.

The rudder provides control of the aircraft about the vertical axis and is mechanically controlled by dual rudder pedals in the flight compartment. The trim tab on the rudder trailing edge is mechanically controlled by rudder trim knob on the control pedestal.

The elevators provide longitudinal control of the aircraft and are mechanically operated by fore and aft movement of the control column. A trim tab is on the trailing edge of each elevator. The trim tab is electrically operated and has manual override control.

A two position horizontal stabilizer system automatically repositions the horizontal (to improve flight characteristics) to one of two positions, a +1° (cruise), when flaps are retracted, or –2° (take-off), when flaps are extended.

The flaps increase the lift and drag of the wing when extended and help to reduce the speed of the aircraft. The flaps are actuated hydraulically and controlled mechanically through the preselect handle and indicator follow-up system.

The speedbrakes provide fast, precise speed control. The speedbrakes are hydraulically actuated and manually controlled by a switch on the throttle quadrant.

Trim Control Surfaces

The aileron left trim tab is an adjustable trim control surface that adjusts the aerodynamic characteristics of the main control surfaces.

Rotate the aileron trim control knob on the control pedestal to obtain aileron trim.

Rudder trim is obtained by rotating the rudder trim control knob on the control pedestal. The rudder trim tab is moved so that aerodynamic forces on the tab move the rudder to the selected trim position. The rudder trim tab operates as a servo tab when the rudder is deflected from trail position.

Elevator trim is obtained electrically by actuating the trim switch on the pilot control wheel. Or use the elevator manual override control wheel on the control pedestal. The elevator trim tabs are moved so that aerodynamic forces on the tab move the elevator to the selected trim position.

NOTES

STABILIZER MISCOMPARE
Steady illumination occurs on the ground if the horizontal stabilizer does not agree with the flap handle position within 30 seconds. This condition contributes to the NO TAKEOFF annunciation.

Flashing annunciation in flight indicates:
1) The horizontal stabilizer does not agree with the flap handle within 30 seconds, or
2) The aircraft has exceeded 200 KIAS after takeoff with the flap handle greater than 0°.

NO TAKEOFF ON GROUND,
Illuminates steady to indicate one or more of the following:
Flaps are <7° or >15°, elevator is out of trim for takeoff, horizontal stabilizer is out of the takeoff position (STAB MISCOMP), and/or the speed brakes are not completely stowed (the parking brake also contributes to the NO TAKEOFF condition on certain European registered aircraft). Advancing power beyond approximately 80% N_1 with any of the above conditions existing, will activate the MASTER CAUTION lights and an aural warning sound.

XL/XLS ANNUNCIATORS

STAB MISCOMPARE

Color	Inhibited By		Debounce
Amber	LOPI		Standard

The logic for the STAB MISCOMPARE caution CAS message resides in the two position tail PCB. The DCU receives two discrete inputs from the two position tail PCB. The Stab Position Master Caution discrete indicates the two position tail is not in the correct position for the aircraft configuration. The Stab Position Fail indicates the inputs to the two position tail PCB are contradictory or invalid and the correct stab position cannot be determined. Either of these discrete will generate the STAB MISCOMPARE caution CAS message.

NO TAKEOFF

Color	Inhibited By		Debounce
Red	LOPI	In Air	Standard
White			

On the ground, the white NO TAKEOFF message will illuminate if one or more of the following conditions exist:
- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position
- **Speed Brakes** are out of takeoff position

As the throttles are advanced beyond 43° TLA, airspeed less than 67 knots, and thrust reversers not deployed, the red NO TAKEOFF message will illuminate if one or more the following conditions exist:
- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position

The red message also produces a voice aural "No Takeoff".

XLS+ CAS MESSAGES

Figure 27-2. Stabilizer Miscompare and No Takeoff Indications

HYDRAULIC PRESSURE
ON GROUND—Annunciator illuminates steady with no illumination of master caution to indicate the hydraulic system is pressurized.
IN FLIGHT—Annunciator illuminates steady with no illumination of master caution to indicate the hydraulic system is pressurized. If still on after 40 seconds, annunciator begins to flash and activates MASTER CAUTION lights.

SPEED BRAKE EXTENDED
Annunciator illuminates steady to indicate both speed brakes are fully extended. On the ground, the NO TAKEOFF annunciator will also illuminate.

XL/XLS ANNUNCIATORS

HYDRAULIC PRESSURE

Color	Inhibited By		Debounce
Amber	*LOPI	*TOPI	
White			*Standard

This message is displayed when hydraulic pressure is in the hydraulic system. Refer to amber EICAS message for details.

SPEED BRAKES

Color	Inhibited By		Debounce
White		TOPI	Standard

This message is displayed when either speed brake panel is extended. On each speed brake, there is a mechanical switch which sends a 28 Volt signal to the EICAS to display the message. When the speed brake is not extended, an open signal is sent to the EICAS system.

XLS+ CAS MESSAGES

Figure 27-3. Hydraulic Pressure Indications

Indicating and Warning Systems

Mechanical indicators on the control pedestal show the amount of trim selected on the aileron, rudder and elevator trim surfaces.

A MASTER CAUTION light on the annunciator panel illuminates to alert the flight crew of an incorrect horizontal stabilizer position. A STAB MIS COMP annunciator (XL/XLS) or amber STAB MISCOMPARE CAS message (XLS+) illuminates if flaps are selected "up" and the horizontal stabilizer does not move to the +1° position within 30 seconds; or if flaps are selected "down" and the horizontal stabilizer does not move to the –2° position within 30 seconds. A NO TAKEOFF annunciator (XL/XLS) or red NO TAKEOFF CAS message (XLS+) illuminates if aircraft is on the ground and the horizontal stabilizer is not at –2° position (Figure 27-2).

Annunciators illuminate when the speedbrakes are operated. A HYD PRESS annunciator (XL/XLS) or white HYDRAULIC PRESSURE CAS message (XLS+) illuminates when the speedbrakes are in transit. A SPD BRK EXTEND annunciator (XL/XLS) or white SPEED BRAKES CAS message (XLS+) illuminates when both speedbrakes are fully extended (Figure 27-3).

NOTES

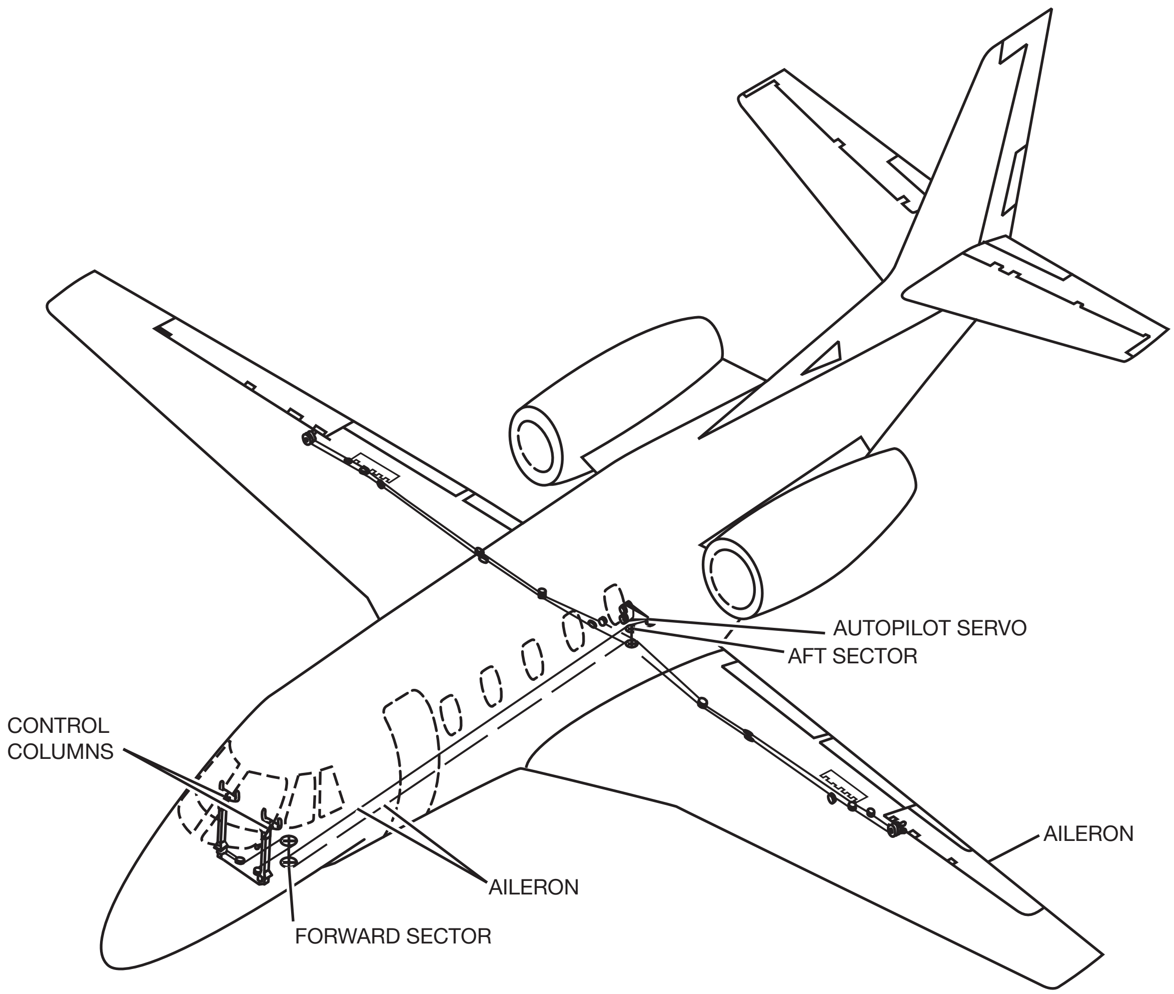


Figure 27-4. Aileron Control System

AILERON SYSTEM

DESCRIPTION

The aileron system provides lateral control of the aircraft. The ailerons are actuated mechanically by moving the control wheel, or electronically by the autopilot servo.

The aileron system includes an aileron on the trailing edge of each wing, and two control wheels in the flight compartment. The control wheels are connected to the ailerons by cables routed through a network of sectors (Figure 27-4). A forward sector assembly is below the cockpit floor (immediately aft of the center pedestal), where the cable system exits the pressure vessel. There is an aft sector assembly on the aircraft centerline aft of the rear spar of the wing, and on the trailing edge of each wing, forward of the ailerons, are the aileron quadrants.

The forward aileron sector assembly provides a sector for attaching the aileron-rudder interconnect pushrod assembly and the aft sector assembly provides a sector for attaching autopilot aileron servo cables.

NOTES

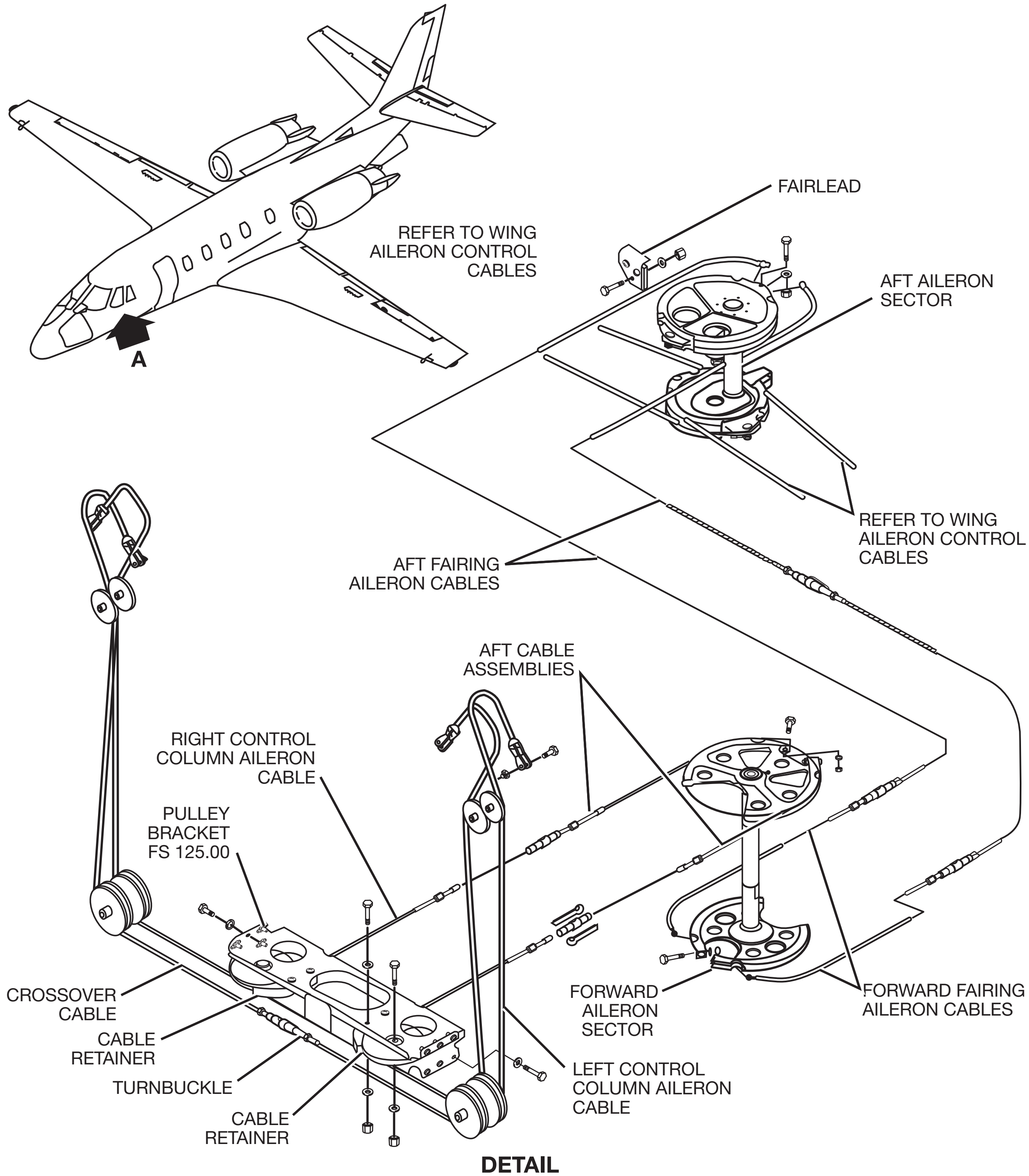


Figure 27-5. Aileron Cockpit/Fairing Cables

OPERATION

When the pilot rotates the aileron wheel the aileron control system is mechanically actuated. Cockpit, fairing cable assemblies transmit the control wheel rotation to the aileron sector assemblies causing them to rotate. Wing cable assemblies transmit the sector assembly's rotation to the wing sectors—that move the ailerons. The aileron on one wing moves up at the same time the aileron on the opposite wing moves down (Figure 27-5).

NOTES

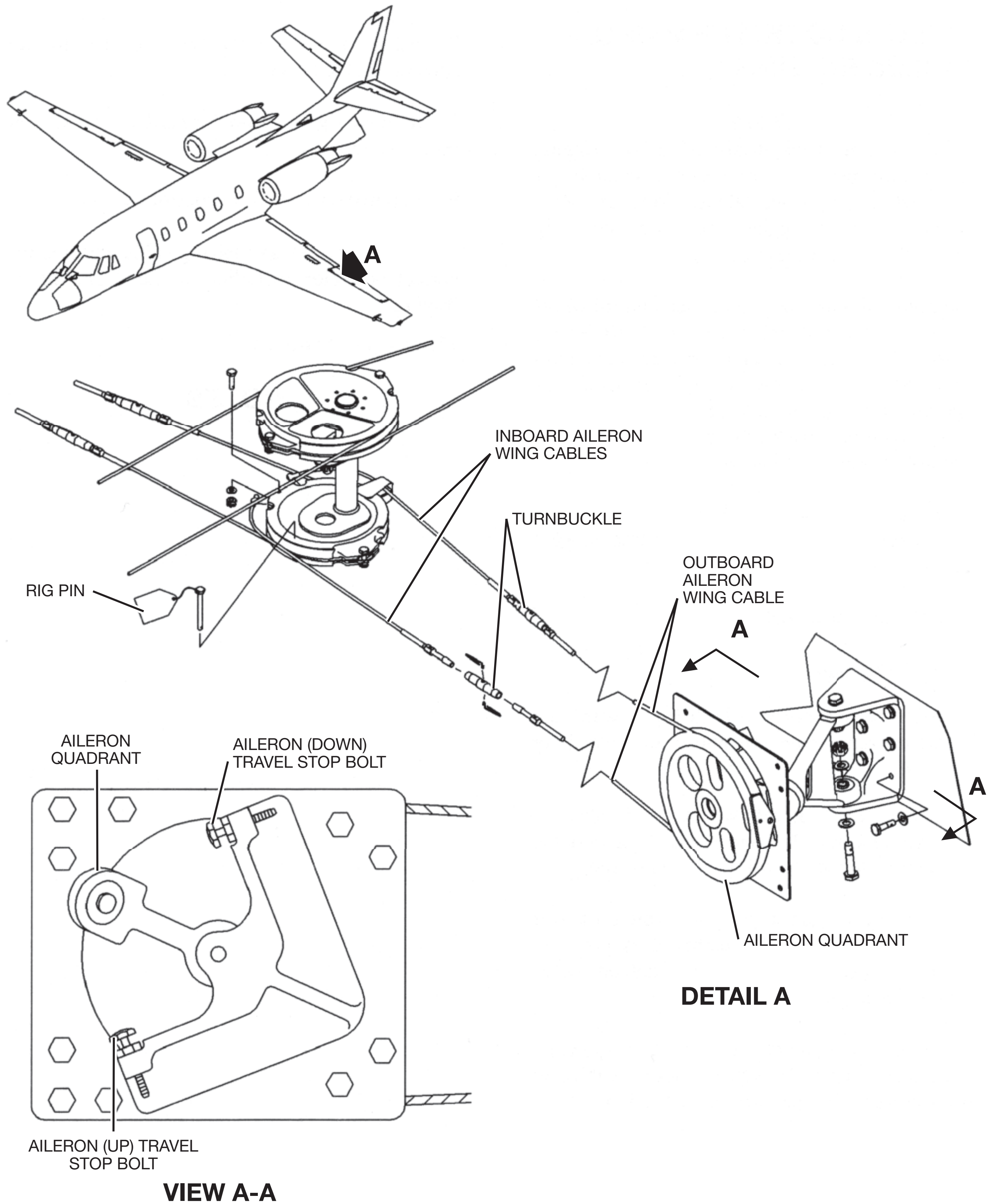


Figure 27-6. Aileron Wing Cables

Electronic actuation of the aileron control system is accomplished when the autopilot aileron servo cables rotate the aft aileron sector assembly. The wing cable assemblies transmit the sector assembly rotation to the wing sectors which move the ailerons. The cockpit, fairing cables attaching to the sector assemblies rotate the control wheels. The autopilot aileron servo has an override function, which means the operator can physically overpower the servo by manually rotating the control wheel (Figure 27-6).

NOTES

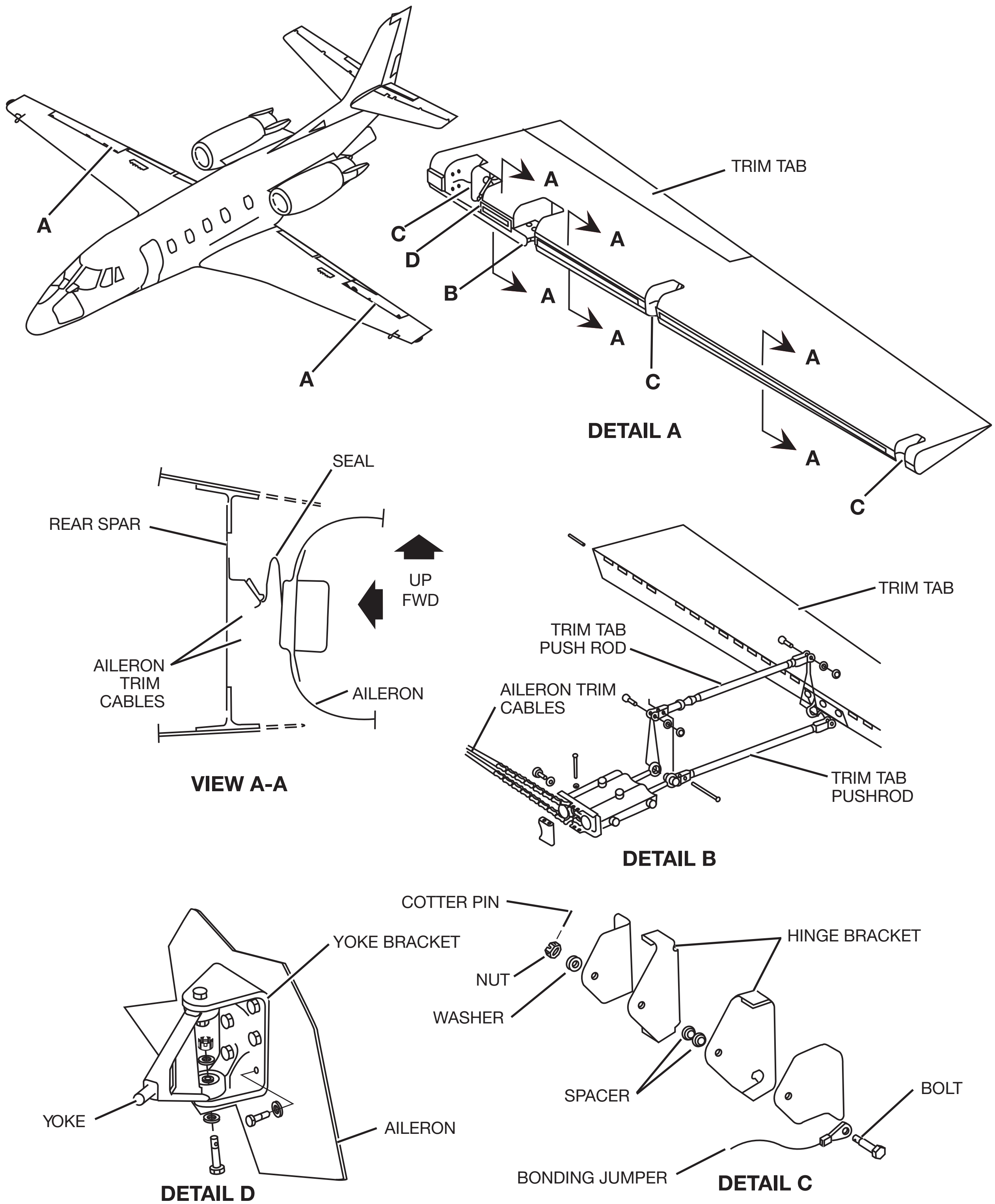


Figure 27-7. Aileron Installation

AILERON TRIM CONTROL SYSTEM

DESCRIPTION

A trim control assembly contains an aileron trim knob for roll control, a rudder trim knob for yaw control, and indicating pointers. The forward control mechanism transfers the rotating action of the trim control wheel to cable movement, that consists of a universal joint and torque tube (Figures 27-7 and 27-8).

NOTES

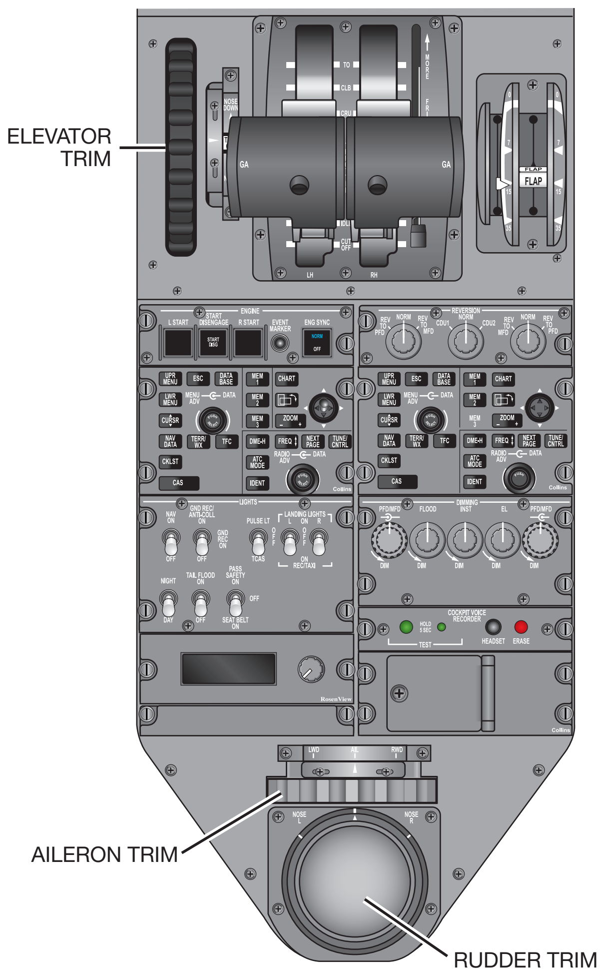


Figure 27-8. Manual Trim Wheels

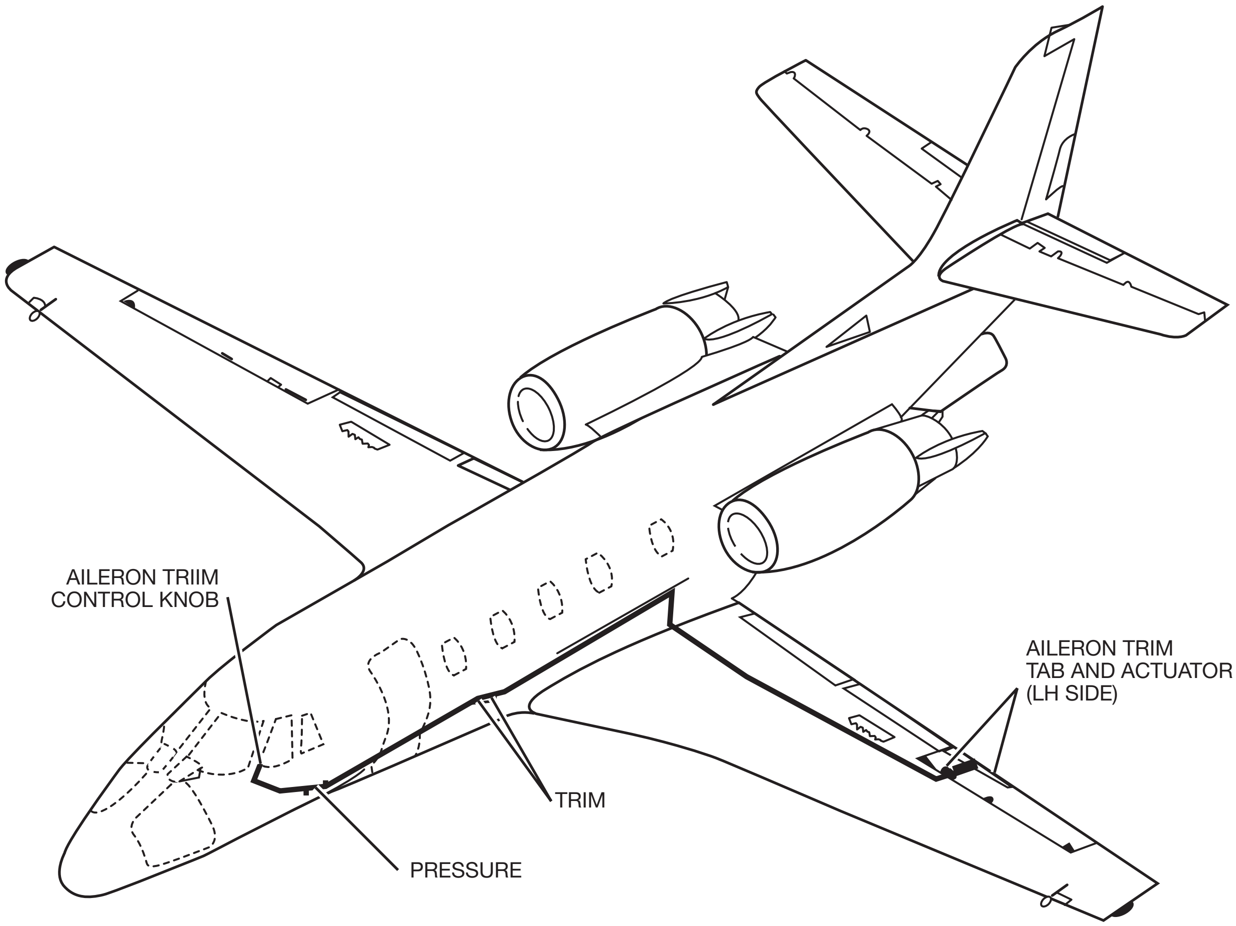


Figure 27-9. Aileron Trim System

The aileron trim tab is a movable airfoil on the inboard trailing edge of the left aileron (Figure 27-6). The aileron trim tab actuator is in the trailing edge of the left wing, forward of the aileron. The actuator has two screws in a single housing. Each screw is connected to the trim tab by a push rod. The trim tab actuator cables connect to a chain that rotates the primary sprocket to drive one screw. The two screws operate together by an interconnect chain and secondary sprockets.

OPERATION

Rotating the aileron trim control knob on the trim control assembly mechanically actuates the aileron trim control system. Cable assemblies transmit knob rotation to the aileron trim tab actuator, moving the screws in the actuator—that in turn move the aileron trim tab.

DIAGNOSTICS

Aileron Cockpit/Fairing Cables Rigging

NOTE

The aileron cockpit/fairing cable rigging procedure may be performed separately or without rigging wing cables. However, when total system rigging is required, rig cockpit/fairing cables first. There are rig pin holes in each control wheel cable drum, forward sector and aft sector assemblies. The rig pin hole in each cable drum and sector assembly is used for preliminary rigging only, since final rigging may require additional adjustment of the control wheel and ailerons.

1. Remove cockpit floor panels and aerodynamic fairing panels to gain access to control cables.
2. Release the control lock.
3. Install rig pins.
4. Align rig pin holes in both the forward and aft aileron sector assemblies; then install rig pins (see Figure 27-5).
5. Center the control wheels and place a channel across the control wheels. Secure the channel to the control wheels with tape.
6. Slide the nylon guard tube over the left crossover cable, so that it is as far outboard as possible. Connect the crossover cables and tighten cable turnbuckle to remove slack.
7. Place a tensiometer on the corresponding cable and alternately adjust cockpit cables at turnbuckles to the specified cable tension.
8. Place a tensiometer on the corresponding cable and alternately adjust fairing cable turnbuckles to the specified cable tension.
9. Adjust the autopilot servo cables to the specified cable tension.
10. Remove the rig pins from the forward and aft aileron sector assemblies.
11. Remove channel from control wheels.
12. Check cockpit and fairing cable systems for correct operation including no cable binding or fraying.
13. Safety check the turnbuckles.

Aileron Wing Cables Rigging

NOTE

Adjust the left and right wing cables simultaneously.

1. Remove the fuselage fairing access panel to gain access to the aft aileron sector assembly.
2. Gain access to the aileron wing sector and aileron wing cable turnbuckles by lowering or removing the flaps.
3. Align the rig pin holes in aft aileron sector and insert the rig pin (see Figure 27-6).
4. Position an inclinometer on the ailerons. The aileron is rigged to the streamline (trail) position (0°).
5. Place the tensiometer on the corresponding cable and alternately adjust turnbuckles on left wing to the specified cable tension.

NOTE

If any wing cable has been replaced, loosened, or disconnected to perform maintenance, perform procedural steps (5) and (6) alternately to prevent excessive cable force on rig pin structure.

6. Place a tensiometer on corresponding cable and alternately adjust turnbuckles on right wing to the specified cable tension.
7. Check the rig pin for binding; if binding occurs, a slight adjustment in cable tension relieves the binding.
8. Remove the rig pin from the aft aileron sector assembly.
9. Aileron trailing edge deflection should read 0° with the control wheels level.
10. Safety check the turnbuckles.

Aileron Deflection Check and Adjustment

1. Position the inclinometer on left and right ailerons. The aileron is in streamline (trail) position (0°) (see Figure 27-7).
2. Rotate the control wheel counterclockwise, to full travel, and measure up deflection of the left aileron.
3. Adjust up the travel stop bolt for proper deflection (from streamline position). The aileron quadrant arm should contact the stop bolt to provide travel limits.
4. Rotate the control wheel clockwise, to full travel, and measure down deflection of left aileron.
5. Adjust down the travel stop bolt for proper deflection (from streamline position). The aileron sector arm should make contact with the stop bolt to provide travel limits.

NOTE

When the up and down travel limits cannot be reached, it may be necessary to back off the travel limit stops on the right aileron quadrant.

6. Rotate the control wheel and check the right aileron for correct deflection. Then adjust in the same manner used on the left aileron.

NOTE

If correct travel limits cannot be obtained on the aileron, after the left aileron has been adjusted, rigging of the aileron wing cable system is incorrect.

7. Safety wire the travel stop bolts.
8. After rigging, move the rudder left. Note that the left aileron moves up and vice-versa.
9. Remove the inclinometer from ailerons.

Aileron Trim Flight Neutral Rigging

The following lists conditions to aid maintenance personnel in selecting procedural steps to properly rig the aileron trim tab.

A complete rerigging of the aileron trim system is required if any of the following procedures are done:

- Installing a new aileron
- Installing a new aileron trim tab
- Installing a repaired aileron
- Installing a repaired aileron trim tab
- A lateral out-of-trim force
- When the trim tab deflected angle identified as flight neutral position becomes unknown

Under certain conditions, flight neutral rigging is not effected, and the aileron tab should be rigged back to the same deflection with the pointer centered to maintain proper flight neutral.

In order to retain flight neutral rigging after maintenance, note the location of the aileron trim tab deflection, with the trim indicator centered and aileron in the streamline position prior to performing disassembly.

The location should be noted prior to the following procedures.

- Installing the same aileron
- Installing the same aileron trim tab
- Replacing a trim tab actuator
- Replacing or adjusting cable assembly, brackets and pulleys
- Replacing or adjusting trim indicator assembly
- Replacing or adjusting trim tab travel stops
- Adjusting cable tension

NOTE

A flight is required to determine flight neutral position of the aileron trim tab system. The streamline (trail) position and flight neutral may not be the same.

Flight for Determining Flight Neutral Position

The aircraft must be fully airworthy prior to flight. Complete all other ground rigging procedures and have all system components properly safety wired and inspected. All inspection and access panels should be installed.

1. Stabilize at 250 KIAS and set aileron trim to produce zero force at the control wheel with the wings laterally level. Mark the aileron trim indicator position (use a grease pencil or suitable substitute) on the pedestal for reference.
2. With the aircraft in level flight, adjust the trim to a flight neutral position, with no significant fuel load differential. The control wheels are considered level when they are within ±1.50° from horizontal. If wheels are not level (greater than ±1.50°), use a grease pencil to mark a position on the control column (and mark the control wheel for a reference) when rigging the aileron trim tabs.

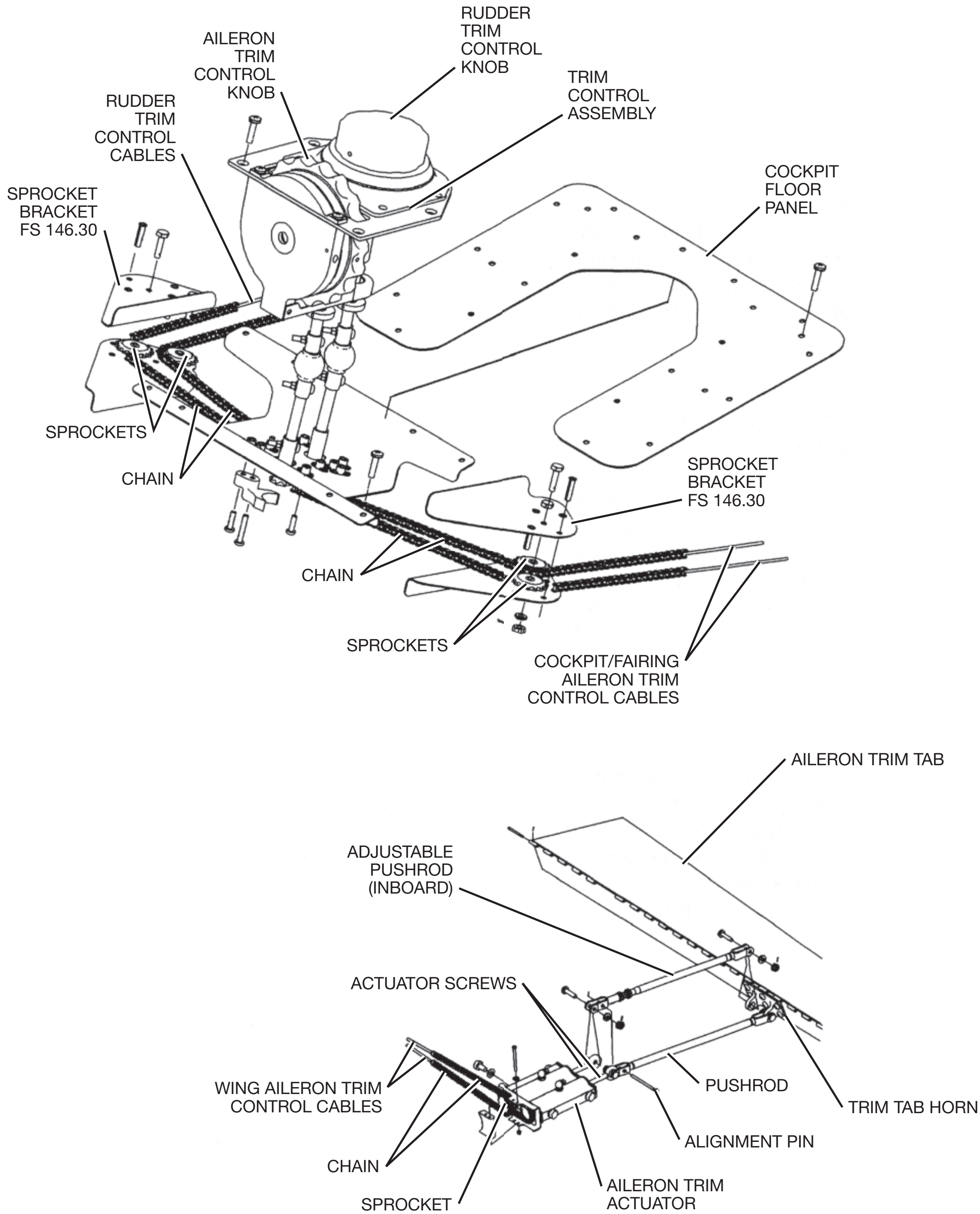


Figure 27-10. Aileron Trim Knob and Actuator

27 FLIGHT CONTROLS

After Flight Ground Adjustment

1. Use the reference position obtained in flight. Rotate the trim control wheel until the aileron trim tab indicator points to the reference mark (Figure 27-10).
2. Hold the aileron trim tab in the position corresponding to the reference mark. Use an inclinometer to measure the trim tab deflection angle.

NOTE

Deflection angle beyond 7° up or down from streamline (trail) position requires rerigging the aileron system.

3. Without moving trim knob, loosen the two screws, securing the indicator to the vernier pointer. Align the pointer with the center tick mark on the aileron trim position scale and tighten the screws.

NOTES

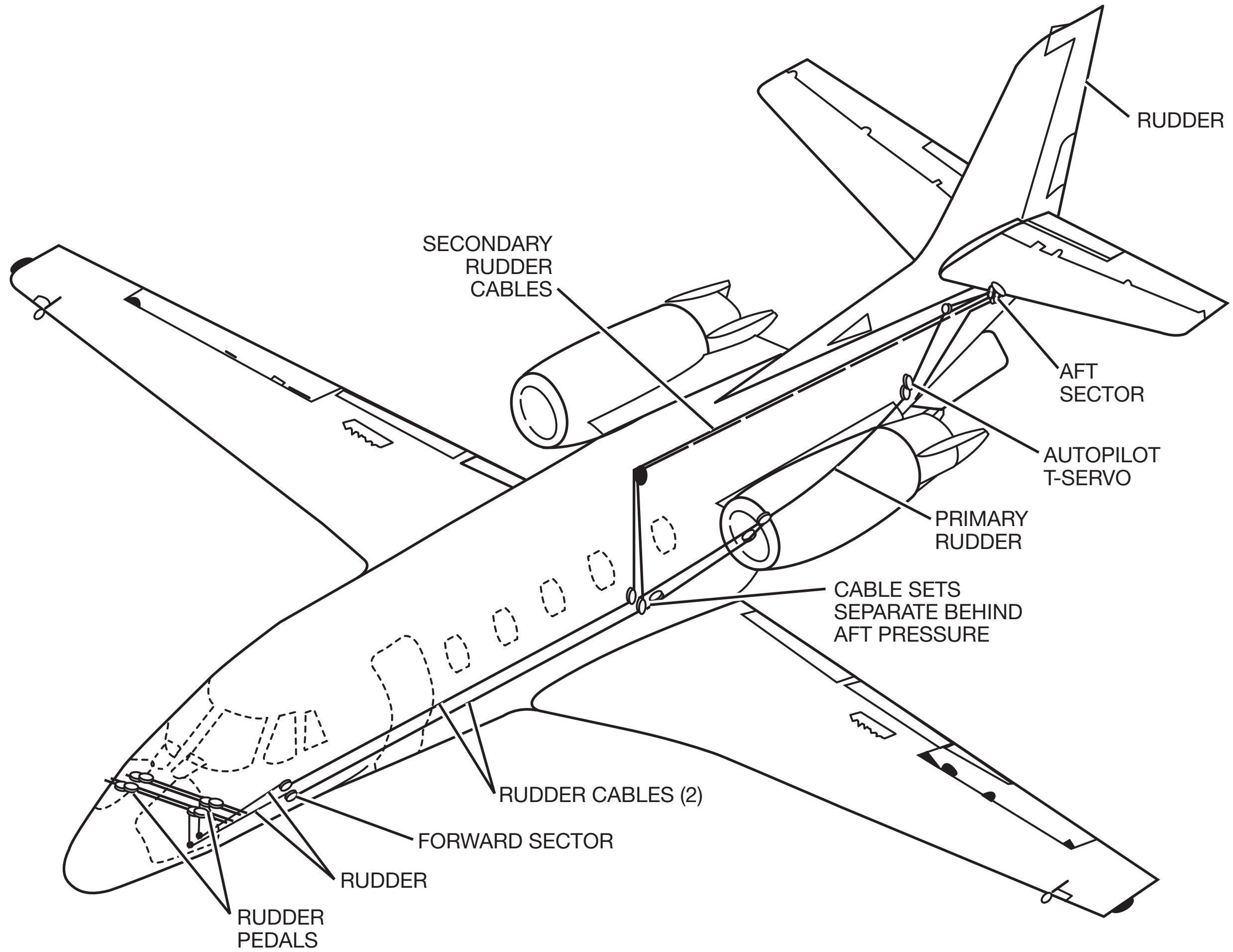


Figure 27-11. Rudder Control System

RUDDER SYSTEM

DESCRIPTION

The rudder system includes:

- Rudder on the trailing edge of the vertical stabilizer (Figure 27-11)
- Rudder sector in the aft section of the tail cone below the rudder
- Rudder pedal assemblies in the flight compartment
- Forward pass-thru sector assembly below the cockpit floor
- Dual rudder control cable assemblies (primary and secondary) between lower forward rudder sector and aft rudder sector

The forward sector assembly provides a sector for attaching the aileron-rudder interconnect pushrod assembly. The aft sector assembly provides a sector for attaching the rudder autopilot servo cables, and rudder bias cables.

OPERATION

The rudder system provides control of the aircraft about the vertical axis. The rudder is mechanically actuated by moving the rudder pedals or electrically by the autopilot servo.

Mechanical actuation of the rudder control system is accomplished when any of the rudder pedals are depressed. A set of rudder pedals is installed at each pilot's station. Torque tube and bridge assemblies connect the rudder pedal sets together. This provides the corresponding rudder pedal movement between rudder pedal sets. Cable assemblies transmit the rudder pedal movement to the upper sector of the rudder pass-thru sector assembly, causing the sector assemblies to rotate. Dual cable assemblies transmit rotation of the lower sector-to the aft rudder sector, which deflects the rudder.

Electrical actuation of the rudder control system is accomplished when the autopilot rudder servo cables apply a force on the rudder aft sector repositioning the rudder. The rudder torque tube attaches to the rudder sector and deflects the rudder. The autopilot rudder servo has an override function, which means the operator can physically overpower the servo by manually depressing the rudder pedals.

NOTES

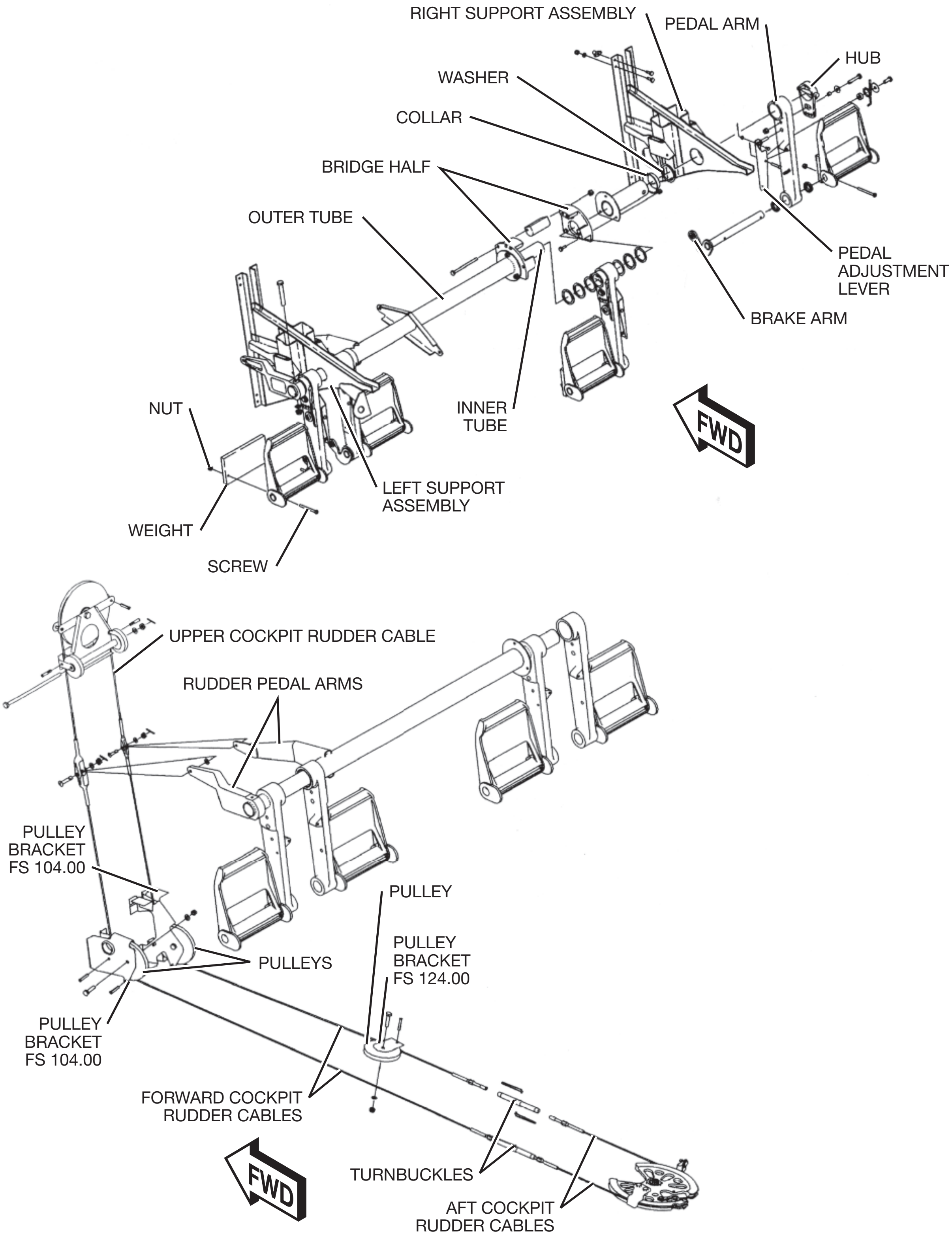


Figure 27-12. Rudder Pedals and Cockpit Cable System

COMPONENTS

Rudder Pedals

The rudder pedals operate the rudder, nose wheel steering and brakes (Figure 27-12). Pushing on the lower part of the rudder pedals operates the rudder and steering. Pushing on the upper part of the pedal operates the brakes.

The rudder pedals are on two tube assemblies. The left pedals are connected to the inner tube assembly; and the right pedals to the outer tube assembly. A bridge transfers the torque of the outer tube assembly across the left pedal (at the copilot position). One arm extending from the inner tube assembly and two arms from the outer tube assembly connect to the control cables. The upper cockpit rudder cable maintains tension on the control cables when a rudder pedal is depressed.

Each rudder pedal adjusts to three different positions by pushing on the lower end of the pedal adjustment lever and moving the pedal to the desired position.

NOTES

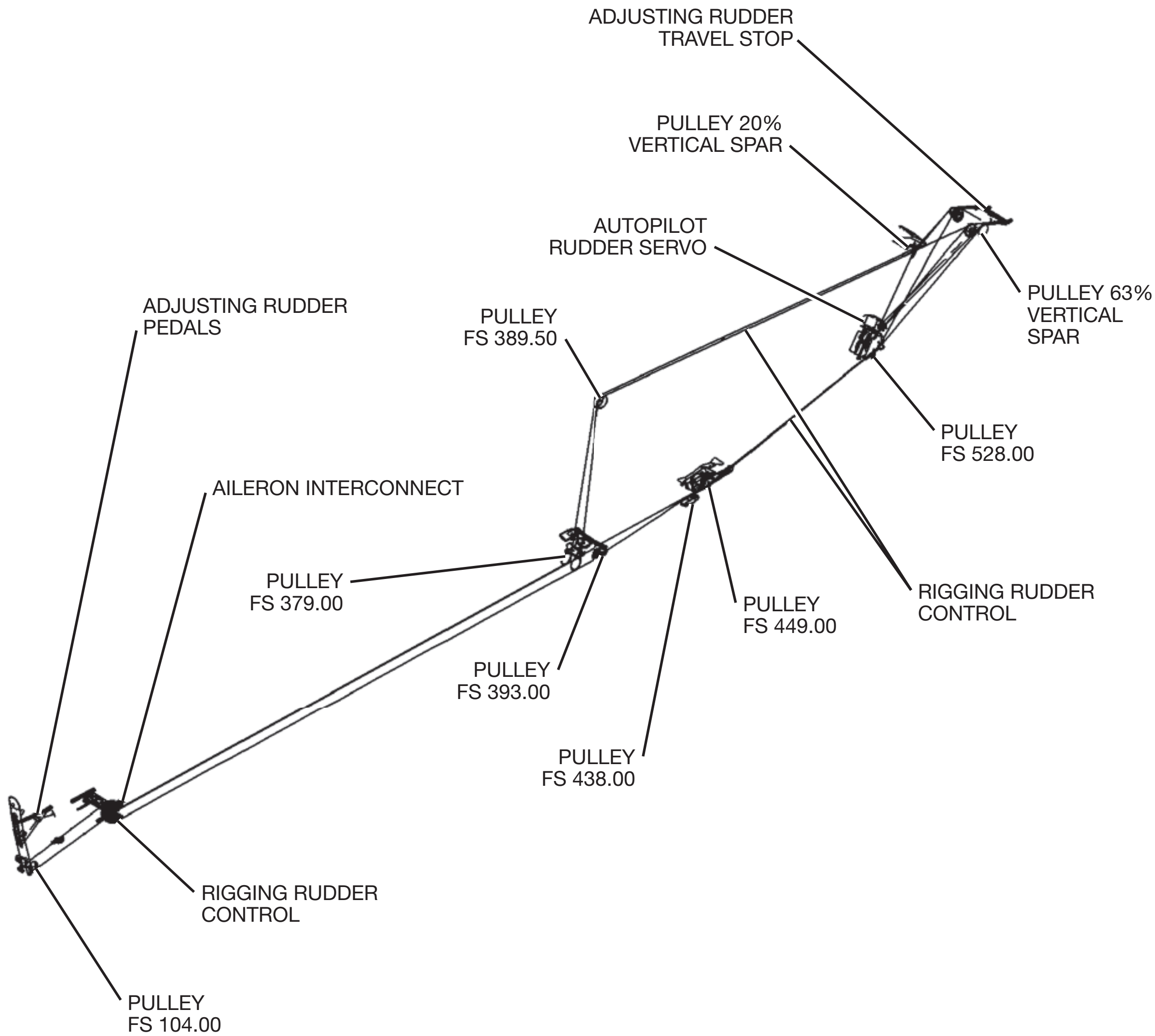


Figure 27-13. Primary and Secondary Rudder Cable Systems

Fairing/Tail Cone Rudder Control Cables

The fairing/tail cone rudder control cables consist of two sets of cables originating at the lower sector of the forward rudder sector assembly—and terminating at the aft rudder sector assembly(Figure 27-13). The cable sets are separated from each other in the uncontained rotor burst zone to minimize the risk associated with the loss of both an engine and rudder control during takeoff. From the forward sector, both cable assemblies route aft. Behind the pressure vessel, the secondary control cables route up and along the upper fuselage, while the primary control cables continue along the lower fuselage (as they both proceed to the aft rudder sector).

Cable Dampener

A cable dampener is installed on each set of rudder cables as well as the elevator cables. By pulling the individual cables against a rub block, minor vibration is cancelled before it has a chance to be magnified in the center of the long unsupported cable length. As aforementioned, tension on the cable dampeners must be released before adjusting cable tensions on the respective system.

Rudder

The rudder is a movable air-foil hinged to the vertical stabilizer rear spar. A sealed bearing is installed in each of the three hinge assemblies to provide a bearing surface for rudder movement.

NOTES

DIAGNOSTICS

Rigging Cockpit Rudder Control System

NOTE

Ensure that the nose wheel is free to rotate during rudder system rigging, or disconnect the nosewheel steering bungee.The position of the rudder pedals are adjustable by depressing the lever on the rudder brake arm. During rigging, place the rudder pedals in the center hole position.

1. Remove the cockpit floor panels to gain access to cockpit rudder cable turnbuckles.
2. Place the rudder pedals in neutral position. Clamp pilot's pedals together using tool CJMDL27-004, so they are not able to move relative to each other.
3. Install the rig pin in the forward rudder sector assembly.
4. Place a tensiometer on corresponding left and right cockpit rudder cables, and alternately adjust, connecting turnbuckles to specified cable tension.

NOTES

Rigging Fairing/Tail Cone Rudder Cables

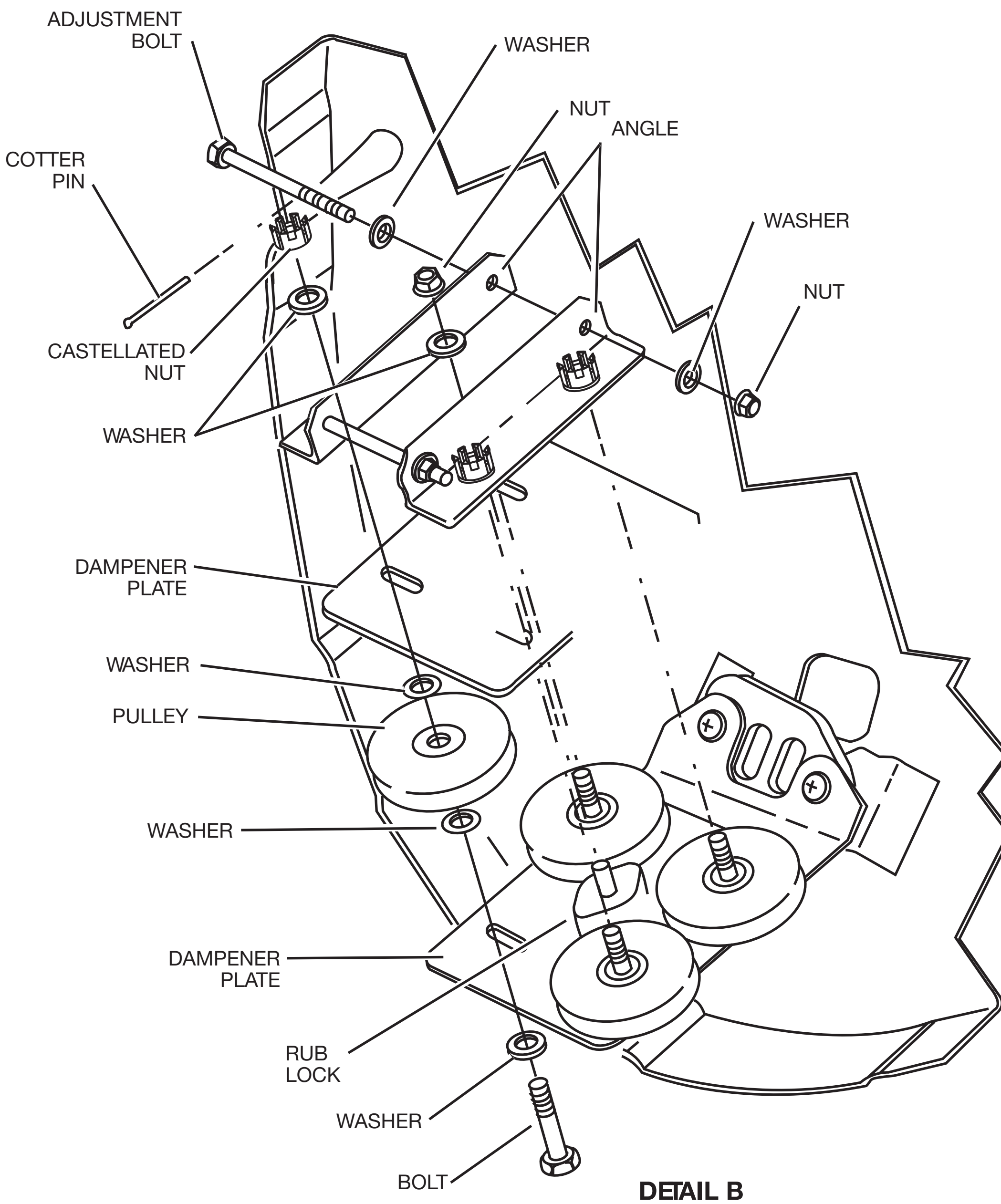
1. Install the rig pin in forward rudder sector assembly.
2. Place CJMDL27-008 Check Fixture on the rudder and secure the rudder in streamline (trail) position.
3. Release tension on the upper and lower rudder cable dampeners (Aircraft 5001, 5025 and subsequent and Aircraft 5002 through 5024 incorporating SB560XL-27-02).
4. Place a tensiometer on corresponding left and right primary rudder cables, and alternately adjust, connecting turnbuckles to specified cable tension.

NOTE

The secondary rudder cable turnbuckles are above the tail cone baggage compartment ceiling panels, and upper fuselage aft of the tail cone baggage compartment. Primary rudder cable turnbuckles are in lower fuselage aft of the tail cone baggage compartment.

5. Place a tensiometer on corresponding left and right secondary rudder cables, and alternately adjust, connecting turnbuckles to specified cable tension.
6. Adjust the autopilot servo cable to its specified tension with the rudder in neutral (trail) position.
7. Safety check the turnbuckles.
8. Remove the rig pin from forward rudder sector.
9. Apply tension to the upper and lower rudder cable dampeners, and check (Aircraft 5001, 5025 and subsequent and Aircraft 5002 through 5024 incorporating SB560XL-27-02).

NOTES



NOTE:
ELEVATOR DAMPENER IS SHOWN, BUT TYPICAL CABLE DAMPENER ASSEMBLY IS USED THREE PLACES. PRIMARY RUDDER ELEVATOR DAMPENER WHILE SECONDARY DAMPENER IS LOCATED AT THE TOP OF THE TAIL CONE.

UNITS 5001, 5025 AND ON
AND 5002 THRU 5024,
INCORPORATING SB 560XL-27-02

Figure 27-14. Rudder Cable Dampener

Cable Dampener Adjustment

NOTE

Fine-tuning the rudder cable dampener requires the upper and lower rudder dampener be installed and fine-tuned simultaneously. All tension measurements should be made on each cable individually, but break-out friction is measured on the system as a whole.

1. Verify the rudder cable tension and rigging.
2. Using a spring scale, measure and record the amount of break-out friction of the rudder pedal from the neutral position.
3. Verify that the rub block (Figure 27-14) has the correct alignment—symmetric about the centerline of the rub block with no cable deflection.
4. Tighten the bolt holding the rub block in place. Do not tighten the pulley bolts.
5. Using the threaded adjustment bolts, adjust until pulleys just make contact with cables and are just nested in radius of each pulley.
6. Tighten the pulley bolts and verify that all pulleys turn when cables move, and that the tension has not increased.
7. Measure and record the distance between pulley-bolt-centers in the direction parallel to the slots.
8. Loosen pulley bolts slightly.
9. Adjust the nuts on the tensioning bolts installed through the angles of the dampener assembly 0.34 inch (approximately 11 full turns of the nut).
10. Using a spring scale, verify that the new breakout friction of the rudder pedals from the neutral position has increased 4 ± 1 pound.
11. If break-out friction needs to be increased or decreased, tighten or loosen the tensioning nuts 1 to 2 turns accordingly. Repeat until desired break-out friction is obtained.
12. Tighten and secure the nuts of the rudder cable dampener assembly using nuts and cotter pins.

NOTES

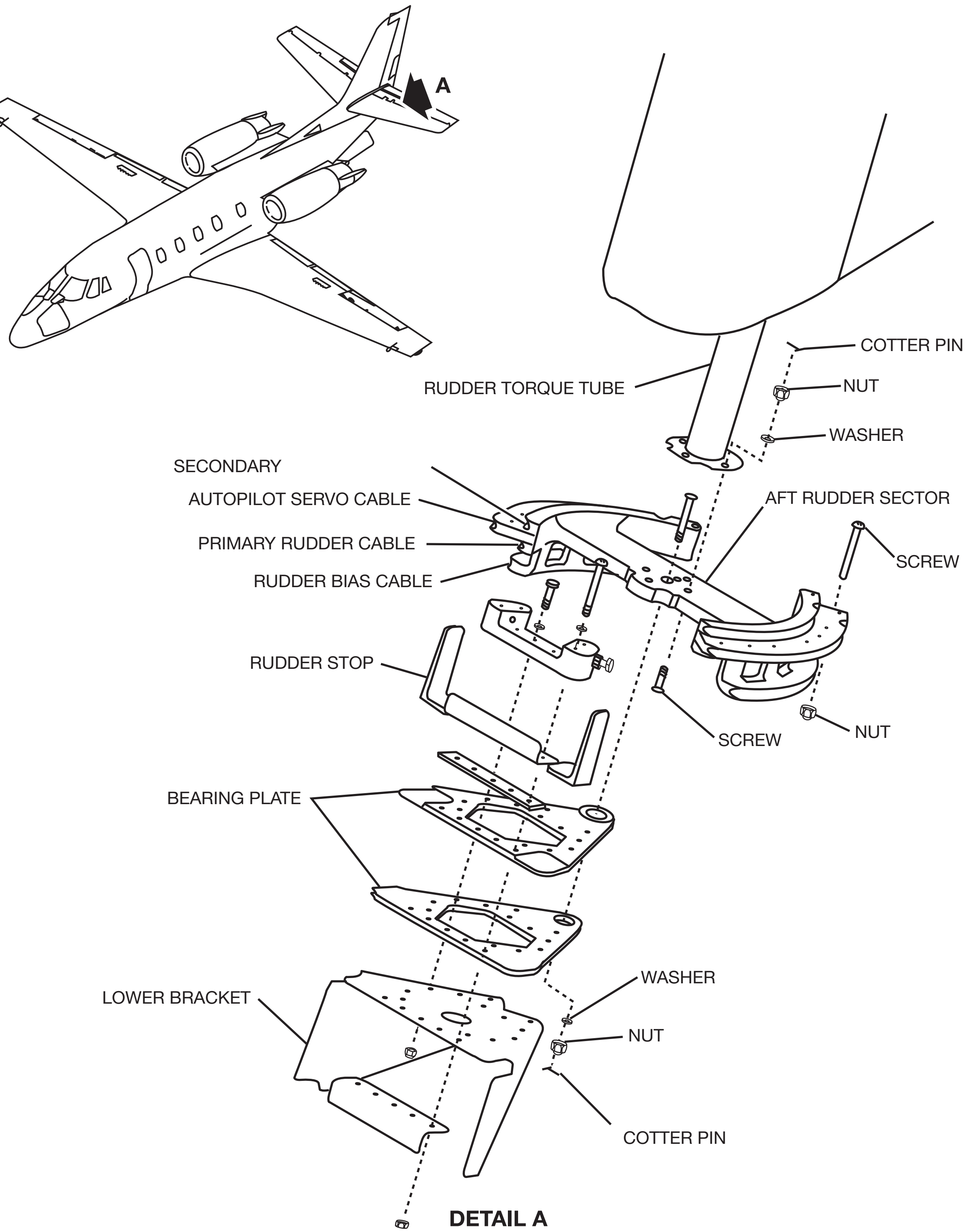


Figure 27-15. Aft Rudder Sector

Adjustment of Rudder Stops

1. Depress the left rudder pedal to full travel. Loosen the travel stop bolt locknut at the rudder bellcrank (Figure 27-15).
2. Measure the rudder left deflection and adjust the left rudder travel stop.
3. Tighten the travel stop bolt locknut.
4. Depress the right rudder pedal. Loosen the travel stop bolt locknut.
5. Measure the rudder right deflection and adjust the right rudder travel stop.
6. Tighten the travel stop bolt locknut.

NOTE

If correct rudder and rudder pedal travel can not be obtained, inspect the condition of the rig.

7. Reinstall the panels, plates, and fairings.

NOTES

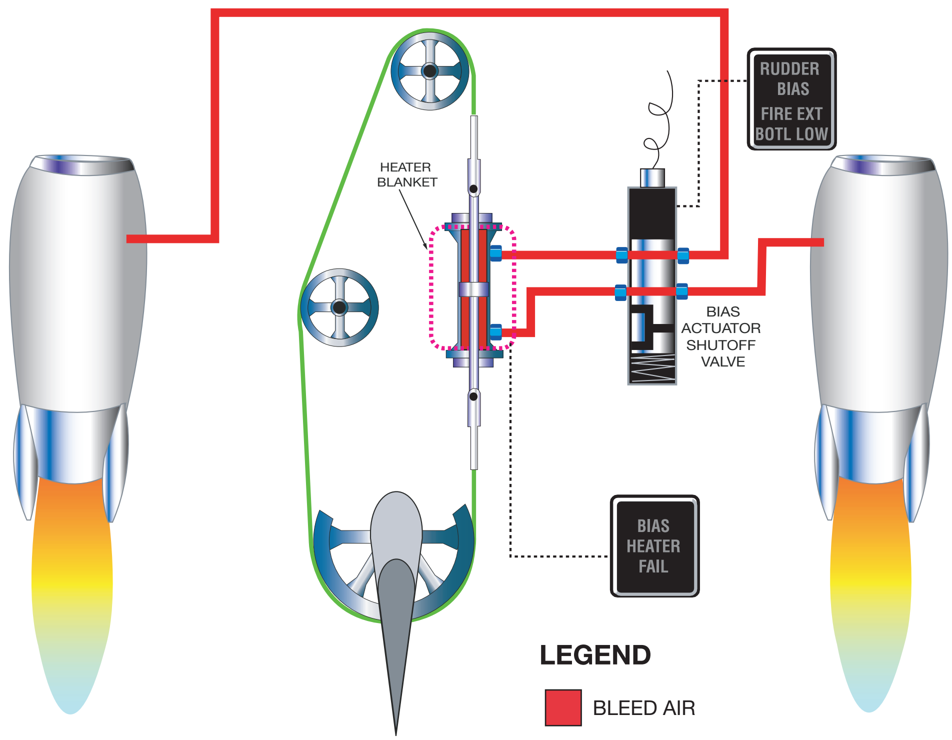


Figure 27-16. Simplified Rudder Bias Bleed Air Flow

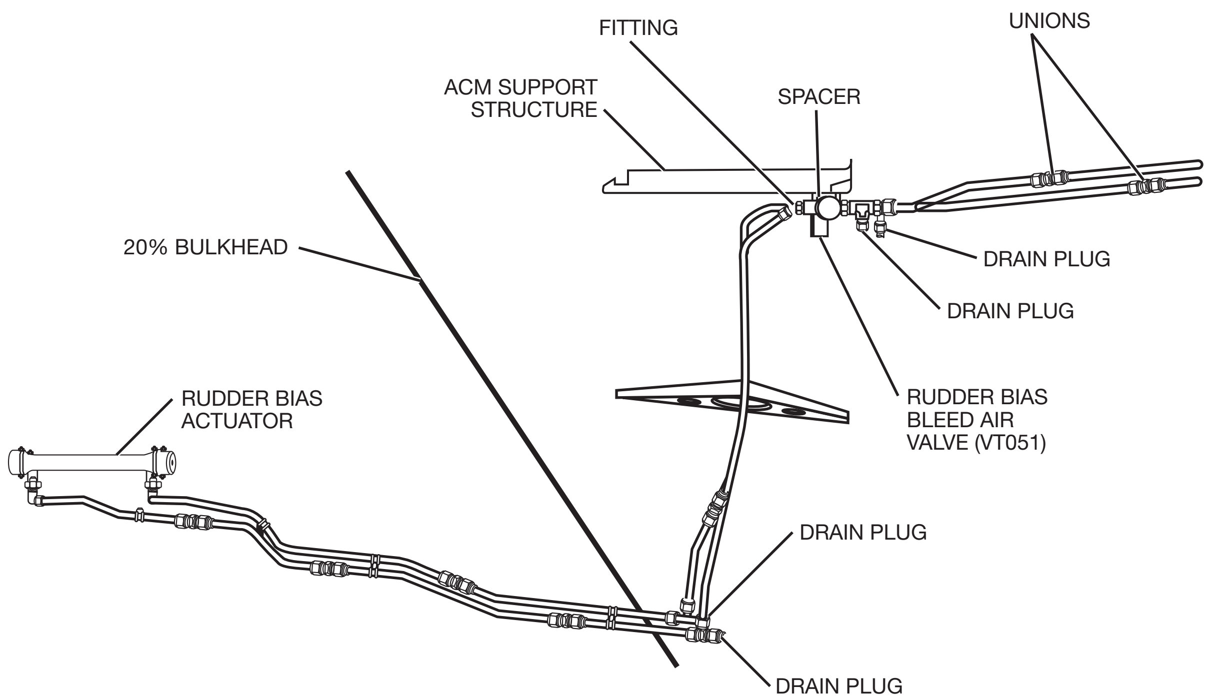


Figure 27-17. Rudder Bias Bleed Air System

RUDDER BIAS SYSTEM

GENERAL

The rudder bias system provides pneumatic assistance to position the rudder in the event of the loss of one engine. It automatically engages upon the loss of one engine. A pneumatic actuator, powered by engine bleed air, pulls the rudder into a position that compensates for asymmetric thrust due to engine failure (Figure 27-16). This system is comprised of separated left and right actuated pneumatic systems plumbed into one dual acting cylinder (Figure 27-17). The pneumatic systems are balanced and do not affect rudder position when acting equally together. It is only in an unbalanced engine thrust condition that the rudder bias system delivers rudder assist to compensate for the resulting yaw.

NOTES

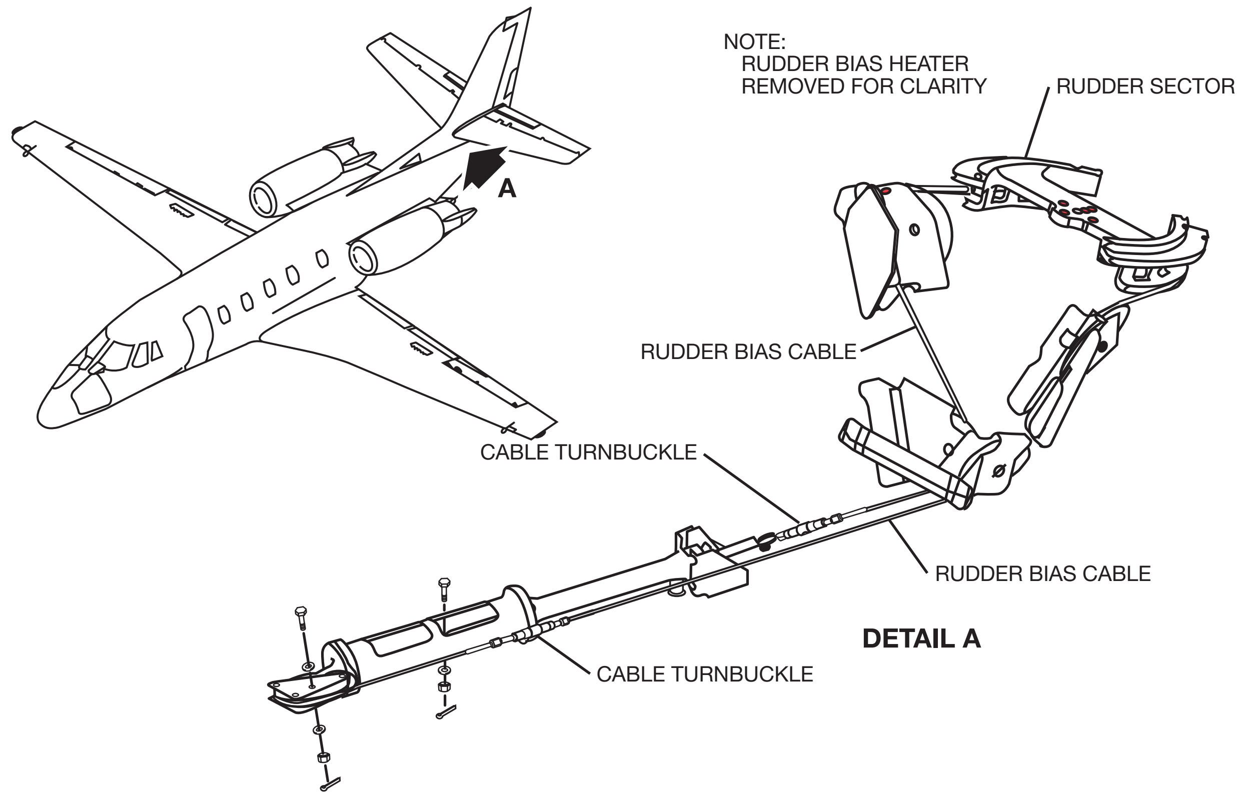


Figure 27-18. Rudder Bias Cable System

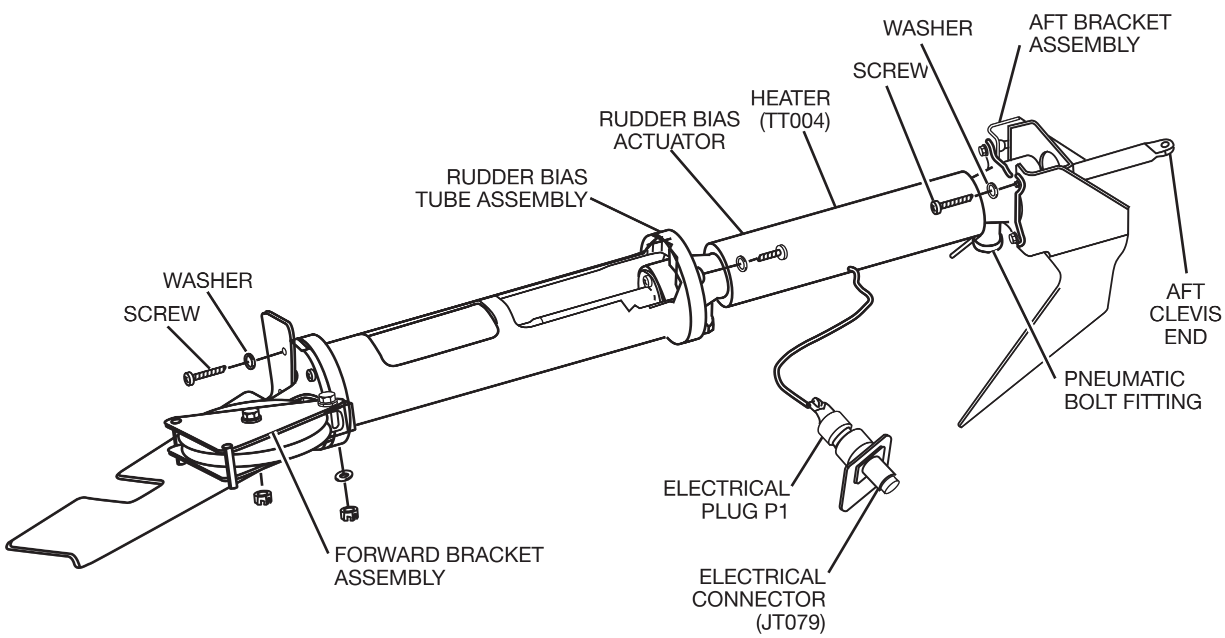


Figure 27-19. Rudder Bias Actuator Assembly

COMPONENTS

Aft Rudder Sector

The aft rudder sector (for rudder bias equipped aircraft) has a smaller diameter at the primary and secondary input cable grooves, that results in a higher gear ratio between rudder pedals and rudder. The result is more rudder travel for the same pedal travel (±28.5° instead of ±22°).

Rudder Bias Cable System

A closed loop cable system has been added in the tail cone between the 20% bulkhead and the 63% bulkhead. This cable is driven by the bias actuator and terminates on the bottom cable groove of the aft rudder sector (Figure 27-18).

Bias Actuator

A bleed-air bias actuator operates the bias cable system to rotate the bias input sector either left or right. A piston-type actuator is controlled by bleed air from the left and right engines (Figure 27-19). Left engine bleed air is supplied to one side of the piston, while right engine bleed air is supplied to the other side.

Bleed-Air Line Drain Plugs

To prevent accumulation of water in the rudder bias system, modified plugs are installed at low points of bleed air lines. One set immediately forward of the rudder-bias bleed air valve and another immediately forward of the 20% bulkhead. Drain plugs have a 0.040 inch hole drilled that allows condensation to be continuously expelled.

Bias Actuator Shutoff Valve

There is a solenoid-operated shutoff valve between the engine bleed-air lines and the actuator. Upon power up, the valve opens and ports right engine bleed air into the rudder right command half. Left engine bleed air ports to the rudder left command half of the bias actuator. When power is removed from the valve, engine bleed air (from both engines) shuts off; and both command halves of the actuator vent into atmosphere. Electrical power automatically ceases when either thrust reverser is deployed (or by manually disengaging the RUDDER BIAS circuit breaker on the left CB panel in the cockpit). The position of the valve is monitored and an annunciator illuminates if it fails.

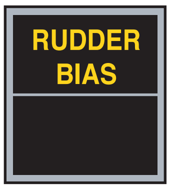
Rudder Bias Heater Blanket

A heat blanket protects the rudder bias system from freezing. The heat blanket becomes operational upon power up. It is a two element system, each controlled by its own thermostat RTD (resistive thermal device) for redundancy. The HZ026 circuit breaker is in the aft J-Box for overload protection. The RUDDER BIAS HEATER PCB (NZ029), in the left-hand logic module box of the aft J-Box, controls the heating elements according to inputs from the two RTD’s (resistive thermal device).

ELECTRICAL OPERATION

The rudder bias control valve is initially powered open when the aircraft battery switch is placed in the BATT position. The valve is momentarily commanded to close when the left or right thrust reverser is in transit, deployed, or emergency stowed.

The RUDDER BIAS annunciator (XL/ XLS) or amber RUDDER BIAS FAULT CAS message (XLS+) illuminates when the valve command signal and the valve position do not agree for more than one second (Figure 27-20).



RUDDER BIAS
Annunciator flashes to indicate rudder bias system malfunction, rudder bias system valve not in commanded position. Activates MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

RUDDER BIAS FAULT			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	1 Second

This message monitors the rudder bias control valve for proper operation. The EICAS system gets 3 inputs: one input is the command going to the valve, and the other 2 inputs are from two mechanical switches within the valve that indicate the position the valve is in. For the command input, 28 Volts means the valve is being commanded to open, and open means the valve is being commanded to close. For the sense inputs, ground means that the valve is in the respective position, and open means the valve is not in the respective position.

XLS+ CAS MESSAGE

Figure 27-20. Rudder Bias Fail Indications

The rudder bias heater PCB performs a test of the heater systems upon initial power up. The BIAS HEATER FAIL switchlight (XL/XLS) on the center instrument panel or amber RUDDER BIAS HEAT FAIL CAS message (XLS+) illuminates if a system failure is detected. Push the BIAS HEATER FAIL switchlight (XL/XLS) to make the light illuminate steady until the failure is cleared (Figure 27-21).

BIAS HEATER FAIL (XL and XLS)
Steady illumination indicates the rudder bias heating blanket is heating. Flashing light indicates blanket sensor failure. Pressing the light causes steady illumination. This annunciator does not activate the MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

RUDDER BIAS HEAT FAIL			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
		*SIPI	

This message is displayed when the rudder bias heater blanket is failed as determined by the Rudder Bias Heater PC card. When the heater blanket has failed, the PC card sends an open signal to the EICAS system, which posts the message. When the heater blanket is operating normally, the PC card sends a ground signal, which causes the EICAS to remove the message.

XLS+ CAS MESSAGE

Figure 27-21. Rudder Bias Heat Fail Indications

NOTES

After initial test, the rudder bias PCB maintains the actuator temperature above freezing. The BIAS HEATER FAIL annunciator (XL/XLS) or white RUDDER BIAS COLD CAS message (XLS+) illuminates steady until the actuator has reached operating temperature (Figure 27-22). A low temp or high temp signal from either sensor while on the ground flashes the warning light. In flight, both sensors are required to detect a low temp or an over temp condition before a warning is annunciated.

RUDDER BIAS COLD			
Color	**Inhibited By**		**Debounce**
White	LOPI	TOPI	Standard
		*SIPI	

This message is displayed while the rudder bias heater system is cold and it is not failed. The rudder bias actuator is wrapped with an electrical heater blanket . The heating is controlled by a Rudder Bias Heater PC card. When PC card senses the heater blanket is cold, the card sends an open signal to the EICAS system, which posts the message if it is not failed. When the heater blanket has warmed up, the card sends a ground, which causes the message to be removed.

* The message is also inhibited by an engine and/or APU start on the ground.

Figure 27-22. Rudder Bias Cold Indication

NOTES

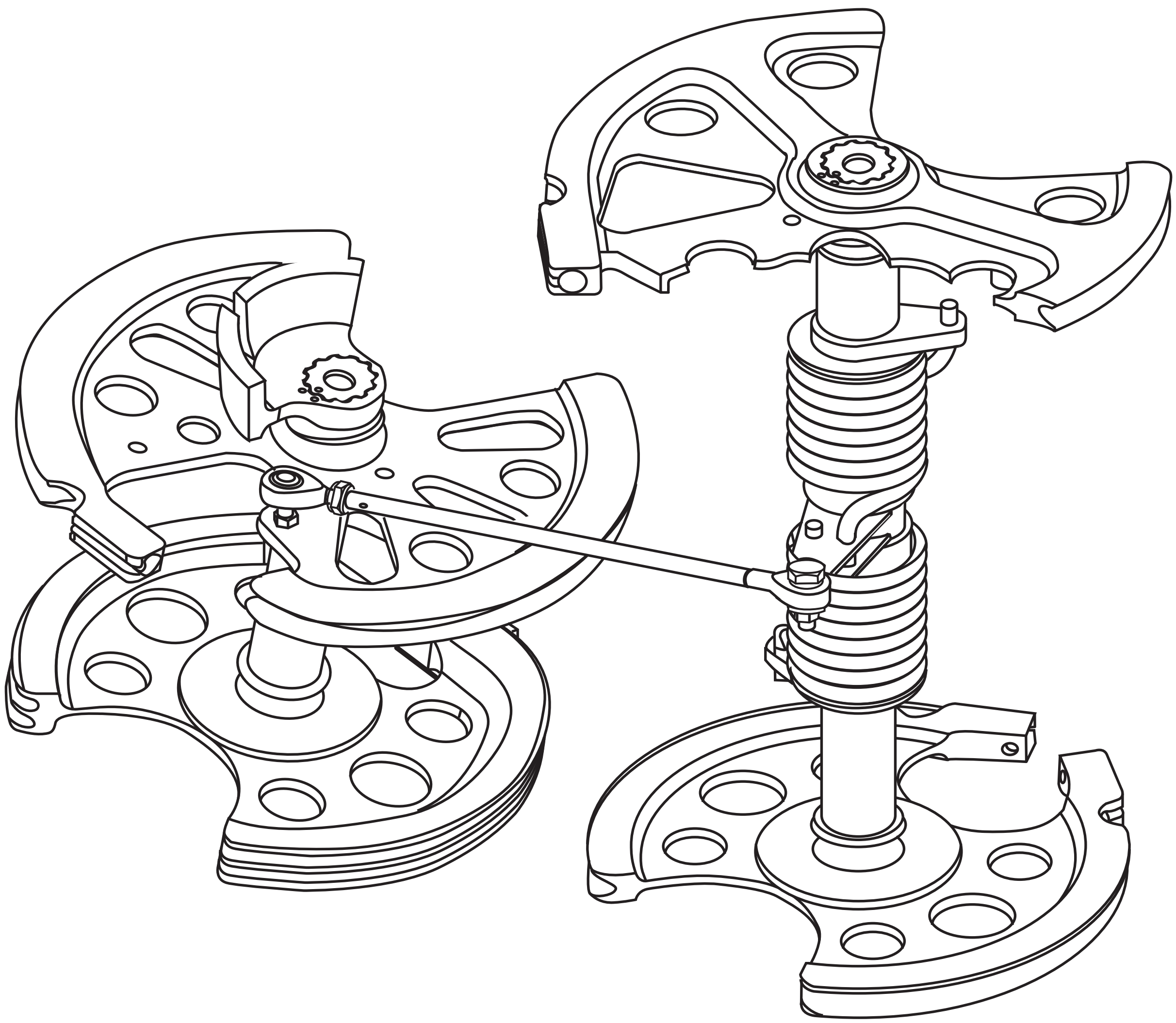


Figure 27-23. Rudder/Aileron Interconnect System

RUDDER/AILERON INTERCONNECT

DESCRIPTION

The rudder and aileron systems are connected by a torsion bungee at the feedthrough sectors. Operation of either system produces a coordinated response on the other system. The interconnect operates in conjunction with the primary controls. When the pilot inputs a left rudder command through the pedals, the torsion bungee imposes a left roll torque to the aileron system. A left roll input likewise produces a left yaw response. Right inputs produce right responses. This allows for a more automatically coordinated turn.

DIAGNOSTICS

Rigging Rudder/Aileron Interconnect

1. Verify that the torsion bungee assembly is complete and aileron and rudder systems are rigged properly.
2. Install the rig pins in aileron and rudder feedthrough sectors.
3. Adjust the rudder/aileron interconnect pushrod so that it fits between rudder sector and bungee without preloading bungee (Figure 27-23).
4. Safety wire the rod end with 0.063 inch safety wire.
5. Install the interconnect pushrod.
6. Remove the rig pins.

NOTES

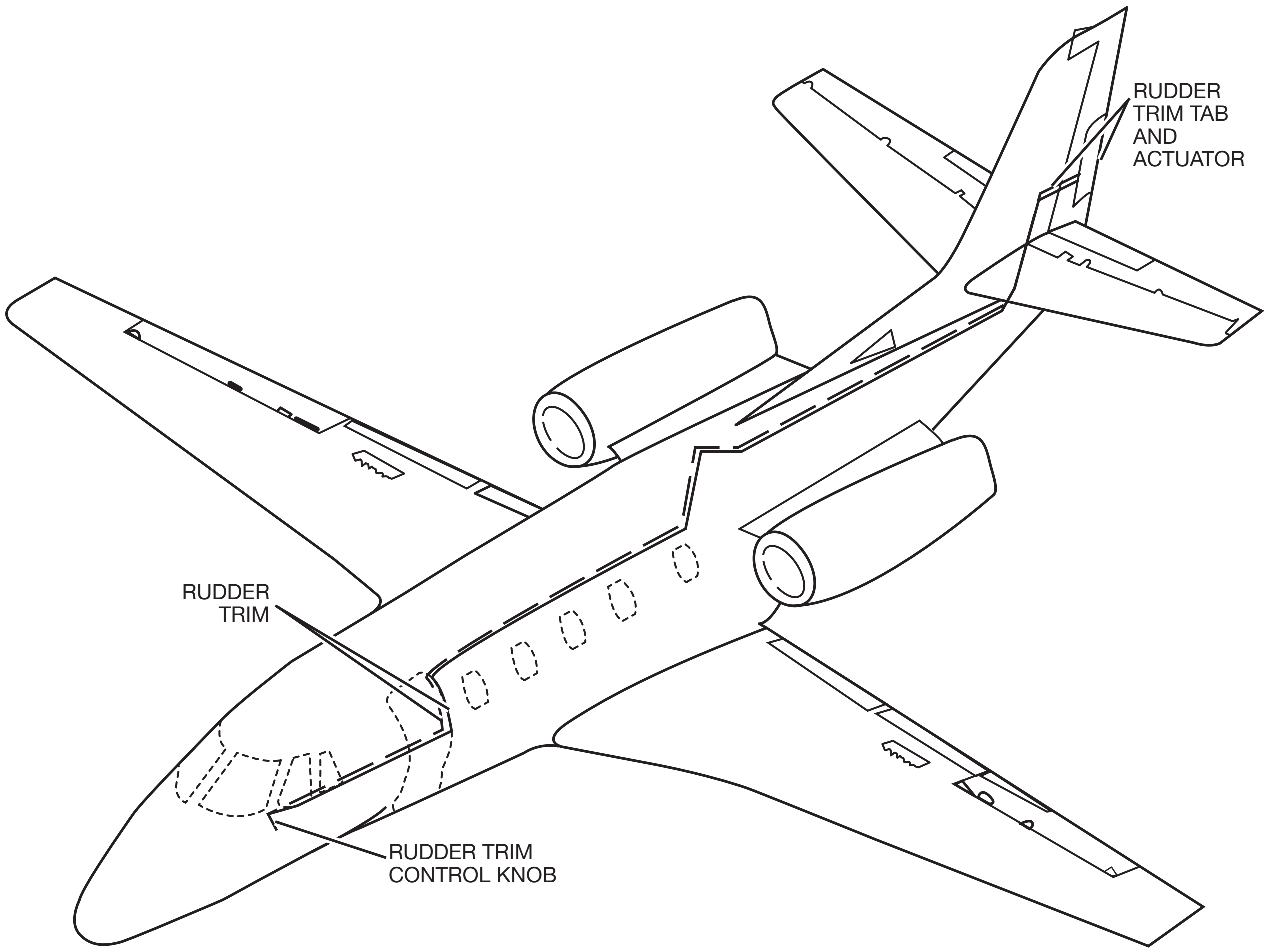


Figure 27-24. Rudder Trim System

RUDDER TRIM SYSTEM

DESCRIPTION

The rudder trim tab system consists of:

- Trim control assembly in the control pedestal
- Cable assemblies
- Rudder trim tab actuator on the trailing edge of the vertical stabilizer
- Trim tab on the trailing edge of the rudder

The rudder trim control cables route aft from the trim control assembly under the cockpit floor panels, through pass-thru seals and out of the pressure vessel. Then they route aft on the lower right side, between the fairing and the fuselage. The cable travels behind the pressure vessel, up to the top of the tail cone where they continue aft over the baggage compartment, and just forward of the 63% bulkhead. They are routed up to the rudder trim tab actuator (Figure 27-24).

OPERATION

Rotate the rudder knob of the trim control assembly on the control pedestal to mechanically actuate the rudder trim control system. Moving the rudder knob repositions the rudder trim tab. This is the primary function of the adjustable trim tab. The secondary function of the rudder trim tab is to serve as a servo boost tab. The rudder trim tab operates as a servo tab which provides a boost to the rudder, when the rudder is not in the neutral position. For each° of rudder deflection, the rudder trim tab (servo function) deflects one-half° in the opposite direction of rudder deflection.

NOTES

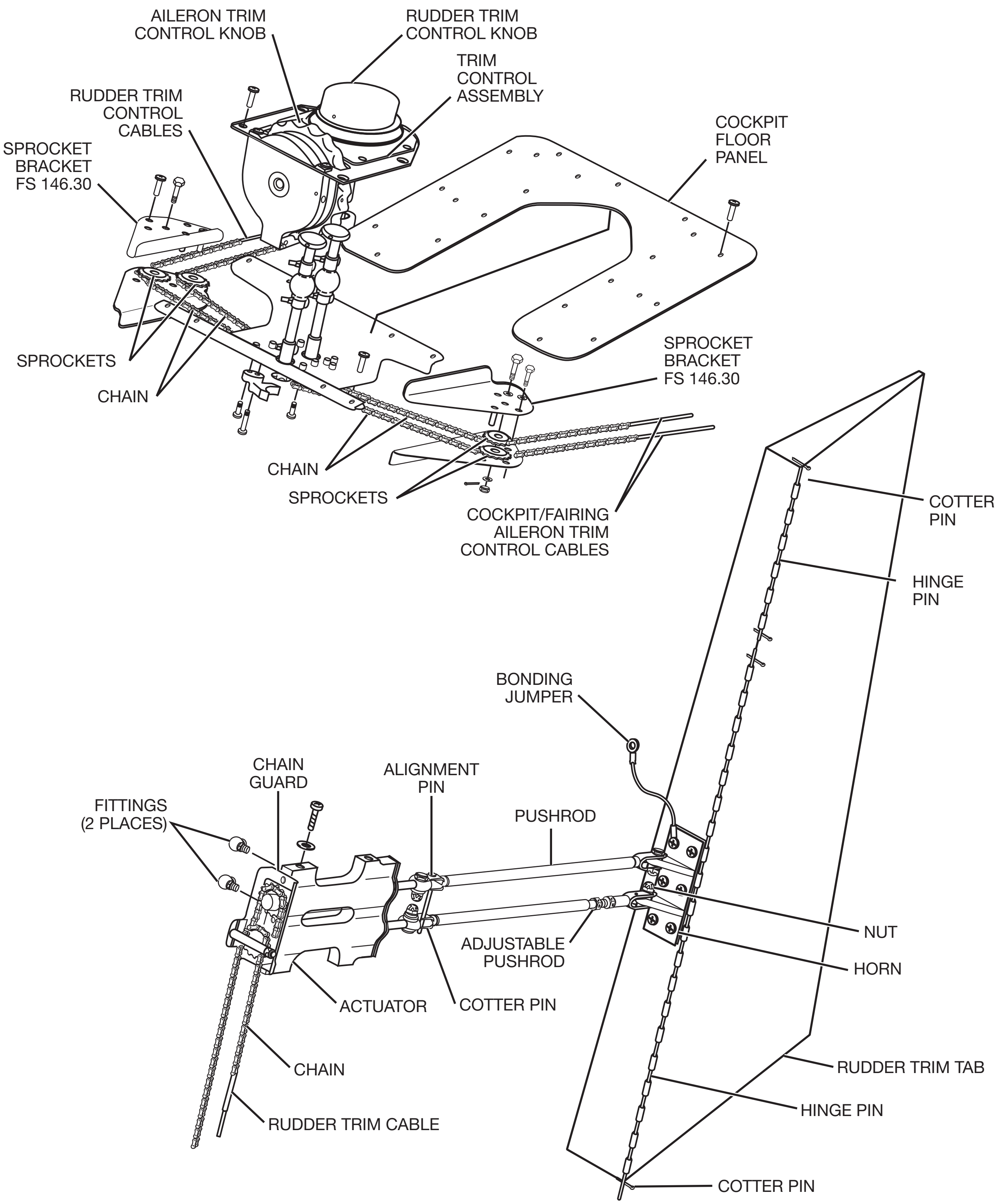


Figure 27-25. Rudder Trim Components

DIAGNOSTICS

Rigging Rudder Trim Control System

1. Remove tail cone baggage compartment access panels to access the rudder trim cable turnbuckles. The trim cable turnbuckles are in the tail cone forward of the baggage compartment (Figure 27-25).
2. Place the rudder pedals in neutral and secure them in neutral position (using tool CJMDL27-004) during rigging of the trim tab.
3. Rotate the rudder trim control knob until the rudder trim tab indicator is centered.

NOTE

Pointer may not be pointing to the center mark.

4. Rotate the rudder trim control knob counter-clockwise until it hits the internal stop.
5. Rotate rudder trim control knob clockwise to the opposite stop while counting the number of rotations.
6. Divide the number of rotations by two.
7. Rotate the knob counterclockwise with amount counted divided by two to establish true center.
8. Check the trim tab actuator pushrods for length. The adjustable pushrod should be the same length as the fixed pushrod.
9. Place a check fixture on the rudder and check the trim tab position. It shall be in the neutral (trail) position. The actuator sprocket can be turned to center the tab.
10. Center the actuator chain on the sprocket, so that there are the same number of links on both sides.

NOTE

Adjust the chain by lifting it over the sprocket. Do not turn the actuator.

11. Check the chain at the outboard sprockets (right side) under the cockpit floor at FS 146.30. There shall be the same number of links aft of the sprockets.

NOTE

Adjust the chain by lifting it over the trim control drive sprocket, do not turn the trim control knob.

12. Place tensiometer on the cables and alternately adjust the corresponding turnbuckles to the specified cable tension.

NOTE

Recheck the trim tab position while tightening the cables to ensure that the actuator has not moved.

13. Rotate the trim tab control knob to FULL NOSE LEFT position (to the right).
14. Rotate the trim tab control knob to FULL NOSE RIGHT, and verify the trim tab deflects the proper travel to the left.
15. Ensure rudder and rudder tab clearance with the rudder at full left and tab full right, then full right and tab full left.
16. Return the tab to the streamline position, and center the trim pointer by removing the trim knob. Lift up the pointer disk. Center the pointer; then set the disk back down.
17. Reinstall trim knob.
18. Safety check the turnbuckles.
19. Remove check fixture.

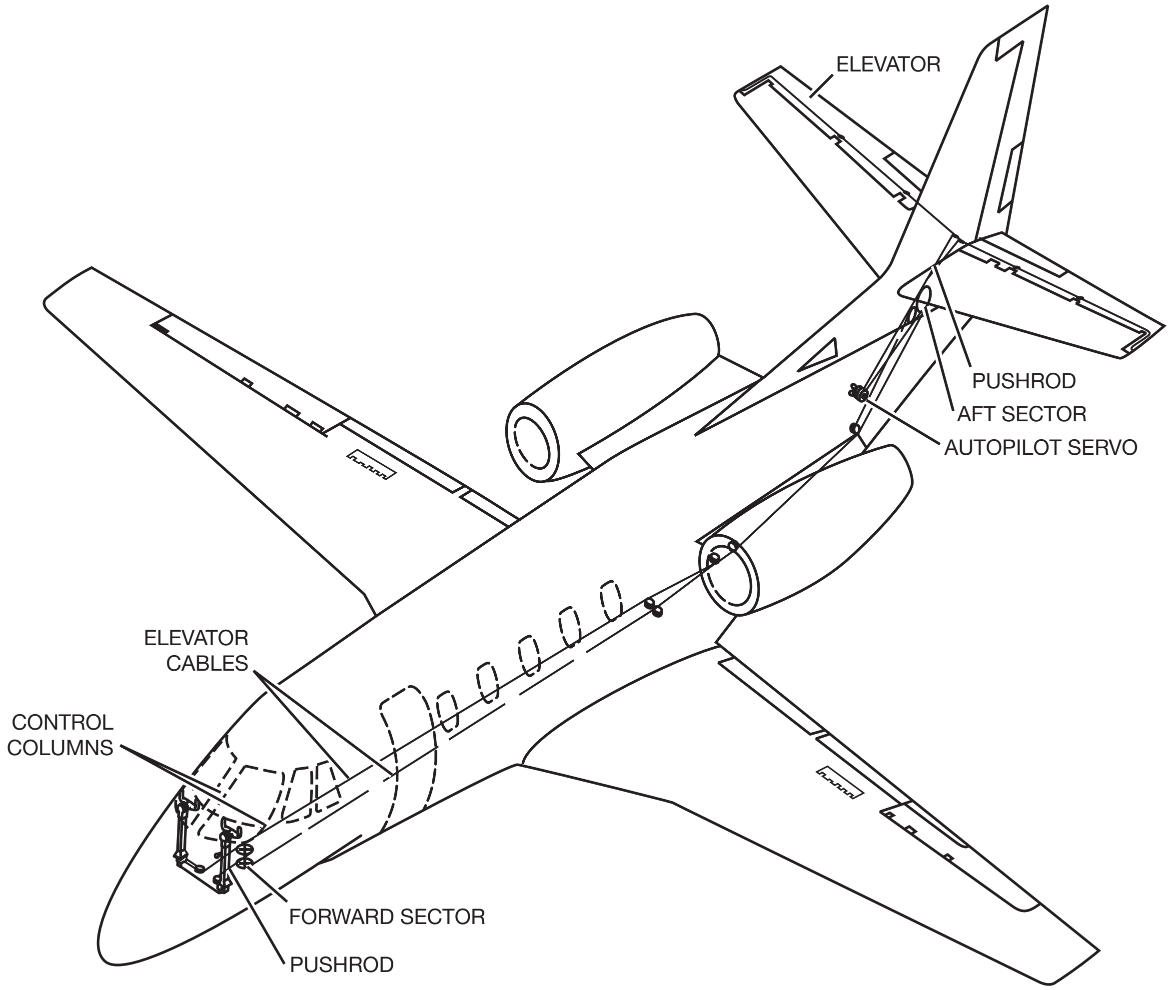


Figure 27-26. Elevator Control System

ELEVATOR SYSTEM

DESCRIPTION

The elevator system provides longitudinal control of the aircraft (Figure 27-26). The elevators are mechanically actuated by moving the control column or electrically by the autopilot servo.

The elevator system consists of:

- Control column assembly
- Pass-thru sector assembly below the cockpit floor
- Cable assemblies
- Bellcrank assembly in the aft section of the tail cone
- Elevators on the trailing edge of the horizontal stabilizer

The elevator bellcrank also provides attach points for the autopilot elevator servo cables.

OPERATION

To mechanical actuate the elevator control system move the control column fore and aft. A torque tube assembly connects the left and right control column. Control column movement transmits to the elevator through the pass-thru sector assembly and bellcrank assembly via cable assemblies.

Electrical actuation of the elevator control system is accomplished when the autopilot elevator servo cables apply a force on the elevator bellcrank deflecting the elevators. The autopilot elevator servo has an override function, which means the operator can physically overpower the servo by manually moving the control column.

NOTES

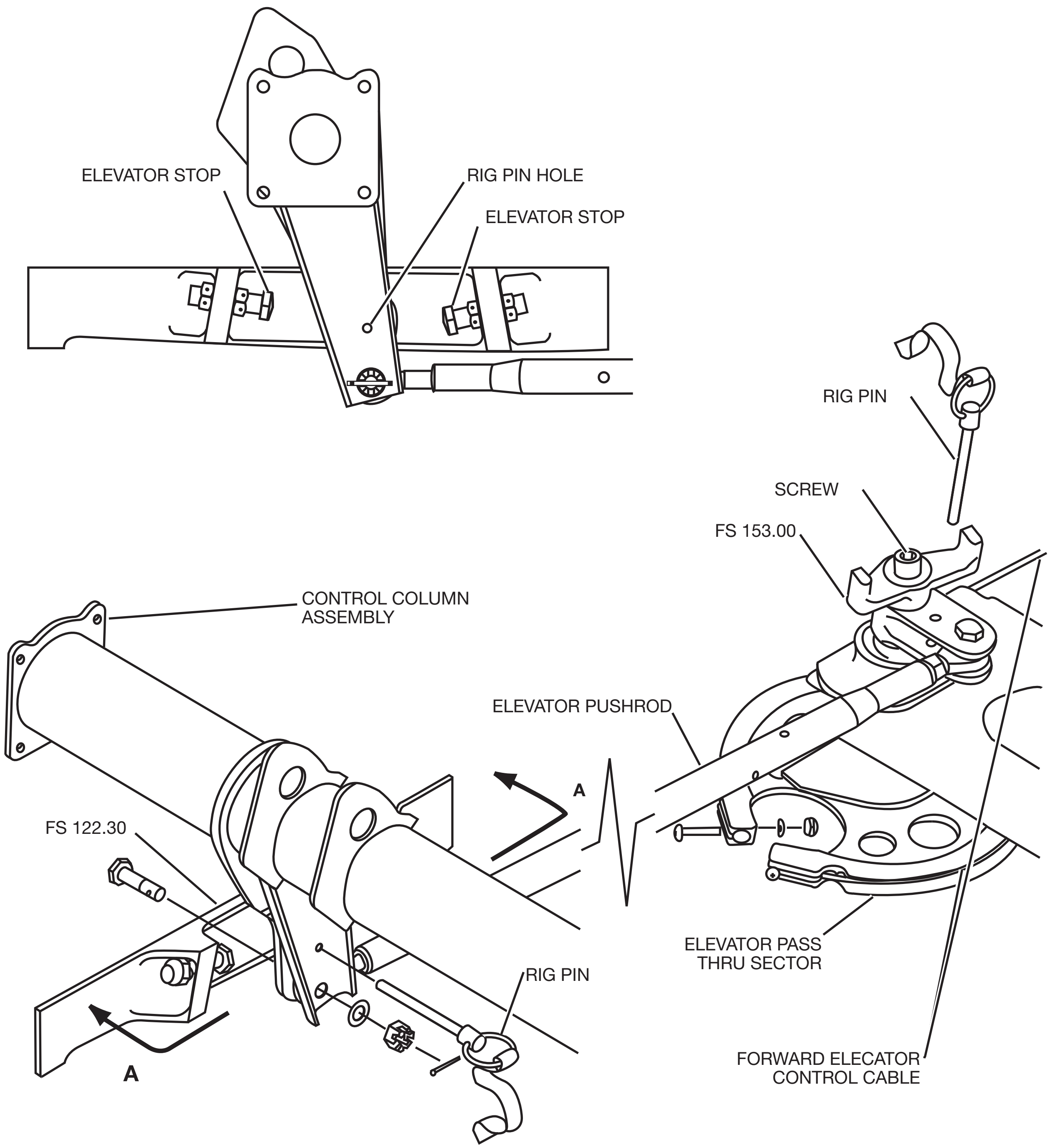


Figure 27-27. Elevator System in the Cockpit

DIAGNOSTICS

Rigging Cockpit Elevator System

1. Remove the flight crew seats and cockpit floor panel to access the elevator pushrod between column and pass-thru sector (Figure 27-27).
2. Place the elevator pass-thru sector assembly in neutral position and insert the rig pin.
3. Disconnect the pushrod (between column and pass-thru sector). Then place the control column in neutral and install the rig pin. Adjust the pushrod to length and install it using a bolt, washer, nut, and cotter pin

NOTE

Rig pin holes are on the column output arm and stop block assembly on the copilot inboard seat rail beam.

4. Remove the rig pins from the elevator sector and control column.

Cockpit Elevator Stop Bolt Adjustment

1. Verify that stops in tail are adjusted properly before checking or adjusting cockpit elevator stops.
2. Adjust the control column stop bolts to maintain 0.10 inch between the control column stop and stop bolts with the aft sector resting first on the upper stop, then on the lower stop.
3. Safety control the column stop bolts.
4. Reinstall the cockpit floor panels and crew seats.

NOTES

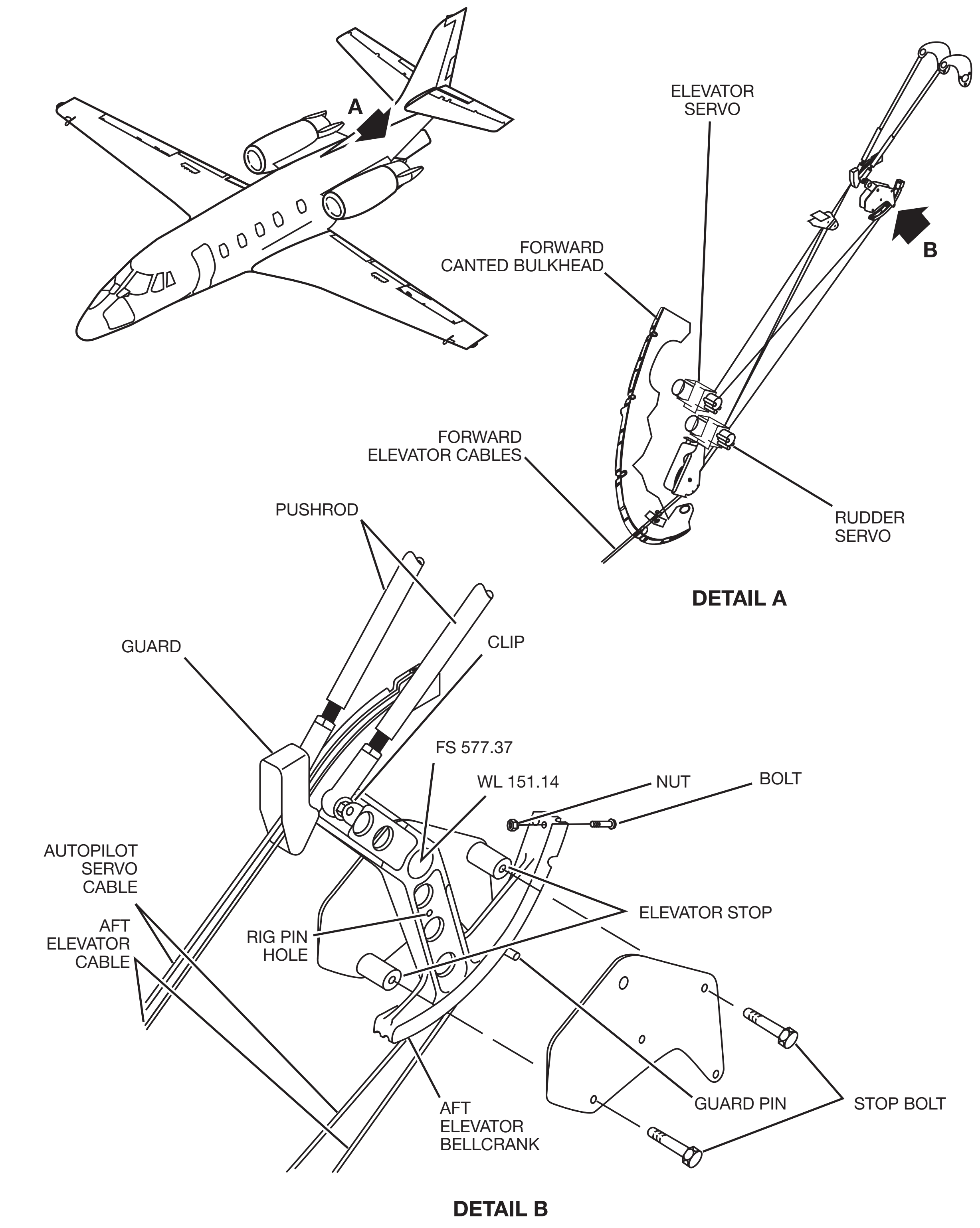


Figure 27-28. Aft Elevator Bellcrank Assembly

Rigging Elevator Cables

1. Remove the flight crew seats and cockpit floor panel access elevator pass-thru sector.
2. Remove the fuselage fairing access panels to gain access to the elevator cables (Figure 27-28).
3. Remove aft tail cone access door to gain access to the elevator cables and autopilot servo.
4. Remove the vertical stabilizer access panels to gain access to the elevator pushrods.
5. Gain access to elevator cable turnbuckles through the forward tail cone access door.

NOTE

Elevator cable turnbuckles are along the lower tail cone (on top of cable tray) inside the forward tail cone access door.

6. Disconnect the elevator down spring from the bellcrank.

CAUTION

Failure to disconnect elevator down spring produces a preload condition in cable tension and erroneous rigging results.

7. Place the elevator bellcrank in neutral position and insert the rig pin. Place the elevator pass-thru sector assembly in neutral position, and insert the rig pin.

NOTE

It may be necessary to loosen cables to allow rig pin to be inserted into both the bellcrank and pass-thru sector assembly.

8. Place a tensiometer on the corresponding elevator control cable then alternately adjust the turnbuckles to the specified cable tension.
9. Adjust the elevator autopilot servo cables to the specified cable tension.
10. Connect the elevator down spring to bellcrank.
11. Install turnbuckle clips.

Aft Elevator Pushrod Adjustment

1. Remove the cotter pins, nuts, washers and bolts connecting elevator pushrods to elevator bellcrank.
2. Position the horizontal stabilizer to the +1° cruise position.
3. Position check fixtures (CJMDL27-002 left or CJMDL27-006 right) on horizontal stabilizer to set elevator to 0° position.
4. Adjust the elevator pushrods to the correct length, between the elevator bellcrank and elevator horns, to achieve elevator zero. Reconnect the pushrods to the elevator bellcrank, then install the bolts, washers, nuts and cotter pins.
5. Remove the rig pins from the elevator bellcrank, and pass-thru sector.

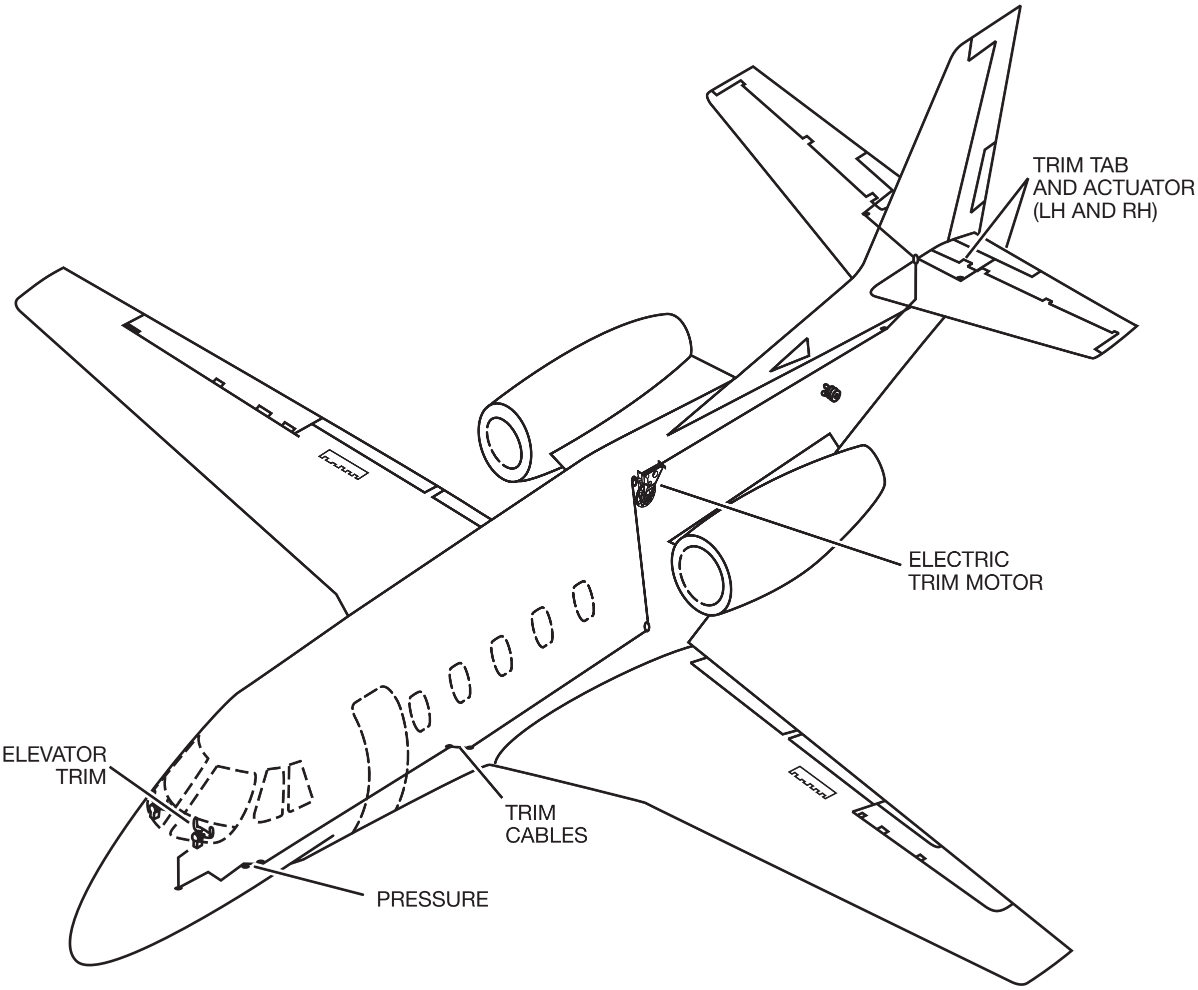


Figure 27-29. Elevator Trim System

Aft Elevator Stop Adjustment

1. Position digital inclinometers on the elevator at SS 25.90 and SS 61.00.
2. Position the check fixtures (CJMDL27-002 left or CJMDL27-006 right) on the horizontal stabilizer to set the elevator to 0° position.
3. Loosen the bolt on the elevator down-travel stop and rotate the stop until proper elevator deflection is indicated on inclinometer. Secure the stop.

NOTE

The bellcrank stop adjustment is made with the bellcrank resting against the stop.

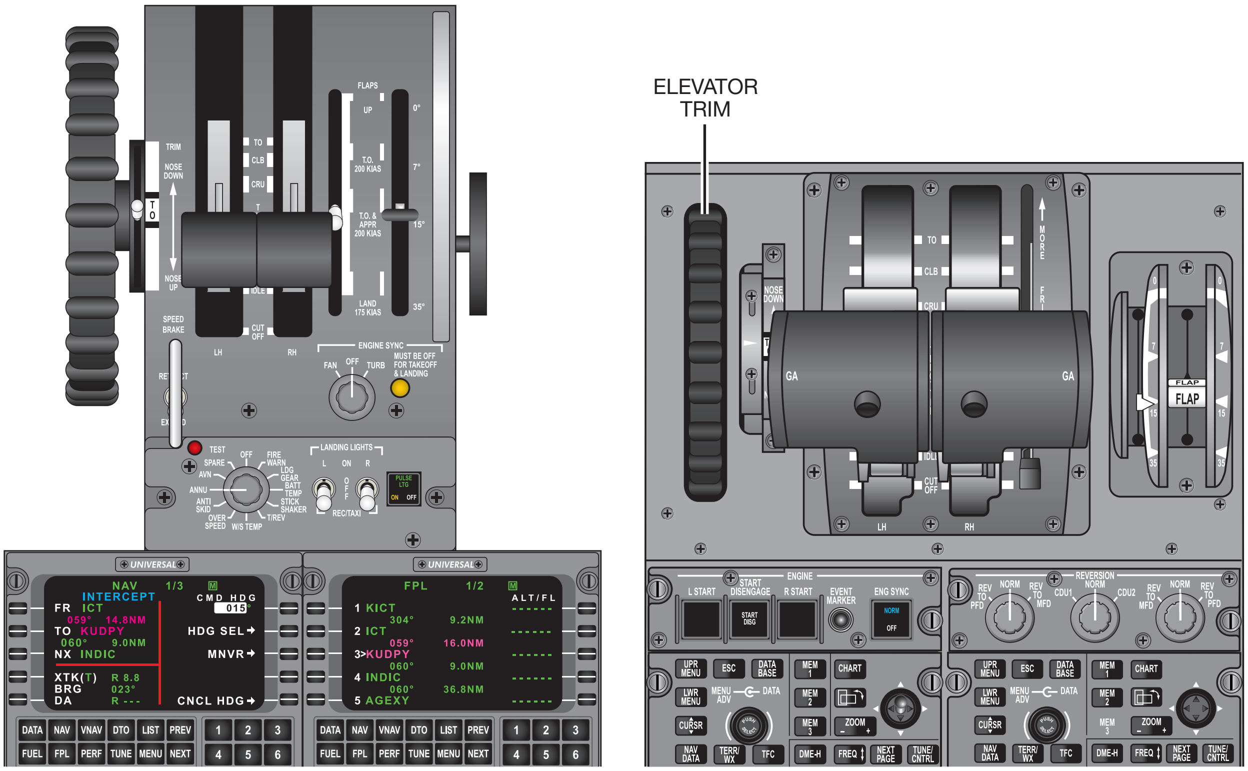
4. Loosen the bolt on the elevator-up travel stop and rotate the stop until the proper elevator deflection is indicated on the inclinometer. Secure the stop.
5. Reinstall the cockpit floor panels, fuselage fairing panels, tail cone access panel and vertical stabilizer access panels.
6. Reinstall the flight crew seats.

ELEVATOR TRIM SYSTEM

DESCRIPTION

The elevator trim system consists of five cable assemblies and one electric trim cable assembly (Figure 27-29). These cables route from the trim control wheel (on the left side of the control pedestal) down and aft, below the cockpit floor; then through the fuselage pass-thru seals, and out of the fuselage (on the lower left side). They continue aft between the fuselage and fairing, and just behind the pressure vessel. They route up into the upper tail cone where they connect to the aft elevator control cables. At this point, the electric trim cables connect and all three turnbuckles are located in the system (above the baggage compartment.) They continue upward to the trim tab actuators that are interconnected with chains and a crossover cable in the horizontal stabilizer.

NOTES



XL/XLS XLS+

Figure 27-30. Manual Trim Wheels

Figure 27-31. Pitch Trim and AP TRIM DISC Switches

OPERATION

The elevator trim control system is mechanically actuated by rotating the elevator trim control wheel on the control pedestal. Moving the trim control wheel repositions the elevator trim tabs. Cable assemblies transmit movement between the trim control wheel and the trim tab actuators, rotating the actuator screws, which extend or retract to deflect the trim tabs (Figure 27-30).

The elevator trim control system actuates electronically by an electric trim tab actuator. The electric trim tab actuator actuates via the pilot or copilot control wheel trim switches, or via autopilot input (monitored by the electric trim logic module assembly). Selecting up or down position on the trim switches on the control wheel or autopilot trim inputs engages the electric motor on the electric trim tab actuator to drive the trim tabs in the appropriate direction. The electric trim tab actuator moves the elevator trim cables, that in turn rotate the trim tab actuator screws. The actuator screws extend or retract to deflect the trim tabs (Figure 27-31).

NOTES

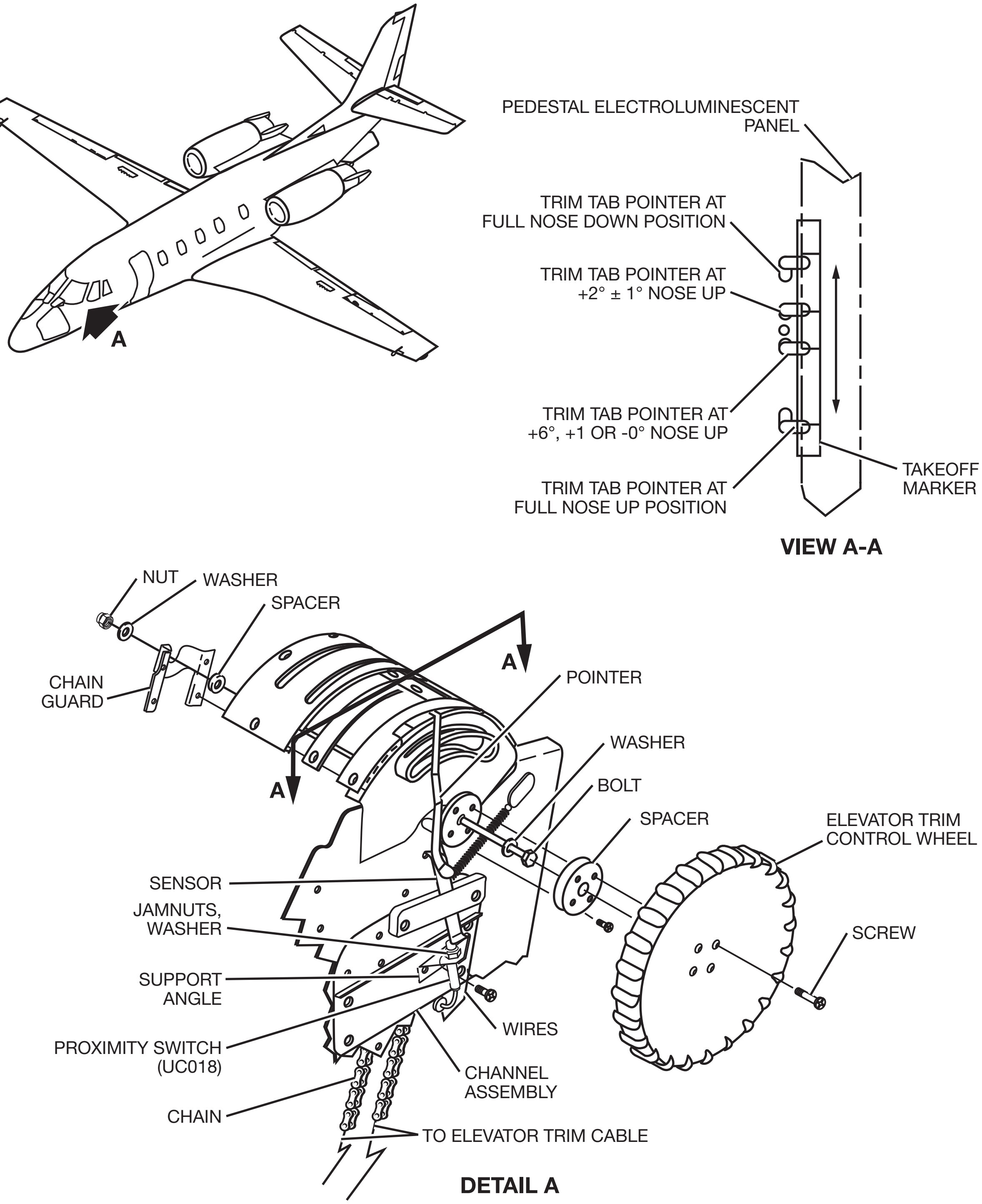


Figure 27-32. Elevator Trim Control and Indication

COMPONENTS

Elevator Trim Tab Control Wheel

The elevator trim tab control wheel is on the left side of the control pedestal. The wheel moves a chain that connects to the control cables to move the trim tabs. A continuous spiral groove makes nine turns around the back of the control wheel controlling the trim indicating pointer (Figure 27-32).

Electric Trim Actuator

The electric elevator trim actuator provides the pilot with electrical control of the elevator trim tab. It is in the upper tail cone and mounts left of centerline to a pulley bracket at FS 389.50. The actuator operates to drive the elevator trim tabs on a command signal from the electric trim switches or autopilot.

Trim Tab Actuator

The elevator trim tab actuator has two screws in a single housing. Each screw is connected to the trim tab by a pushrod. The trim tab actuator cables connect to a chain. The chain rotates the primary sprocket to drive one screw. The two screws operate together via an interconnect chain and secondary sprockets. Zerk fittings are installed in the housing for screw lubrication.

Trim Tab

The elevator trim tab is a movable airfoil on the inboard trailing edge of each elevator. The elevator trim tab actuator is in the horizontal stabilizer with pushrods extending through the elevator to the trim tab.

DIAGNOSTICS

Elevator Trim No Takeoff Warning

The elevator trim "no takeoff" warning system makes use of a proximity switch to monitor the trim indicator position. Correct operation can be verified as follows:

1. Move the elevator trim tabs to neutral when the trailing edge of the trim tab is streamlined with the trailing edge of the elevator.
2. Install an inclinometer on the tab. Zero the inclinometer if needed.
3. With electrical power applied, move the elevator trim in the "nose up" direction until the inclinometer reads +2°.
4. Verify that trim pointer is at the bottom line of the take off range; and verify that the "no take off" light is extinguished. If not, adjust per *AMM*.
5. Move the elevator trim toward the nose up direction until the inclinometer reads +6°. Verify that the pointer is at the upper line of the take off range; and verify that the "no take off" annunciator is extinguished.
6. Move the elevator trim toward the nose up direction and verify that the "no take off" light illuminates when the pointer is beyond the +6° travel of the trim tab.
7. Move the elevator trim toward the nose down direction and verify that the "no take off" light illuminates below the +2° position, as indicated by the inclinometer on the elevator trim tab.

NOTE

The light should be extinguished between the +2° to +6° take off range.

8. Remove the inclinometer and power from the aircraft.

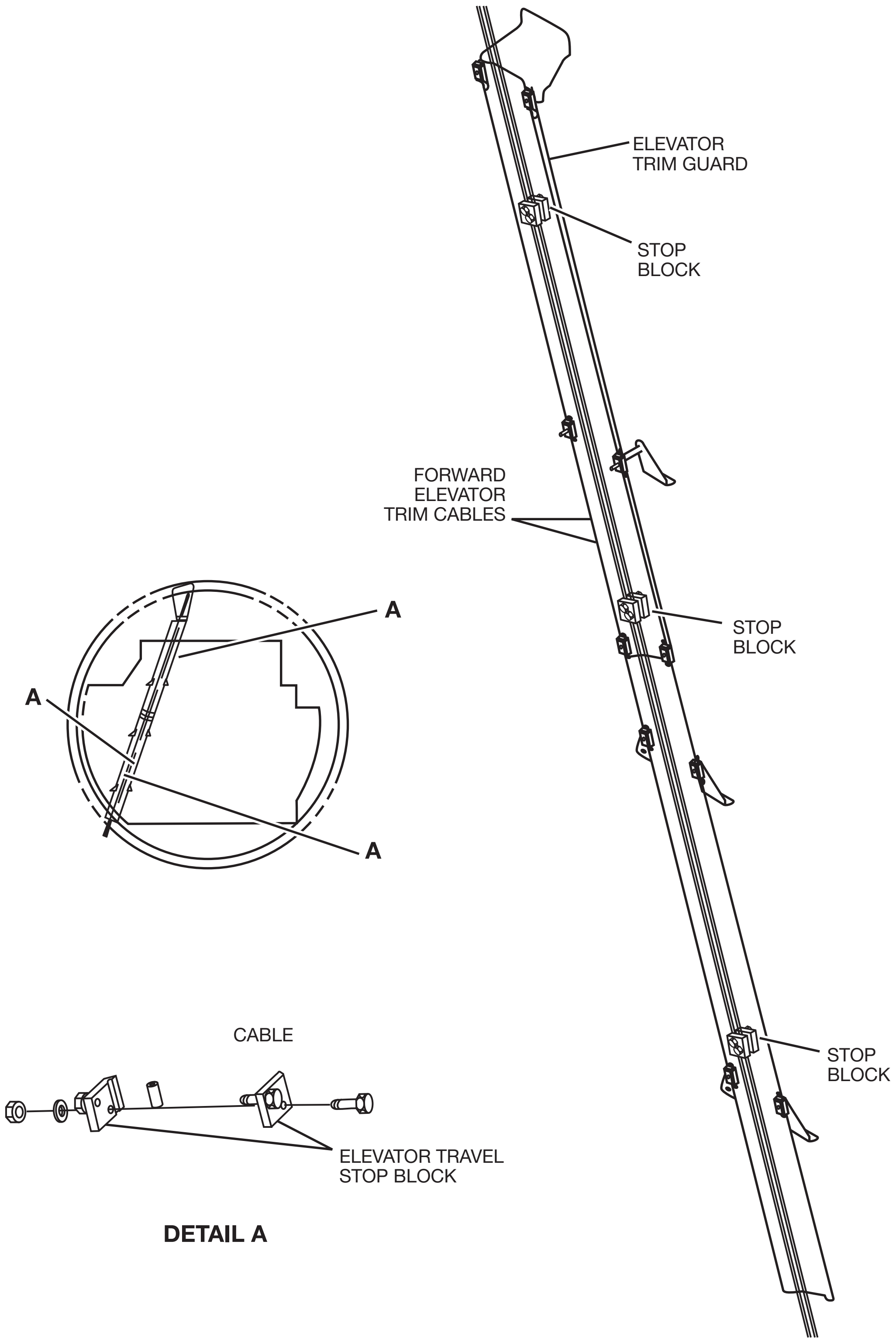


Figure 27-33. Elevator Travel Stop Blocks

Elevator Trim Tab Travel Adjustment

There are trim cable stop blocks on the trim cables forward of the elevator trim guard assembly at FS 387. 54 (forward of the tail cone baggage compartment forward panel on left side) (Figure 27-33).

1. With the tab streamlined, clamp the center stop block to the vertical portion of the outboard elevator trim cable forward of the baggage compartment. The center of the stop block should be 27.5 inches, ±0.5 inches above the baggage compartment floor.
2. Turn the trim wheel towards NOSE UP until the trim tabs are 15° ± 1°, trailing edge down. The center stop block should have moved UP about 7.5 inches (190 mm). Clamp the upper stop block to the inboard cable above, and in contact with, the center stop block.
3. Turn the trim wheel towards NOSE DOWN until the trim tabs are 5° ± 1° trailing edge up. The center stop block should be about 30 inches (762 mm) below the upper stop block. Clamp the lower stop block to the inboard cable below, and in contact with, the center stop block. Cycle the system. Observe the reaction of the stop blocks on the cable, to account for cable twist and to obtain the maximum flush contact between blocks.

NOTES

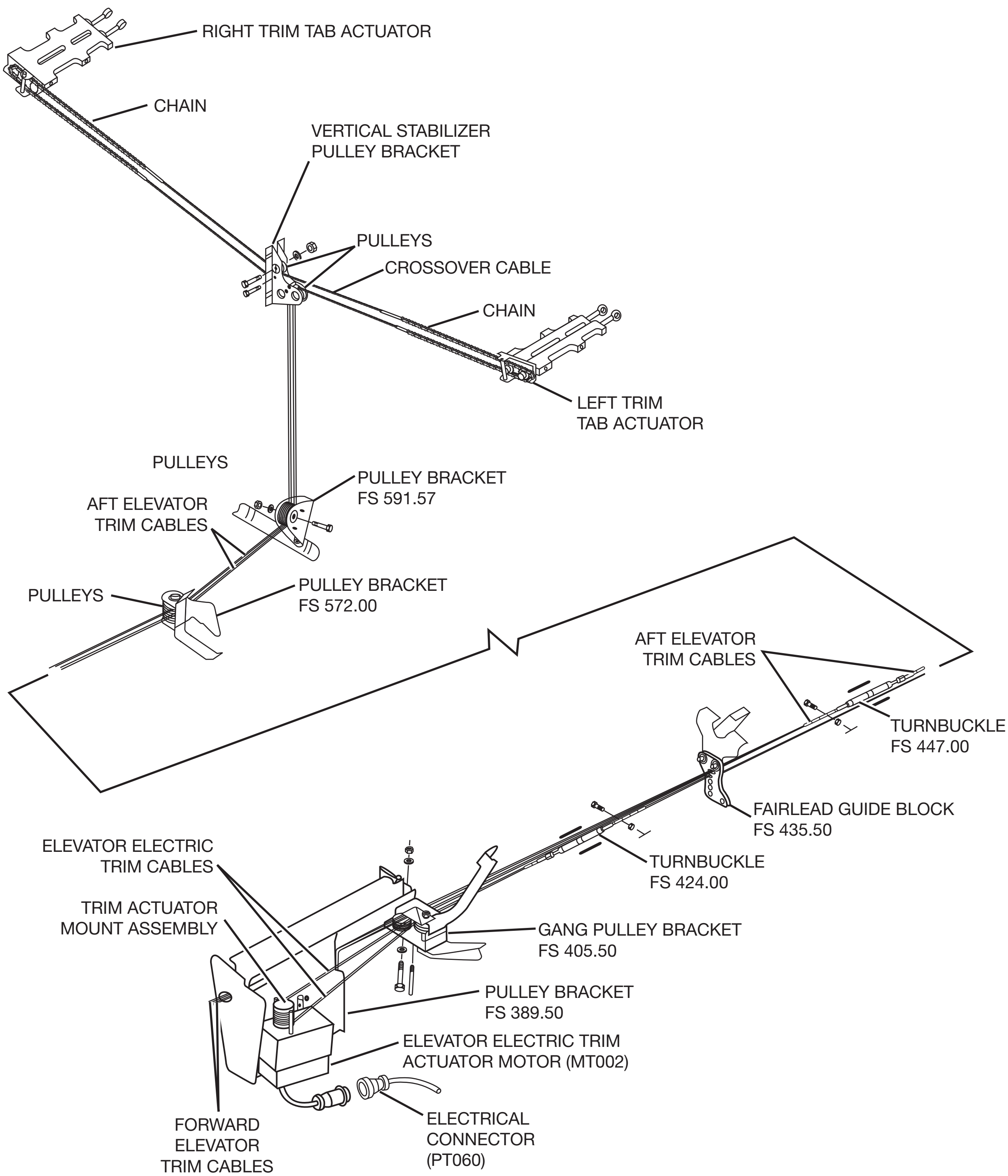


Figure 27-34. Elevator Electric Trim and Tab Actuators

Functional Test

A slide switch on the left grip of the pilot control wheel electrically controls elevator trim. Trim override is provided by momentarily depressing the autopilot/trim disengage button on the pilot left control wheel grip.

1. Adjust the ground power unit to supply 28.5 volts to the aircraft.
2. On the left CB panel, ensure the PITCH TRIM circuit breaker is engaged.
3. Pulling the trim control switches forward shall cause the elevator trim tab to move upward (nose down trim) (Figure 27-34). Drive the tab to the upper limit.
4. Pulling the trim control switches aft shall cause the elevator trim tab to move downward (nose up trim). Drive the tab to the lower limit. The elevator trim control wheel shall drive three revolutions out of the nose down limit in 39.0 to 48.0 seconds.
5. Verify that the electric trim drives the elevator trim control wheel three revolutions (out of the nose up limit) in 39.0 to 48.0 seconds.
6. Hold the trim switches in the DOWN position. Momentarily depress the autopilot disengage button. The trim motor should stop until the trim switches are returned to neutral and pushed DOWN again.
7. Hold the trim switch in the UP position. Momentarily depress the autopilot disengage button. The trim motor should stop until the trim switches are returned to neutral and pushed UP again.
8. Verify the pilot side control switches have priority over the copilot side switches by pulling the copilot trim control switches aft. The trim tab should move downward (nose up). While holding the copilot switches, push the pilot trim control switches forward. The trim tab should move upward (nose down). Repeat, moving the trim control switches in the opposite direction.

NOTES

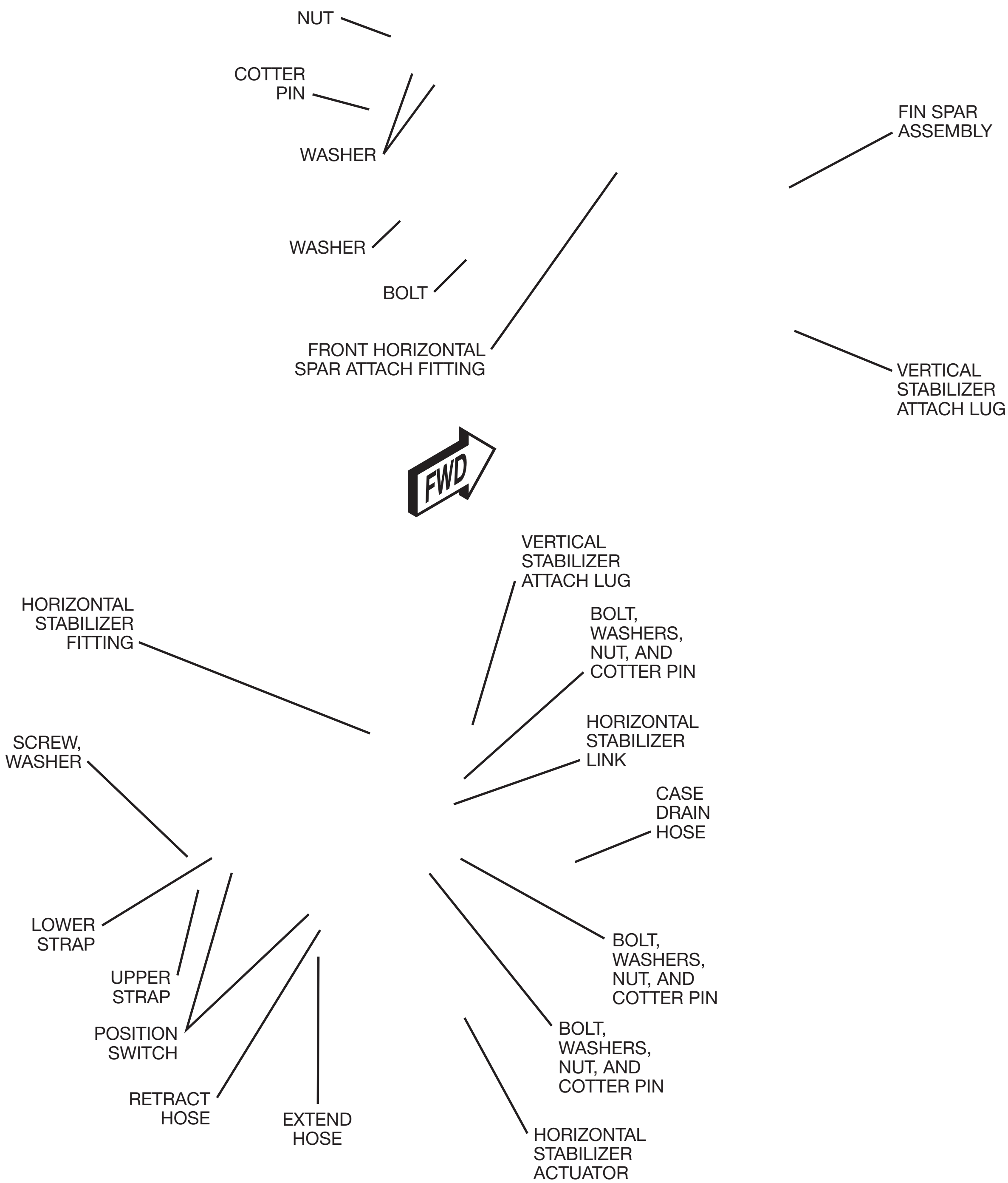


Figure 27-35. Two Position Horizontal Stabilizer System

HORIZONTAL STABILIZER

DESCRIPTION

The two-position horizontal stabilizer system automatically repositions the aircraft horizontal stabilizer to improve flight characteristics. The horizontal stabilizer positions to one of two positions: a +1° (cruise) or –2° (takeoff). The angle of incidence position depends on the flap handle position and airspeed by moving the entire horizontal stabilizer (Figure 27-35). When airspeed is greater than 215 knots ±10, the airspeed switch (XL/XLS) disables the arming valve preventing stabilizer movement to the –2° position. The XLS+ stabilizer is inhibited by a discrete input from the ADC at airspeeds greater than 215 ± 10 knots.

COMPONENTS

Stabilizer Actuator

The actuator is a self-contained unit consisting of:

- Valve body
- Hydro-mechanical motor
- Gearbox
- Screw assembly

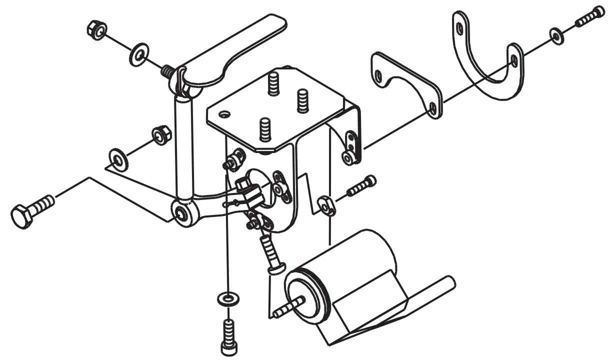
It is in the lower vertical stabilizer, below the horizontal stabilizer. The actuator is suspended under an attach lug assembly that is connected to the forward vertical stabilizer spar (at its top mounting point) and to the vertical stabilizer rib (lower mounting point). The actuator jackscrews attach to fittings on the forward spar of the horizontal stabilizer.

Position Switches

The XL utilizes two sets of position switches are in brackets below horizontal stabilizer rib. The switches connect to the horizontal stabilizer by arms and pushrods. Each set contains three individually adjustable switches known as decks of switches (Figures 27-36 through 27-38). On the left set:

- Deck A is the up-limit
- Deck B is the down limit
- Deck C is not used

On the right set, neither Deck A nor B is used; while Deck C is the “No Takeoff” indication from the horizontal tail.r



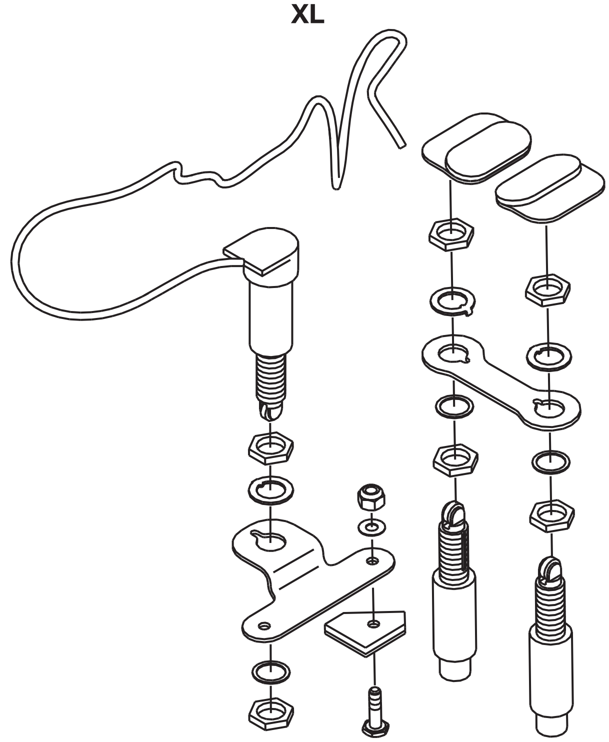


Figure 27-36. Stabilizer Position Switches

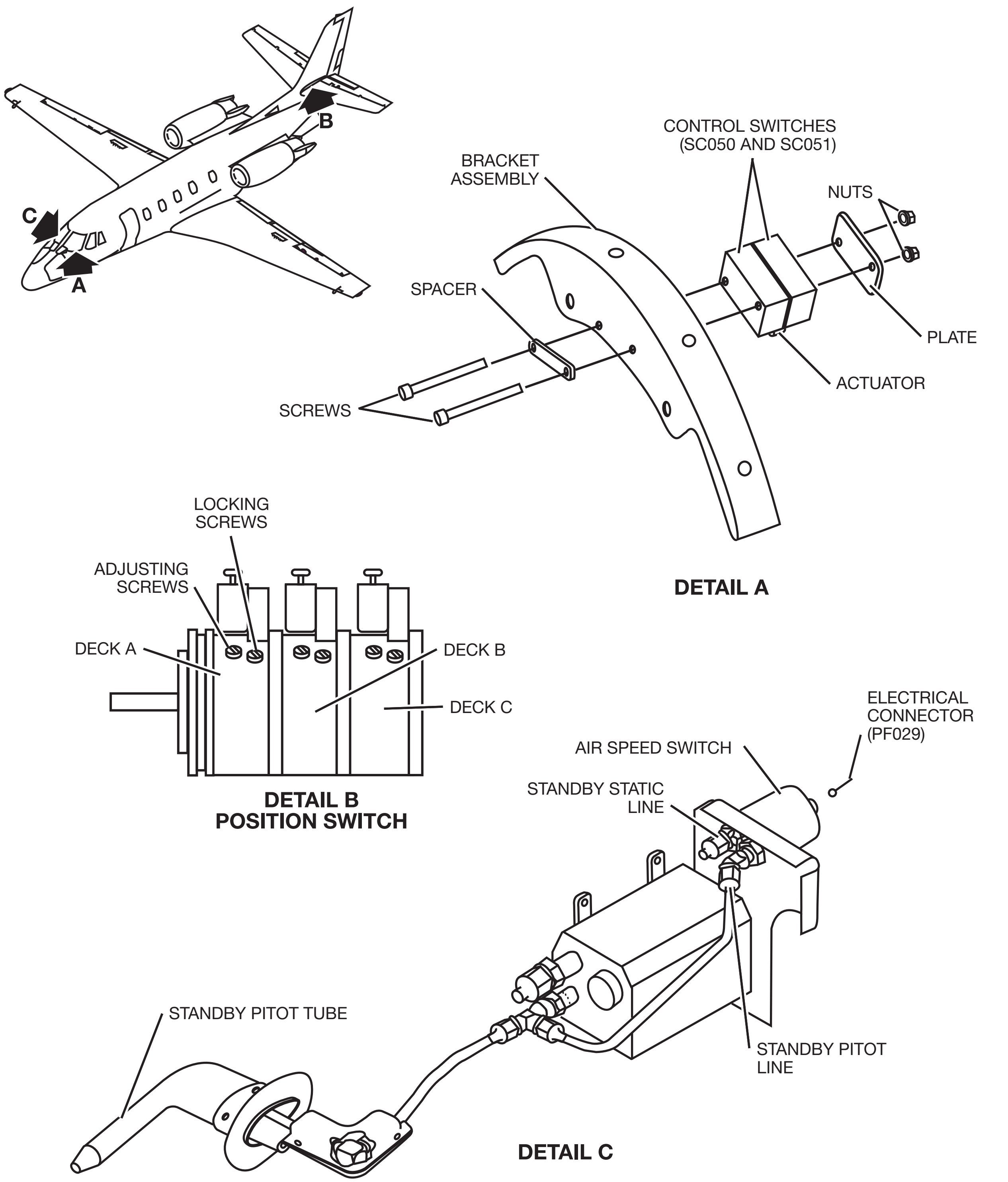


Figure 27-37. Horizontal Stabilizer Electrical Components (XL/XLS)

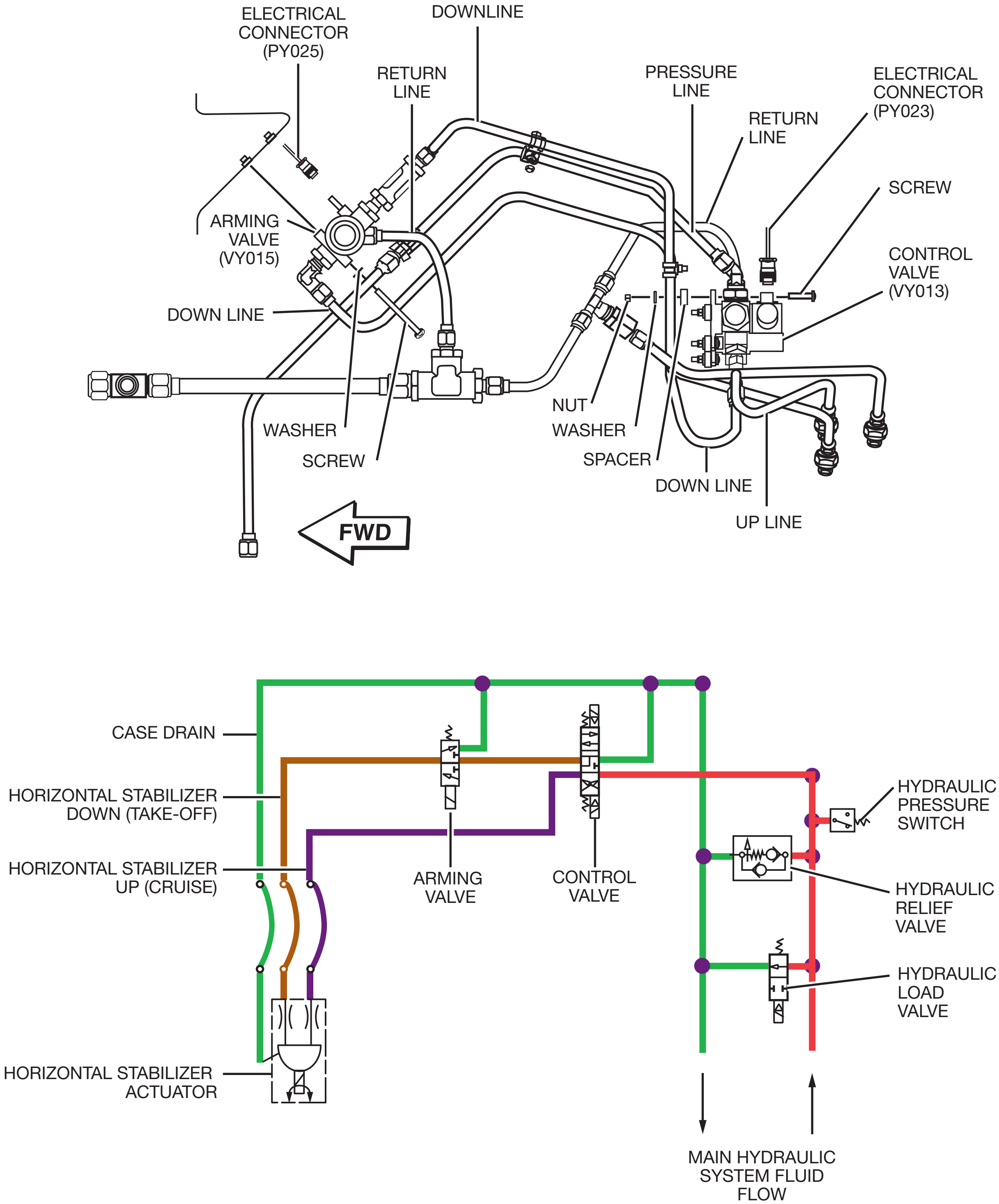


Figure 27-38. Horizontal Stabilizer

Airspeed Switch (XL/XLS)

The airspeed switch senses airspeed from the standby pitot static system and enables or disables the horizontal tail from downward movement towards the takeoff and approach position or upward movement towards the cruise position—based upon the airspeed sensed. The horizontal tail is enabled if airspeed is less than 215 ±10 knots; or disabled it if airspeed is greater than 215 ±10 knots. It is behind the copilot side panel, above the armrest.

Control Switches

There are two control switches on the right side of the throttle quadrant. The switches actuate simultaneously by a cam attached to the flap handle. The only time the switches are not actuated is when the flap handle is in the full up position (0°).

Stabilizer Control Valve

The stabilizer control valve routes hydraulic pressure to the extend or retract port of the horizontal stabilizer actuator. It is accessible through the most aft access panel (313EC) on the lower fuselage fairing.

Arming Valve

The arming valve prevents horizontal stabilizer movement down towards the takeoff and approach position (if airspeed is greater than 215 ± 10 kts and the flap handle is moved out of the 0° flap position). If airspeed is less than 215 ± 10 kts and the flap handle is moved out of the 0° flap position, then the arming valve is armed (energized); and allows hydraulic pressure to proceed to the retract port of the actuator.

CONTROLS AND INDICATIONS

Stabilizer Monitoring System

The two-position horizontal stabilizer control system is controlled by a flap-handle position and airspeed. With the flap-handle in the FLAPS UP detent position the horizontal stabilizer has an incidence of +1. With the flap handle in any position other than the FLAPS UP detent and the airspeed no greater than 215 ± 10 kts, the horizontal stabilizer has an incidence of –2. The horizontal stabilizer cannot move down to an incidence of –2° if the airspeed is greater than 215 ± 10 kts. It is prevented from moving in either direction if the landing gear is in motion.

STABILIZER MISCOMPARE
Steady illumination occurs on the ground if the horizontal stabilizer does not agree with the flap handle position within 30 seconds. This condition contributes to the NO TAKEOFF annunciation.

Flashing annunciation in flight indicates:

1)The horizontal stabilizer does not agree with the flap handle within 30 seconds, or

2)The aircraft has exceeded 200 KIAS after takeoff with the flap handle greater than 0°.

XL/XLS ANNUNCIATOR

STAB MISCOMPARE			
Color	**Inhibited By**		**Debounce**
Amber	LOPI		Standard

The two position tail PCB will set the Stab Position Master Caution discrete for the following conditions:

1. If the stab position does not reach the up position within 32 ± 3 seconds after flaps retracted, or within 42 ± 3 seconds of landing gear operation.
2. If the stab is moving at airspeeds greater than 215 Kts.

-OR-

1. If the flap handle switches indicate flaps up and flaps down simultaneously.
2. If the stab position does not reach the up position within 32 ± 3 seconds after flaps retracted, or within 42 ± 3 seconds of landing gear operation.
3. If the stab is moving at airspeeds greater than 215 Kts.
4. If the stab position does not reach the down position within 32 ± 3 seconds after flaps are moved out of the 0° position or within 42 ± 3 seconds of landing gear operation.

XLS+ CAS MESSAGE

Figure 27-39. Stabilizer Miscompare Indications

The two-position tail printed circuit board (N2017) monitors the horizontal stabilizer position. The PCB flashes the STAB MIS COMP annunciator (XL/XLS) or amber STAB MISCOMPARE CAS message (XLS+) and illuminates the MASTER CAUTION RESET switchlight under the following conditions (Figure 27-39):

- Anytime the flap handle is not in the "FLAPS UP" detent position and the stabilizer has not reached the incidence of –2° within the predetermined time limit of 30 seconds (XL/XLS) or 32 seconds (XLS+). Annunciation is extended to 40 seconds (XL/XLS) or 42 seconds (XLS+) if the landing gear is actuated simultaneously.
- Anytime the flap handle is in the "FLAPS UP" detent position and the stabilizer has not reached the incidence of +1 within the predetermined time limit of 30 seconds (XL/XLS) or 32 seconds (XLS+). Annunciation is extended to 40 seconds (XL/XLS) or 42 seconds (XLS+) if the landing gear is actuated simultaneously.
- Anytime the PCB senses flap handle selected up and flap-handle is selected down concurrently.

No Takeoff Warning System

Deck C of the right set of position switches is connected to the "no takeoff" warning system. The switch is rigged to detect when the stabilizer is in the takeoff and approach position. At any time the aircraft is on the ground and the stabilizer is not in the takeoff and approach position, the NO TAKEOFF annunciator (XL/XLS) or white NO TAKEOFF CAS message (XLS+) illuminates (Figure 27-40). If both throttles are advanced beyond 54° TLA (XL/XLS) or 43° TLA (XLS+) an aural warning is triggered. Additionally, the white NO TAKEOFF CAS message on the XLS+ turns red and flashes the MASTER WARNING if these conditions are met. This system is completely independent of the stabilizer monitoring system.

NO TAKEOFF ON GROUND, Illuminates steady to indicate one or more of the following: Flaps are <7° or >15°, elevator is out of trim for takeoff, horizontal stabilizer is out of the takeoff position (STAB MISCOMP), and/or the speed brakes are not completely stowed (the parking brake also contributes to the NO TAKEOFF condition on certain European registered aircraft). Advancing power beyond approximately 80% N_1 with any of the above conditions existing, will activate the MASTER CAUTION lights and an aural warning sound.

XL/XLS ANNUNCIATOR

NO TAKEOFF			
Color	**Inhibited By**		**Debounce**
Red	LOPI	In Air	Standard
White			

On the ground, the white NO TAKEOFF message will illuminate if one or more of the following conditions exist:
- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position
- **Speed Brakes** are out of takeoff position

NO TAKEOFF			
Color	**Inhibited By**		**Debounce**
Red	LOPI	In Air	Standard
White			

As the throttles are advanced beyond 43° TLA, airspeed less than 67 knots, and thrust reversers not deployed, the red NO TAKEOFF message will illuminate if one or more the following conditions exist:
- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position

The red message also produces a voice aural "No Takeoff".

XLS+ CAS MESSAGES

Figure 27-40. No Takeoff Indications

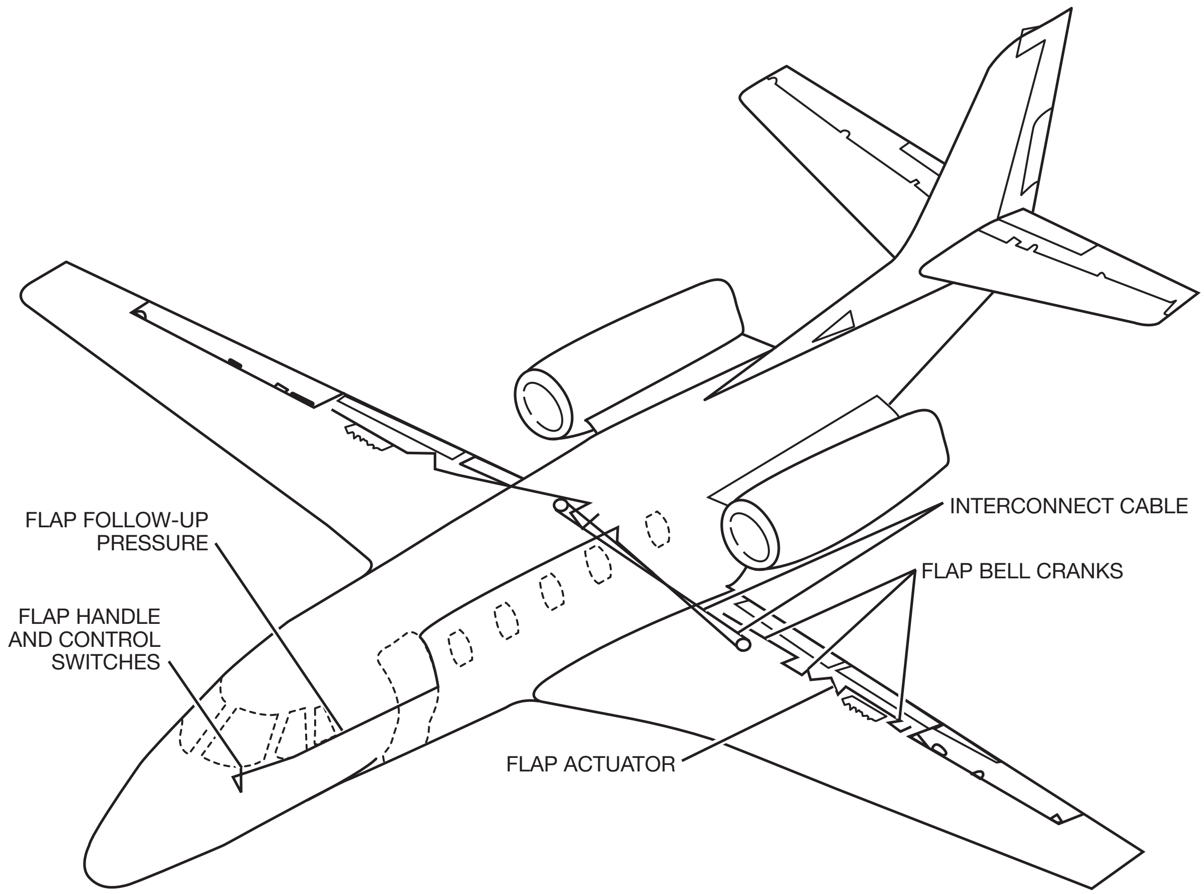


Figure 27-41. Flap Control System

OPERATION

The system is armed when airspeed is below 215 knots, ±10. When the flap control handle is moved (up or down), power routes through the position switches in the horizontal stabilizer and to the hydraulic control printed circuit board (PCB) (see Figure 27-38). The hydraulic loading valve closes, building pressure. The control valve opens and ports pressure to the extend or retract side of the actuator. When the stabilizer reaches its position, the switches send signals to the hydraulic control printed circuit board, cutting off power to the loading valve.

FLAP SYSTEM

DESCRIPTION

The flap system consists of two flaps, constructed of graphite composite laminates, per wing (Figure 27-41). They are electrically controlled, hydraulically actuated and operate through a range of 0 to 35° of travel. The flaps travel on rollers that are on tracks at the ends of each flap. The mechanical control system utilizes bellcranks and pushrods to push the flaps down or pull them up as they travel in the tracks. The bellcranks, in the trailing edge of the wing at the inboard and outboard end of each flap, are rotated through a mechanical linkage system powered by a hydraulic actuator in each wing. The left and right wing flap systems connect together with an interconnect cable to prevent a split flap condition. Bridled onto the interconnect cables is a follow-up cable system to transmit flap position to an indicator, as well as to control switches in the cockpit.

NOTES

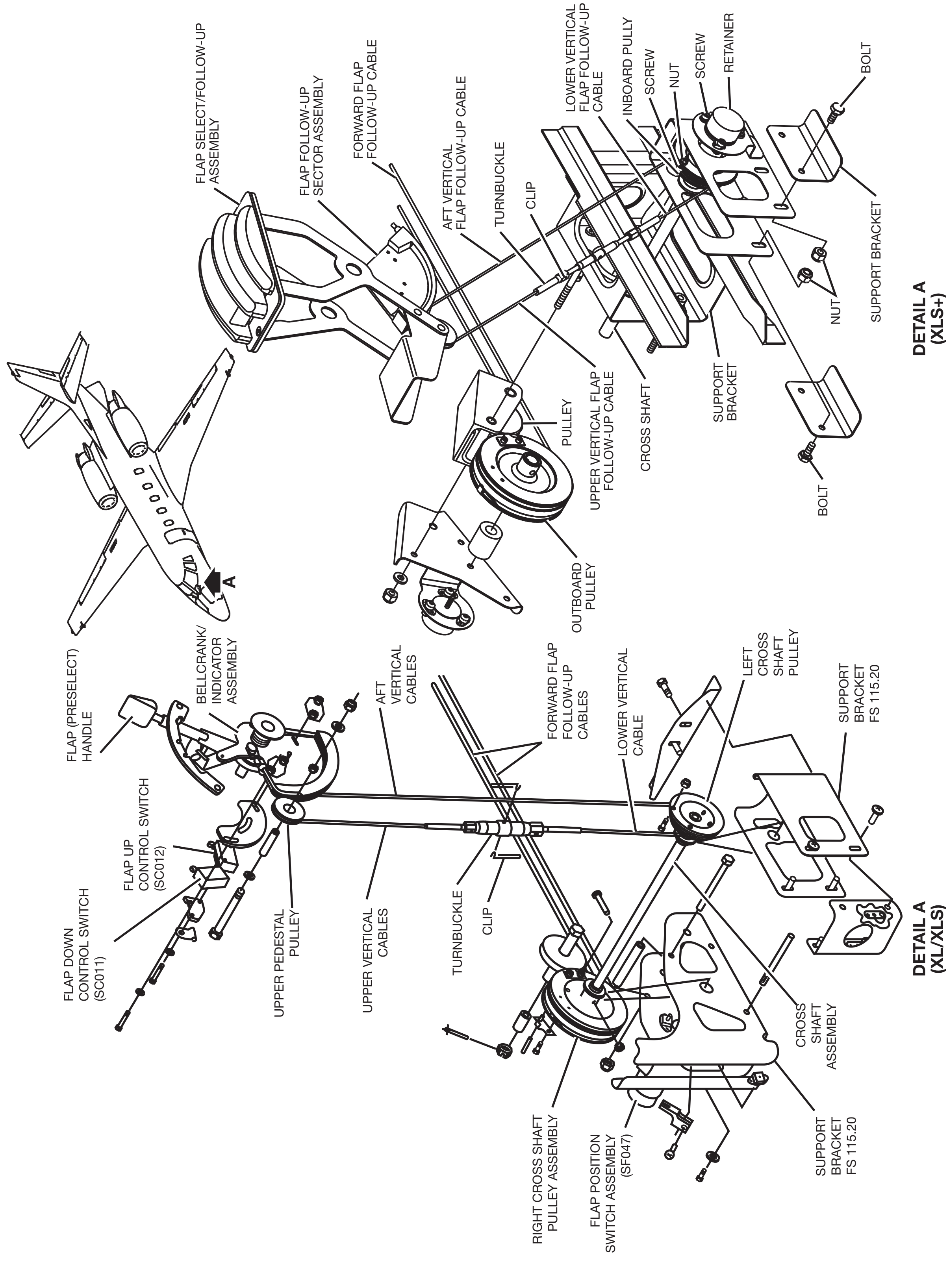


Figure 27-42. Cockpit Flap Control and Indicating System

COMPONENTS

Flap (Preselect) Handle

The flap (preselect) handle is a spring-loaded assembly (Figure 27-42). The spring is compressed when downward force applied to the flap handle moves the pin from the detent position. As the handle moves, the cam contacts either the up or down control switch—depending upon which direction the handle is moved. The flap control switches on the pedestal control power to the flap control valve, which controls pressure to the flap actuators. The detents for the preselect are set for 7°, 15° and 35° of flap travel. When the flap (preselect) handle is in the fully retracted position, the "up control" switch remains on, and the system is shut off by up-limit switches on each actuator.

Flap Control Valve

The flap control valve is a 3-position, 4-way solenoid operated valve. Moving the flap preselect lever to the "down" position, the valve is electrically positioned to direct inlet flow toward the extend port of the flap actuator; and to direct the returning hydraulic fluid flow (from the actuator to return). Moving the flap preselect lever to the "up" position, the valve is electrically positioned to direct the inlet flow toward the retract side of the flap actuator, and to direct the returning flow from the actuator to return. When the flaps reach the preselected position, the control valve deenergizes to the neutral position. All four ports are blocked in this position.

Follow Up System

The follow up system consists of a 1/16 diameter stainless steel cable loop attached to the flap interconnect cable by means of clamp blocks. The follow-up cables exit from behind the aft spar, out the right side, and over the right wing next to the fuselage. They pass over a couple sets of pulleys, which guide the cables through pressure vessel seals where they enter the fuselage under the copilot seat. The cables continue forward under the copilot floorboards until they reach a cross-shaft assembly in front of the copilot control column to which they are attached. To the right of the cable attach point is the flap position switch assembly which is actuated by the cross shaft rotation. To the left of the cable attach point and connected to the cross shaft is another cable pulley, which has a loop of cable connecting it to the bellcrank/indicator assembly in the control pedestal. The up and down control switches included with the bellcrank/indicator assembly are now repositioned with flap movement due to the follow up cable system.

NOTES

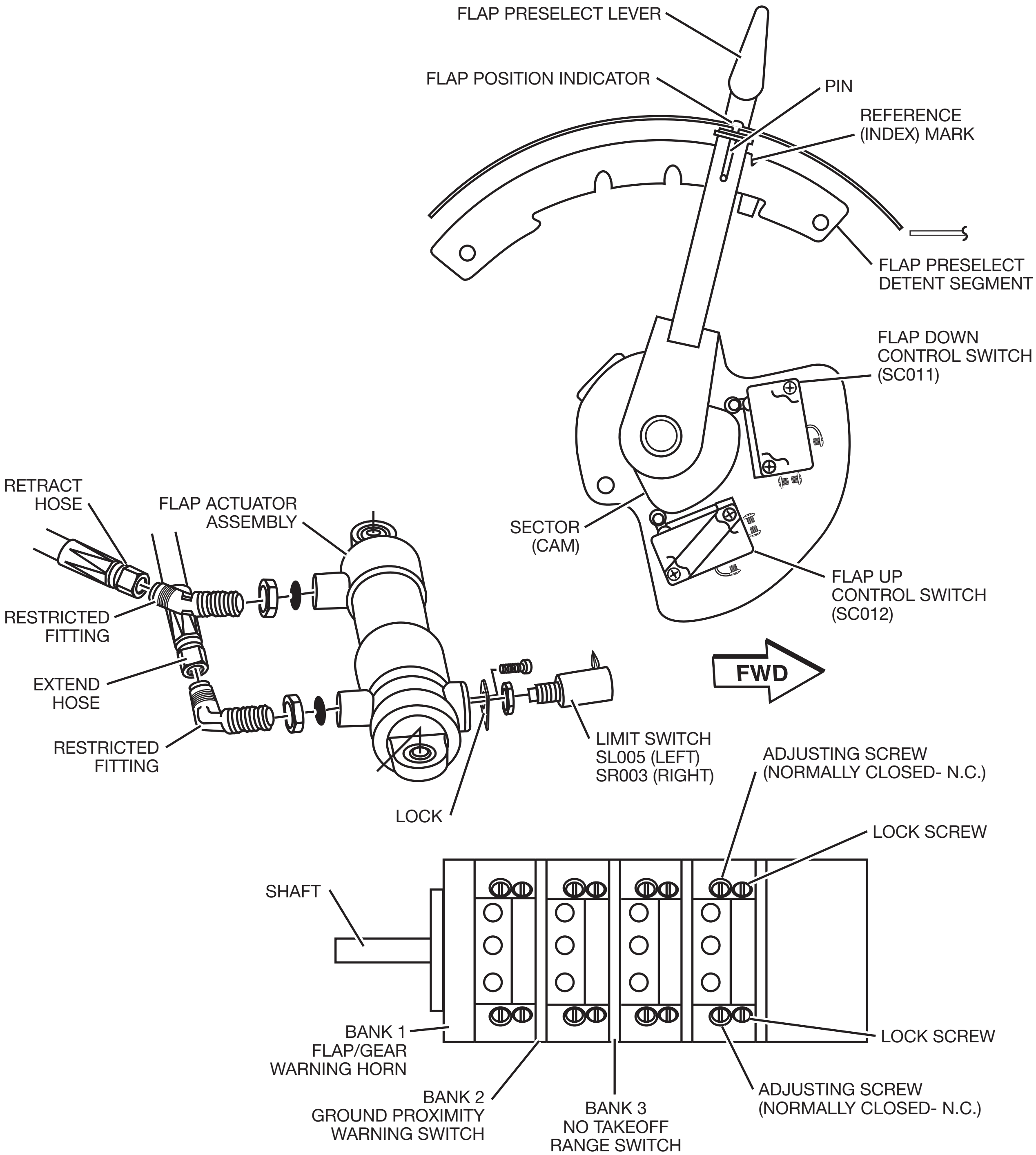


Figure 27-43. Flap Control and Indicating Electrical Components (XL/XLS)

ELECTRICAL OPERATION

When the flap preselect lever is moved, one of the two control switches (up or down) is closed by the cam (Figure 27-43). The electrical signal triggers flap motion (in the preselected direction) by activating the flap control valve. As the flaps move, the follow-up cable loop moves with them and rotates the control switch mounting plate in the same direction that the preselect lever was moved. When the plate and switches “catch up” with cam, the electrical signal is cut off, and the flaps stop in the “preselected” position. The only exception is that the flap up control switch remains closed in the 0° flap position to assure that both actuators reach their up and locked positions. When the left and right actuators reach their up and locked positions, the up-limit switches (at the actuators) deenergize the system.

NOTES

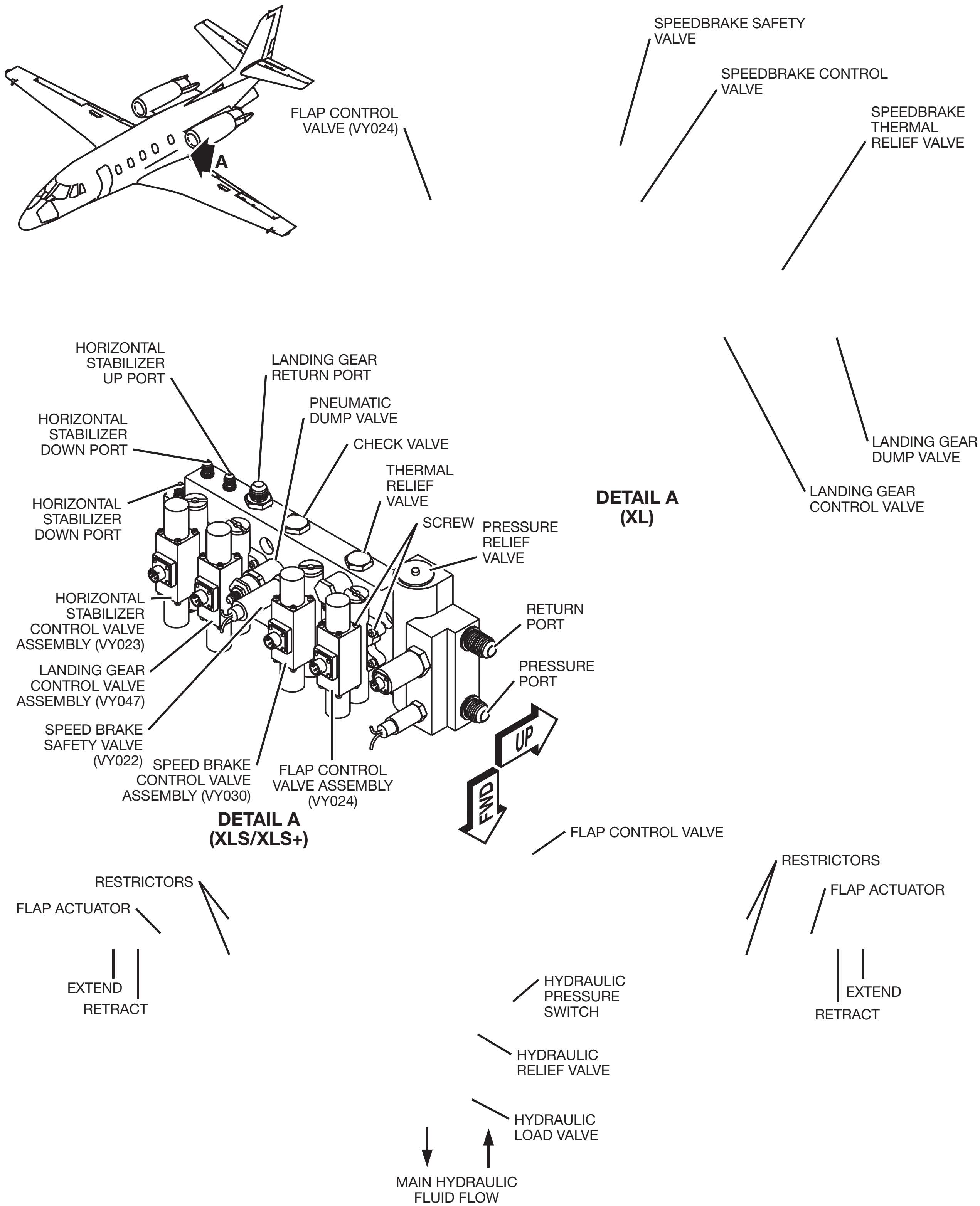


Figure 27-44. Flap Control Hydraulic System

HYDRAULIC OPERATION

The hydraulic system consists of two hydraulic actuators, one in each wing. Each actuator drives its portion of the flap mechanism. Either actuator can drive the entire system through the inter-connecting cable loop. The up or down control switch activates when a flap position is selected by means of the flap handle. This energizes the hydraulic load valve closed; and returns the flap control valve to the selected position (Figure 27-44). With the hydraulic system loaded, and the control valves shuttled to the extend or retract position, hydraulic fluid under pressure is provided to the appropriate end of the flap actuators. The actuators then drive the flap mechanism until the follow-up system operates the switch to signal the hydraulic system to shut off. When the flaps are retracted to the full up position, a mechanical lock in each flap actuator locks the flaps in the "up" position. The up-limit switches assure that both actuators are locked up before the hydraulic system shuts off.

DIAGNOSTICS

Flap Position Adjustment

1. Apply electrical and hydraulic power to the aircraft.
2. Move the flap control lever to 0° position and make sure that the flaps are full up.
3. Place a digital inclinometer on the center trailing edge of the inboard flap and use this position as 0° reference on the inclinometer.
4. Disconnect the flap position switch plug JF048, forward of the copilot control column under the floor.
5. While monitoring continuity at pins 1B and 1C on plug JF048 with a multimeter, adjust Bank 1 (Flap/gear warning horn) to achieve no continuity from 0° to 15.9°, and continuity from 16° to 35°.
6. While monitoring continuity at pins 1E and 1F, adjust Bank 2 (Ground Proximity Warning) to achieve continuity at a flap position between 30° and 35°, and no continuity elsewhere.
7. While monitoring pins 2A and 2B, adjust Bank 3 (No Takeoff), to achieve no continuity at flap positions from 6° to 16°, and continuity at all other flap positions.
8. Reconnect flap position switch plug JF048 to Flaps Position Switch Assembly SF047.

Flap Up-Limit Switch Adjustment

1. Remove the flap actuator assembly
2. Connect a hydraulic hand pump to retract (port of flap actuator assembly). Apply hydraulic pressure to unlock the flap actuator assembly.
3. Connect an ohmmeter to pins 1 and 3 of the limit switch.
4. Remove a screw from the switch lock.
5. Loosen the limit switch jamnut and rotate the limit switch clockwise, until continuity is indicated on the ohmmeter.
6. Rotate the limit switch counterclockwise, until the ohmmeter indicates no continuity.
7. Rotate the limit switch clockwise (1/2 to one turn) until a hole in the lock aligns with a hole in flap actuator assembly. Install a screw here. Tighten the switch jamnut.
8. Connect an ohmmeter to pins 1 and 2 of the limit switch.
9. Connect a hydraulic hand pump to the extend port. Fully extend the actuator. The ohmmeter should show continuity when the actuator is fully extended.
10. Safety wire the screw and jamnut.
11. Install the flap actuator assembly.

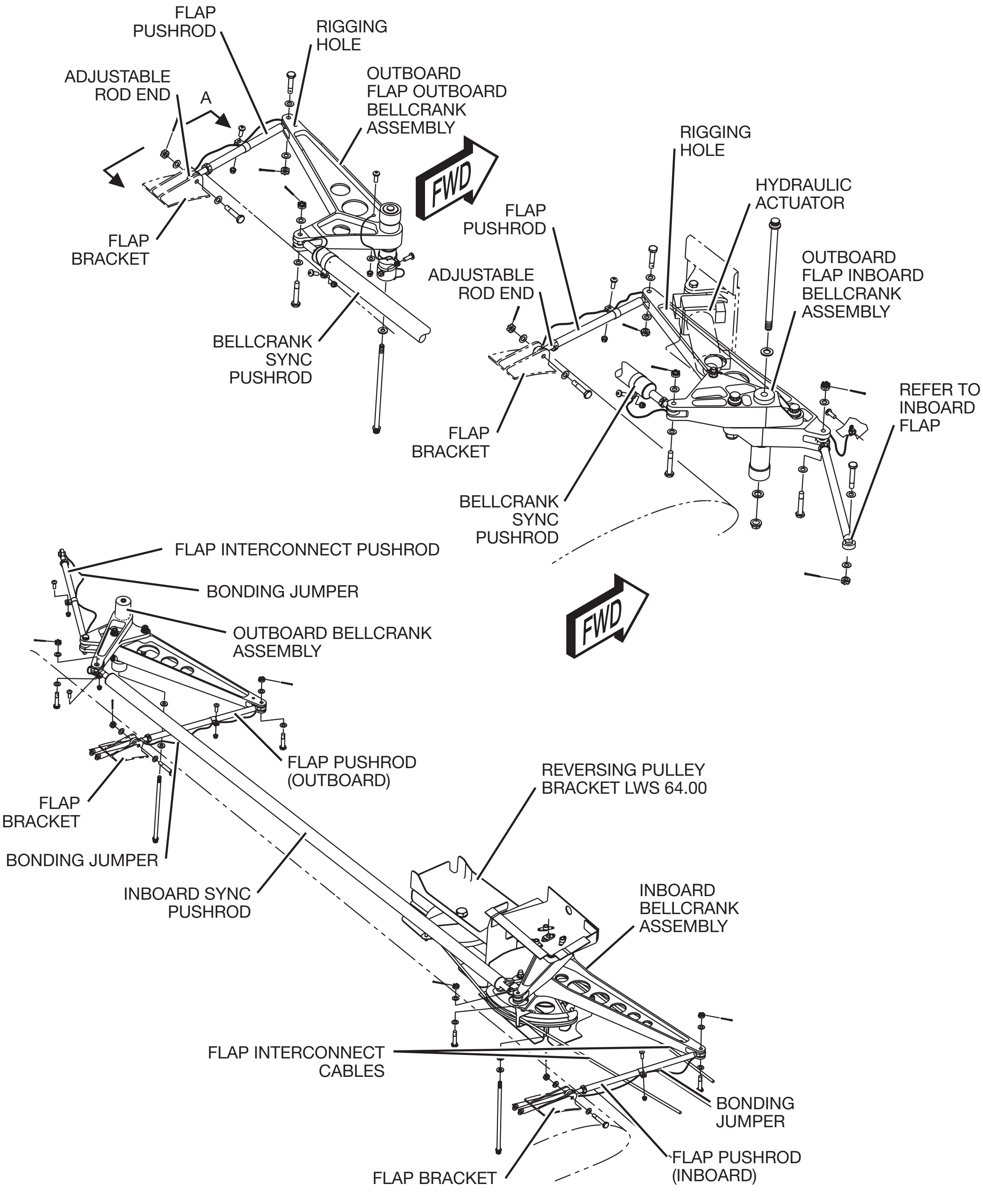


Figure 27-45. Flap Bellcranks and Pushrods

Flap System Adjustment

The general procedure for flap rigging is as follows:

1. Rig each flap panel independently.
2. Connect the inboard flap panels to outboard flap panels.
3. Attach the flap actuators and adjust the locking pressures.
4. Connect and adjust the interconnect cables.
5. Set follow-up (feedback) cable loop.
6. Adjust the approach and the interconnect switches.

Flap System Rigging

1. Remove the fuselage fairing access panels 313AL and 314AR to access the flap interconnect cables.
2. Using locally fabricated rigging templates and short Number #10 screws, locate the templates at the rigging holes, in the upper clevis lugs of the long bellcrank arms.
3. For each panel, adjust the bellcrank sync pushrod until both templates touch the rear spar web. Templates may swivel to touch the rear spar web (Figure 27-45).
4. Finger tighten the lock nuts and remove the templates.
5. Adjust the flap pushrod rod ends to 1.3 inches from the shoulder of the tube to the center of the bolthole. Finger tighten the lock nuts.
6. With the flap panel fully extended, examine the rod end alignment with flap attach clevis pins.
7. If the pushrods do not align with the bolt clevis pins, split the difference by shortening one rod end and lengthening the other (the same number of turns).
8. Install bolts.
9. Apply 5 to 20 pounds of pressure at the center of the trailing edge. Push the flap panel up against the flap seal strip (0° position) and measure the distance from the fixed roller to the end of the track slot at each end of the flap.

NOTE

The measurements should not differ by more than 0.05 inch (1.27 mm) on the outboard flap or 0.10 inch (2.54 mm) on the inboard flap. If they do, a tracking adjustment is required.

10. Install inboard flap/outboard flap interconnect pushrods and adjust so that the trailing edges align with each other at full up (0°) position.
11. Attach the flap actuator to the bellcrank.
12. Perform a flap actuator locking pressure adjustment/check as follows:
    A. Connect an ohmmeter to the left actuator limit switch (aft wing root connector PM001, pins J to H) and monitor continuity. Locate connector under wing fairing (165 CL).
    B. Retract flaps with a hand pump and adjust the actuator rod end so that locking occurs at 600 to 800 PSI (Lengthening the rod end decreases locking pressure while shortening the rod end increases locking pressure). Do not exceed 800 psi. Continuity is broken when the actuator locks.
    C. Monitor the right actuator (aft wing root connector PS001, pins J to H) and repeat. Access under fairings: 165AL and 166CR.
    D. Connect the electrical connectors (PM001 and PS001).
13. Hand pump both sides of the system to the up-and-locked position.
14. Connect and adjust the interconnect cable tension to 90 pounds, ±40.
15. Safety check all connections.
16. Remove the hydraulic hand pump, reconnect the lines and install fuselage fairing access panels.

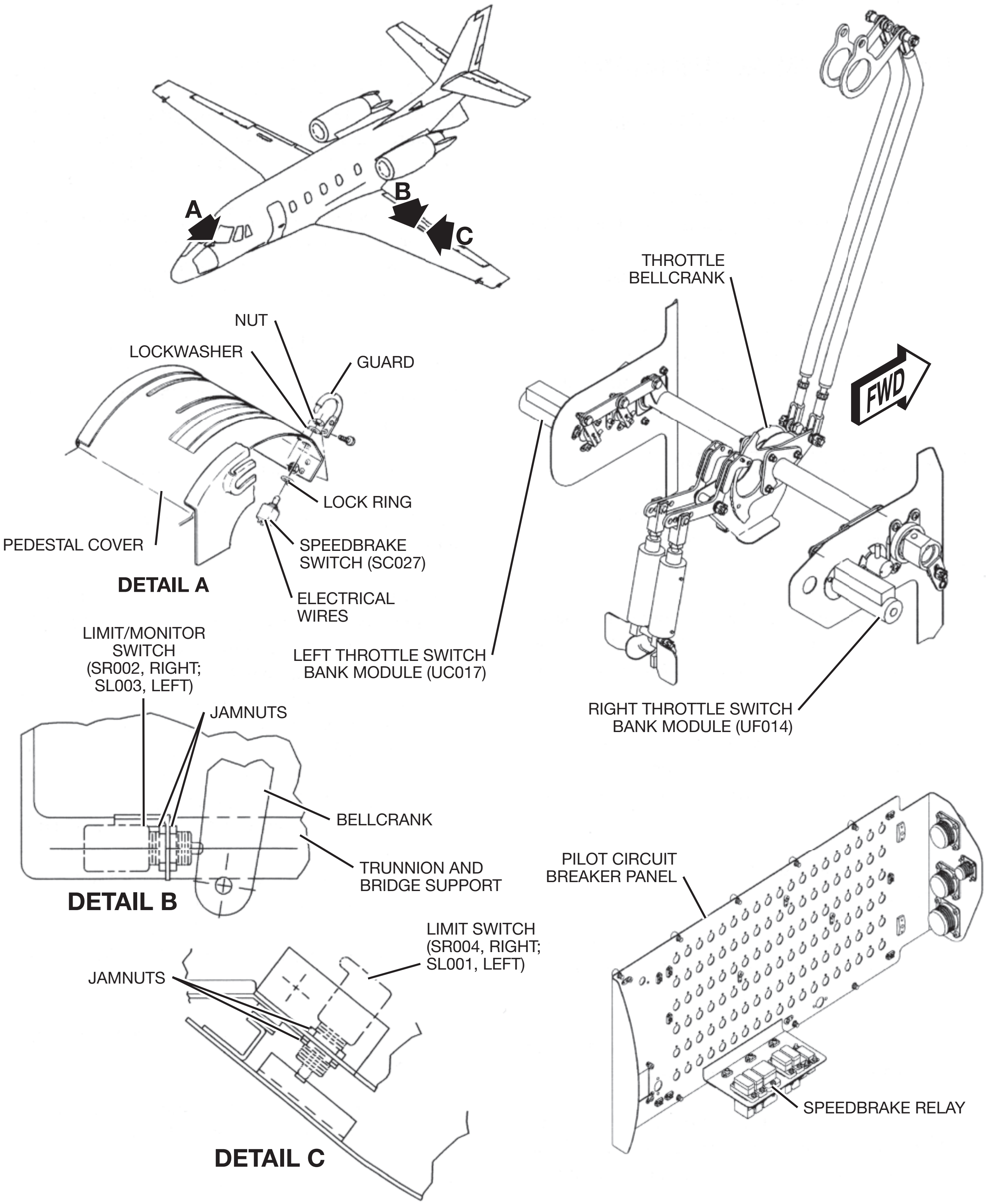


Figure 27-46. Speedbrake Electrical Control Components (XL/XLS)

SPEEDBRAKES

DESCRIPTION

The speedbrakes are on the upper and lower surfaces of each wing forward of the flaps. They are electrically controlled and hydraulically operated.

The speedbrake electrical control components (Figure 27-46) includes:

- Speedbrake control switch
- Speedbrake relay
- Two throttle switches
- Two speedbrake limit/monitor switches

The hydraulic control components include:

- Control valve
- Safety valve
- Thermal relief valve
- Two check valves
- Two actuators

The speedbrakes are monitored by an indicating system through the use of limit/monitor switches. The limit/monitor switches are on the wing structure and are actuated by the speedbrake bellcrank and lower door. The bellcrank-actuated monitor switches are connected to a SPEEDBRAKE EXTEND annunciator (XL/XLS) or white SPEED BRAKES CAS message (XLS+).

COMPONENTS

Control Switch

The speedbrake control switch is on the pedestal between the manual pitch trim wheel and the throttle levers (XL/XLS) or on the side of the throttle knobs (XLS+). The switch is a momentary switch, where the center position is neutral and the extend and retract positions are both momentary. Momentarily pushing the switch to either extend or retract electrically commands the system to move the speedbrake doors to the desired position. Once the switch is released it automatically returns to the neutral position, while the system continues to transition the speedbrake doors to the selected position.

Speedbrake Relay

The speedbrake relay is a latching relay on the left CB panel. When the relay is deenergized the speedbrakes are electrically commanded to retract, or when energized, the speedbrakes are electrically commanded to extend.

Throttle Switches (XL/XLS)

The speedbrake throttle switches are under the cockpit floor directly below the respective pilot or copilot seats. If either or both throttle levers are pushed above the 54° TLA power setting, the ground path for the speedbrake relay is broken and the relay deenergizes, automatically retracting the speedbrakes.

Limit/Monitor Switches

The speedbrake limit/monitor switches are installed on the wing structure. They monitor the position of the speedbrakes causing the hydraulic system to unload, and the speedbrake control valve to return to the center neutral position when the speedbrake doors reach the commanded position. When both sets of speedbrakes reach the full extend position, the extend limit switches cause the SPD BRK EXTEND annunciator on the annunciator panel (XL/XLS) or the white SPEED BRAKES CAS message (XLS+) to illuminate (Figure 27-47). The SPD BRK EXTEND annunciator extinguishes as soon as the speedbrake bellcrank moves away from either extend limit switch during the retract cycle.

SPEED BRAKE EXTENDED
Annunciator illuminates steady to indicate both speed brakes are fully extended. On the ground, the NO TAKEOFF annunciator will also illuminate.

XL/XLS ANNUNCIATOR

SPEED BRAKES			
Color	**Inhibited By**		**Debounce**
White		TOPI	Standard
This message is displayed when either speed brake panel is extended. On each speed brake, there is a mechanical switch which sends a 28 Volt signal to the EICAS to display the message. When the speed brake is not extended, an open signal is sent to the EICAS system.			

XLS+ CAS MESSAGE

Figure 27-47. Speedbrakes Indications

Speedbrake Actuators

There are two speedbrake actuators, one in each wing to operate both upper and lower doors through the use of a bellcrank. The actuators are attached to the wing structure and to the bellcrank for extending and retracting the speedbrake doors. The actuator operates at up to 1,500 psig hydraulic pressure.

Speedbrake Hydraulic Control Components

Speedbrake hydraulic control components are on the hydraulic sub-panel in the fuselage fairing area aft of the wing (Figure 27-48).

The speedbrake hydraulic control system includes:

- Three position (extend, block, and retract) control valve
- Thermal relief valve
- Safety valve
- Two check valves

The control valve directs the hydraulic fluid so that the speedbrakes move to the selected position. The thermal relief valve relieves excess hydraulic fluid pressure caused by an increase in temperature in the speedbrake stow line. The safety valve provides a redundant path to return for removing extension pressure in the event of a control valve failure or in the event of an electrical failure. In order to prevent a vacuum in the system due to either failure, a check valve is installed in the stow line. This allows fluid to enter the system as air loads, and pushes the speedbrake doors to the trail position. The check valve on the extend line prevents return line pressure spikes (created during other system operations) from inadvertently extending the speedbrakes.

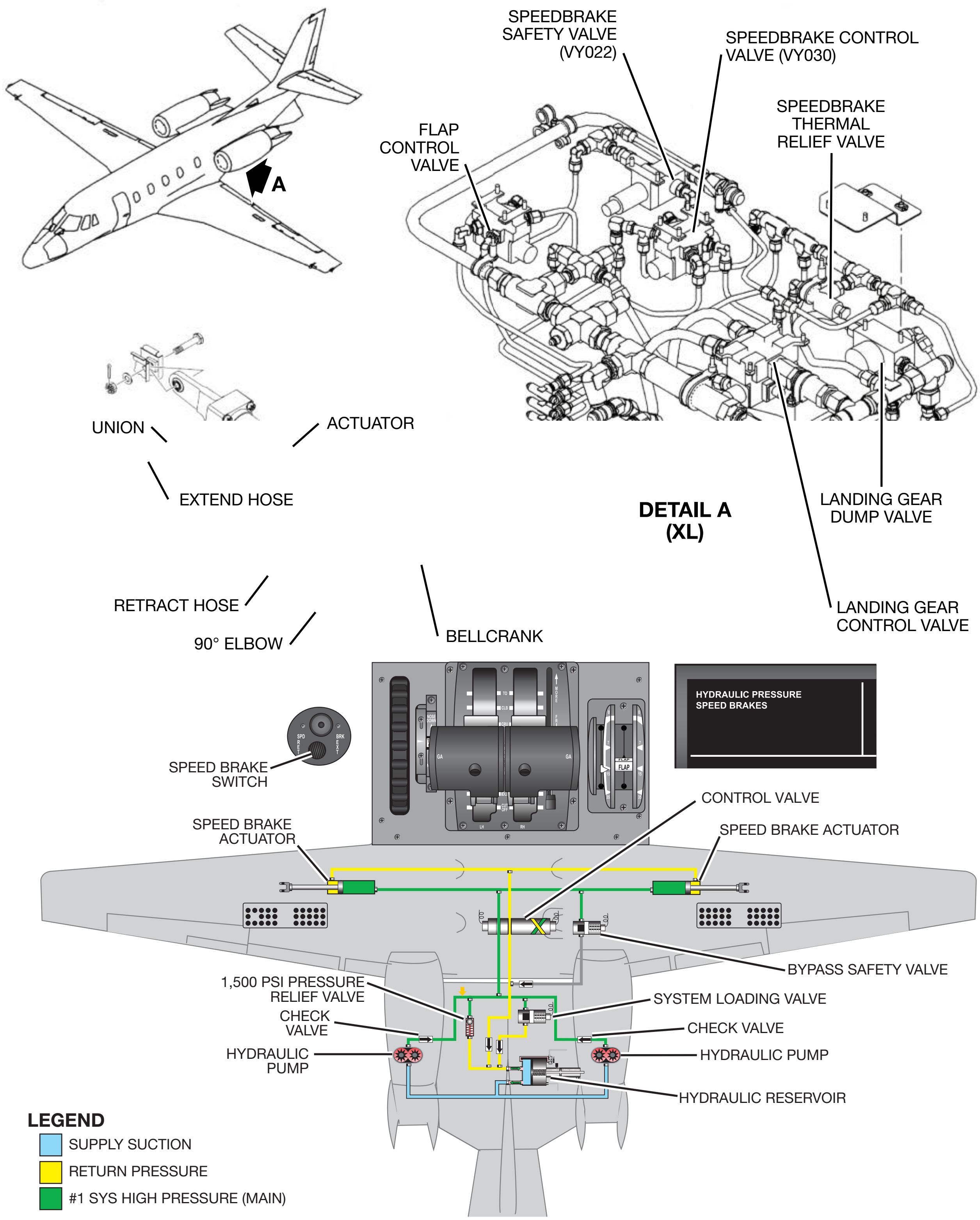


Figure 27-48. Speedbrake Hydraulic Control System

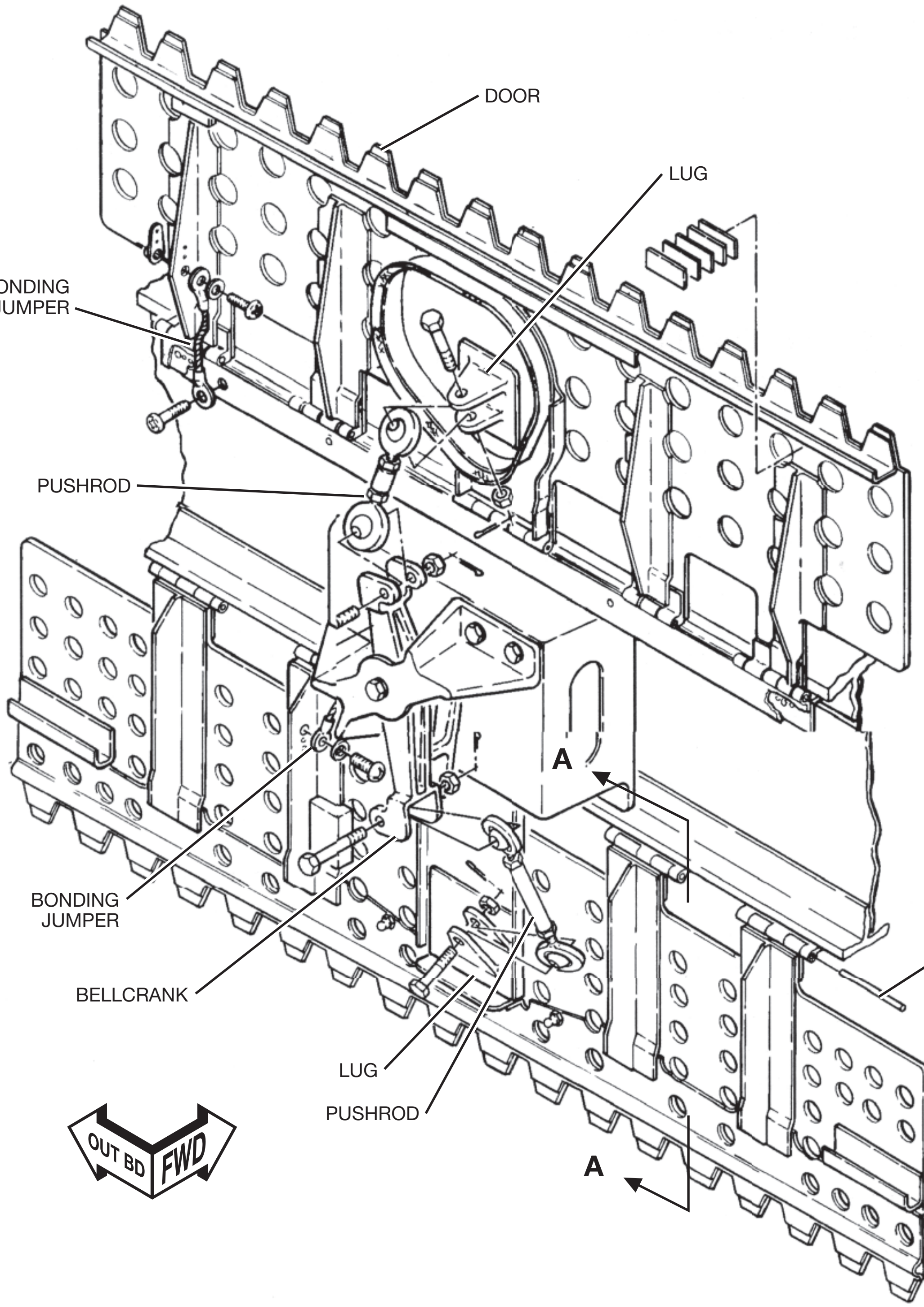


Figure 27-49. Speedbrake Bellcrank and Doors

OPERATION

Hydraulic Operation

Placing the speedbrake position switch in the extend position pressurizes and illuminates the HYD PRESS ON annunciator. This also energizes the speedbrake relay, which in turn energizes and extends the speedbrake safety valve closed; and energizes the control valve to the extend position. The control valve directs pressure to the extend side of the actuators, while the safety valve prevents fluid from freely flowing back to return.

When the speedbrakes are fully extended, the safety valve remains energized, but the control valve returns to neutral, blocking all fluid lines to the actuators—keeping the panels extended.

When the speedbrakes need to be retracted, the speedbrake position switch on the pedestal is momentarily placed in the “retract” position. The speedbrake relay deenergizes. The hydraulic system pressurizes again, and the control valve energizes, to direct pressure toward the stow side of the actuators. The safety valve deenergizes open. The speedbrake panels retract and upon contacting the stow limit switches, the hydraulic system unloads and the control valve returns to neutral. The lower speedbrake panels are held in the retract position with two retainers on each door to prevent droop after hydraulic pressure is removed.

In the event of an electrical failure with the speedbrake doors extend in flight. The safety valve fails open allowing air to blow the speedbrake doors to trail.

Mechanical Operation

The mechanical linkage of the speedbrake system (Figure 27-49) consists of:

- Two upper doors
- Two lower doors
- Two bellcranks

The speedbrake doors are on the upper and lower surfaces of each wing forward of the flaps. Each door is hinged to the wing’s rear spar with five hinge points. The center hinge point on each door incorporates a lug, mechanically linked to a bellcrank through the use of pushrods. The hydraulic actuator rotates the bellcrank as it extends, pushing both the upper and lower doors to the extend position. At the point of full extend, the bellcrank arm contacts the extend limit/monitor switch. As the actuator retracts, it rotates the bellcrank the opposite direction, retracting the speedbrake doors. At the point of full retract, the lower door contacts the stow limit switch.

NOTES

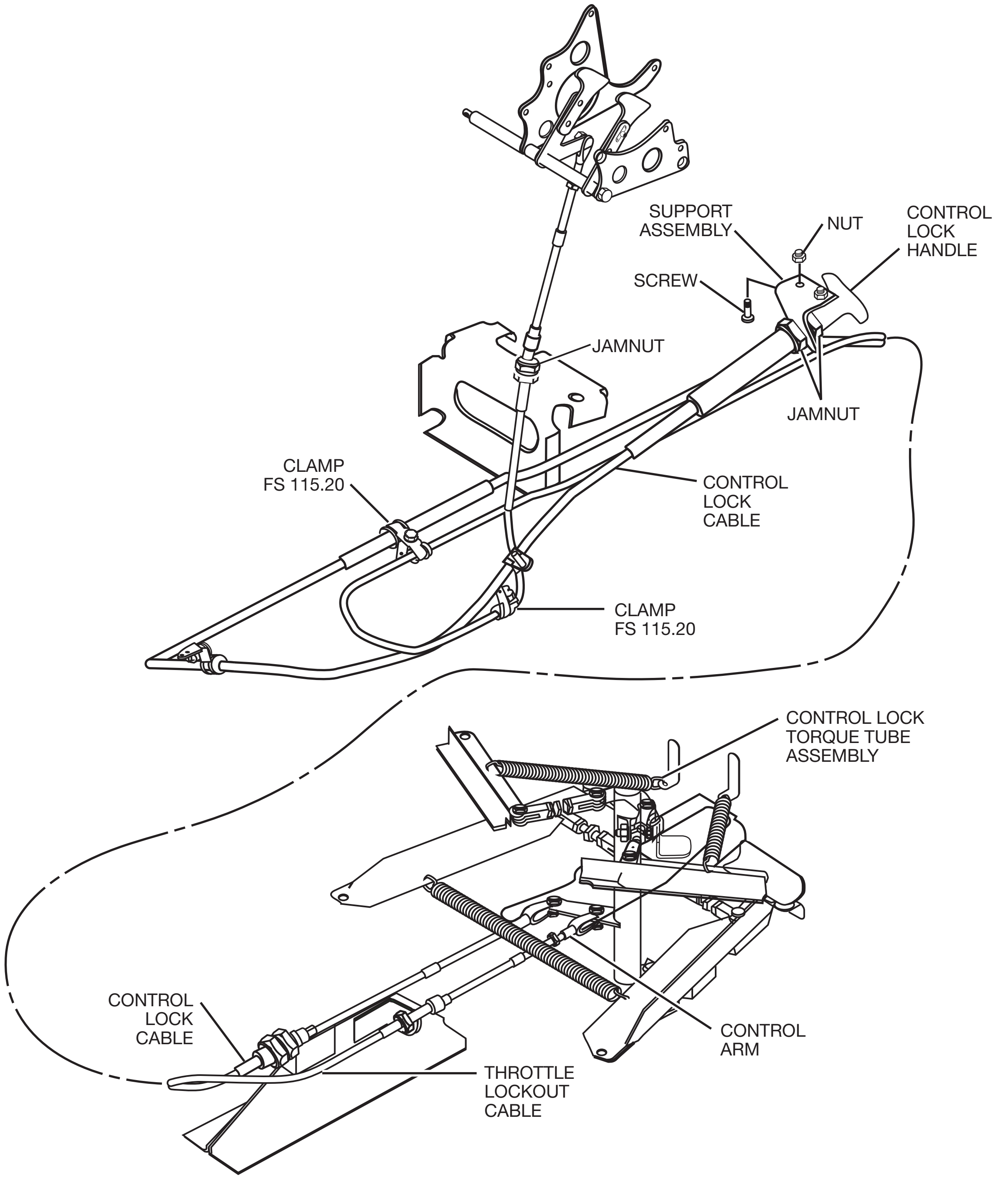


Figure 27-50. Control Lock System

CONTROL LOCK SYSTEM

DESCRIPTION

The control lock is engaged by pulling a cockpit T-handle. The mechanism requires that the throttles be in CUTOFF to engage the lock. The throttles are locked in the CUTOFF position by cams, and cannot be advanced from CUTOFF until control lock is disengaged. The primary flight control feedthrough sectors are locked by sliding bars, operated by the control lock torque tube (Figure 27-50). These bars force the sectors to “neutral” position as the control lock is engaged; and prevent rotation of the sectors until the control lock is disengaged. The torque tube goes over center to prevent airloads or control inputs from disengaging the control lock.

NOTES

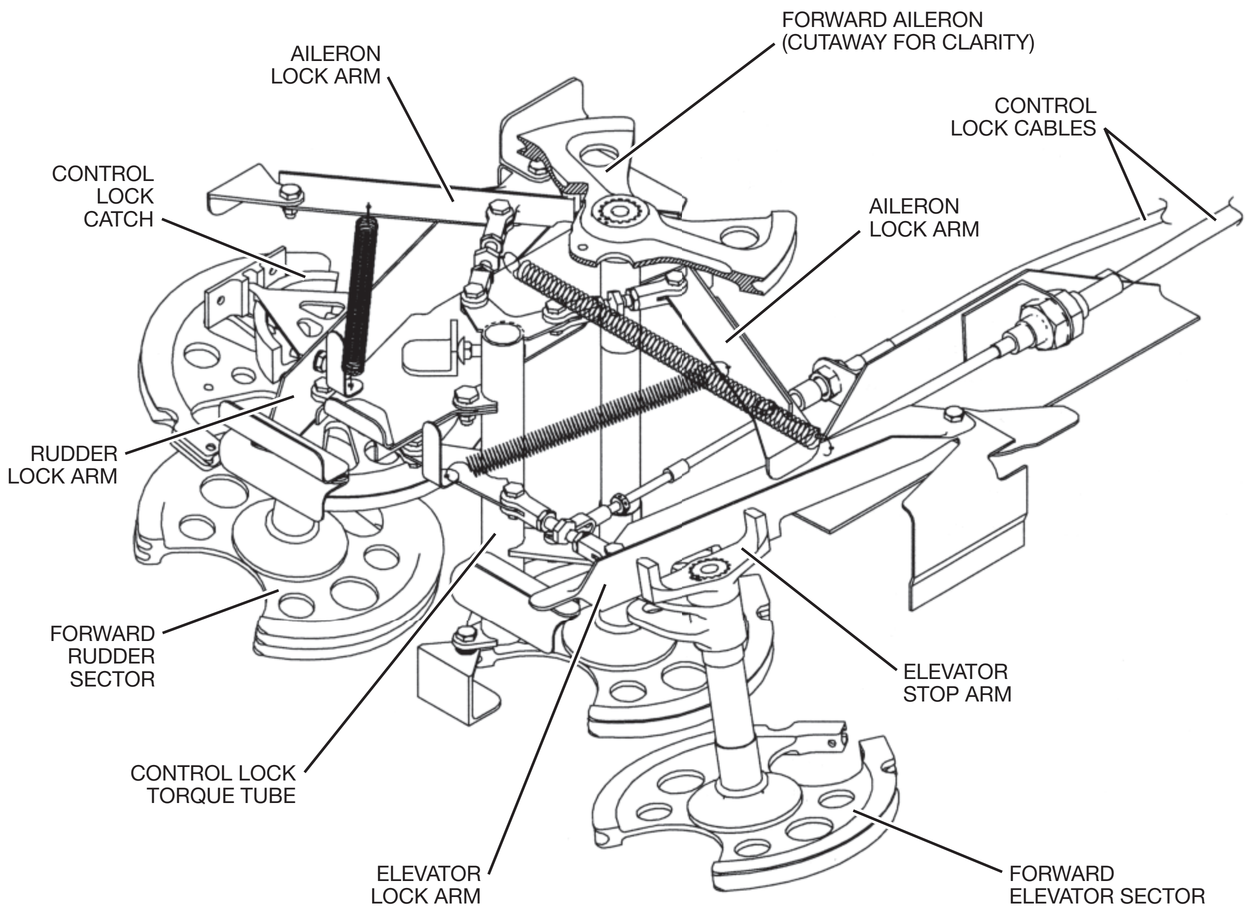


Figure 27-51. Control Lock Torque Tube and Sector Arrangement

COMPONENTS

Control and Throttle Lockout Cable

The control and throttle lockout cables are enclosed push-pull cables. They route from the control handle and throttle lock bellcrank aft under the cockpit floorboards to the control lock mechanism (below cockpit aft center floor panel). The control cable has an adjustable clevis end where it attaches to the control lock torque tube control arm. The throttle lockout has adjustable clevis ends at each end of the cable (Figure 27-51).

The control lock cable attaches to the control handle at one end and the control locking mechanism at the other. The primary action of the control cable is to pull the torque tube control arm, that rotates the torque tube over-center and drives the pushrods to position the lock arms.

The throttle lockout cable attaches to the throttle lock bellcrank at the forward end and to the torque tube control arm (at the aft end). As the torque tube control arm moves forward, the lockout cable is pushed, forcing the forward end to position the throttle lock bellcrank cams under the throttle linkage, thus locking the throttles in cutoff.

OPERATION

To engage the control lock system, align rudder pedals fore and aft, aileron control wheels level, elevators neutral, and the throttles in cutoff position. Pull control lock handle; this rotates the control lock torque tube, moving the locking arms against the elevator. Aileron and rudder pass thru the sector stop arms. At the same time, the bellcrank rotates the throttle locking cams below their respective throttles.

To disengage the control lock system, move the control lock handle to the stowed position. This rotates the control lock torque tube, and retracts the locking arms from the pass thru sectors, and rotates the throttle locking cam from below the throttles.

DIAGNOSTICS

Operational Check

1. Remove cockpit aft center floor panel 141DTC to gain access to control lock torque tube and lock arms.
2. Remove control pedestal access panel 244AL.
3. Move the flight controls and throttles throughout their travel and ensure there is no binding or obstruction from the control lock system.
4. Place both throttles in CUTOFF position. Place the flight control surfaces in "neutral" position.
5. Engage the control lock system.
6. Verify that the throttle locking cams have moved under the throttle arms and the throttles are locked in the CUTOFF position.
7. Verify that the control lock torque tube has moved to the overcenter position and that there is no tension on the control lock cable (Figure 27-39).
8. Verify that the upper lock arms have moved forward against the aileron sector bosses and that the lower lock arms have moved outward against the elevator and rudder sector stop arms, holding the control surfaces in neutral.
9. Verify the control column, pedals and control surfaces can not be moved and throttle cannot be taken out of CUTOFF position.
10. Adjust the control lock system if any discrepancies are found.
11. Replace all access panels and floor panels.

QUESTIONS

1. The ailerons are operated by:
- A. Hydraulic pressure
- B. Mechanical inputs from the control wheels
- C. Fly-by-wire system
- D. Active control system that totally eliminates adverse yaw

2. The aileron trim tab is operated by:
- A. Electrically operated trim tab motor
- B. Hydraulically operated trim tab motor
- C. Mechanical trim knob on the center pedestal
- D. Changing the angle of the aileron fence

3. Regarding the rudder:
- A. The pilot and copilot rudder pedals are interconnected.
- B. The trim tab actuator is powered only electrically.
- C. The servo is connected to the air data computer to restrict rudder pedal deflection at high airspeeds.
- D. It is independent of the nosewheel steering on the ground.

4. The elevator:
- A. Trim tab is controlled only electrically
- B. Runaway trim condition can be alleviated by momentarily depressing the red AP/TRIM DISC switch
- C. Electric pitch trim has both high speed and low speed positions
- D. Trim tab is on the right elevator only

5. If hydraulic power is lost:
- A. The flaps are inoperative.
- B. The flaps operate with the backup electrical system, but extend and retract at a reduced rate.
- C. There is no effect on wing flap operation.
- D. A split flap condition could result if the flaps are lowered.

6. The wing flaps:
- A. If the wing flaps are positioned UP prior to takeoff, no visual or oral warning is present.
- B. Depend on both actuators to function to prevent a split flap condition.
- C. Can be lowered manually if electrical power is lost, but only if all hydraulic fluid has not been lost.
- D. Indirectly controls the position of the horizontal stabilizer position.

7. Regarding the gust lock:
- A. The engines may be started with it engaged.
- B. The aircraft should not be towed with it engaged.
- C. It must be engaged for towing.
- D. If the aircraft is towed, nosewheel steering may be damaged. It is still permissible to fly the aircraft if the gear is left down.

8. If hydraulic failure occurs with the flaps extended, the flaps:
- A. May blow upward immediately, depending on airload if the flap handle is moved.
- B. Cannot be fully retracted.
- C. Can be retracted up electrically
- D. Flaps remain in present position regardless if the flap handle is moved.

9. Extended speedbrakes are maintained in position by:

A. Continuous system hydraulic pressure.
B. Trapped fluid in the lines from the solenoid control valve.
C. Internal locks in the actuators.
D. External locks on the actuators.

10. The amber HYDRAULIC PRESSURE CAS message appears during speedbrake operation:

A. When the speedbrakes are fully extended.
B. While the speedbrakes are extending and retracting.
C. Both A and B.
D. Neither A nor B.

11. A true statement concerning the speedbrakes is:

A. The white SPEED BRAKE EXTEND CAS message displays whenever both sets of speedbrakes are fully extended.
B. If DC electrical failure occurs while the speedbrakes are extended, they remain extended since the hydraulic pressure is trapped on the extend side of the actuators.
C. If hydraulic pressure loss should occur while the speedbrakes are extended (Hydraulic system loading valve fails open), the speedbrakes automatically blow to trail.
D. The speedbrakes can only be retracted by placing the speedbrake switch to RETRACT.

12. The rudder bias system:

A. Is inoperative with the thrust reversers deployed.
B. Is inoperative with either emergency stow switch in EMER STOW.
C. Utilizes main system hydraulics.
D. Both A and B above.

CHAPTER 28
FUEL

CONTENTS

ILLUSTRATIONS

TABLE

CHAPTER 28
FUEL

INTRODUCTION

This chapter presents the fuel system for the Citation 560 XL/XLS/XLS+ aircraft and is limited to the airframe fuel system only. System discussion begins from the point of fueling the aircraft and continues to delivery of fuel to the engine, with emphasis given to components and their operation. General maintenance considerations are included, accompanied by functional and operational checks. References for this chapter can be found in Chapters 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 28—"Fuel," of the *Aircraft Maintenance Manual (AMM)*.

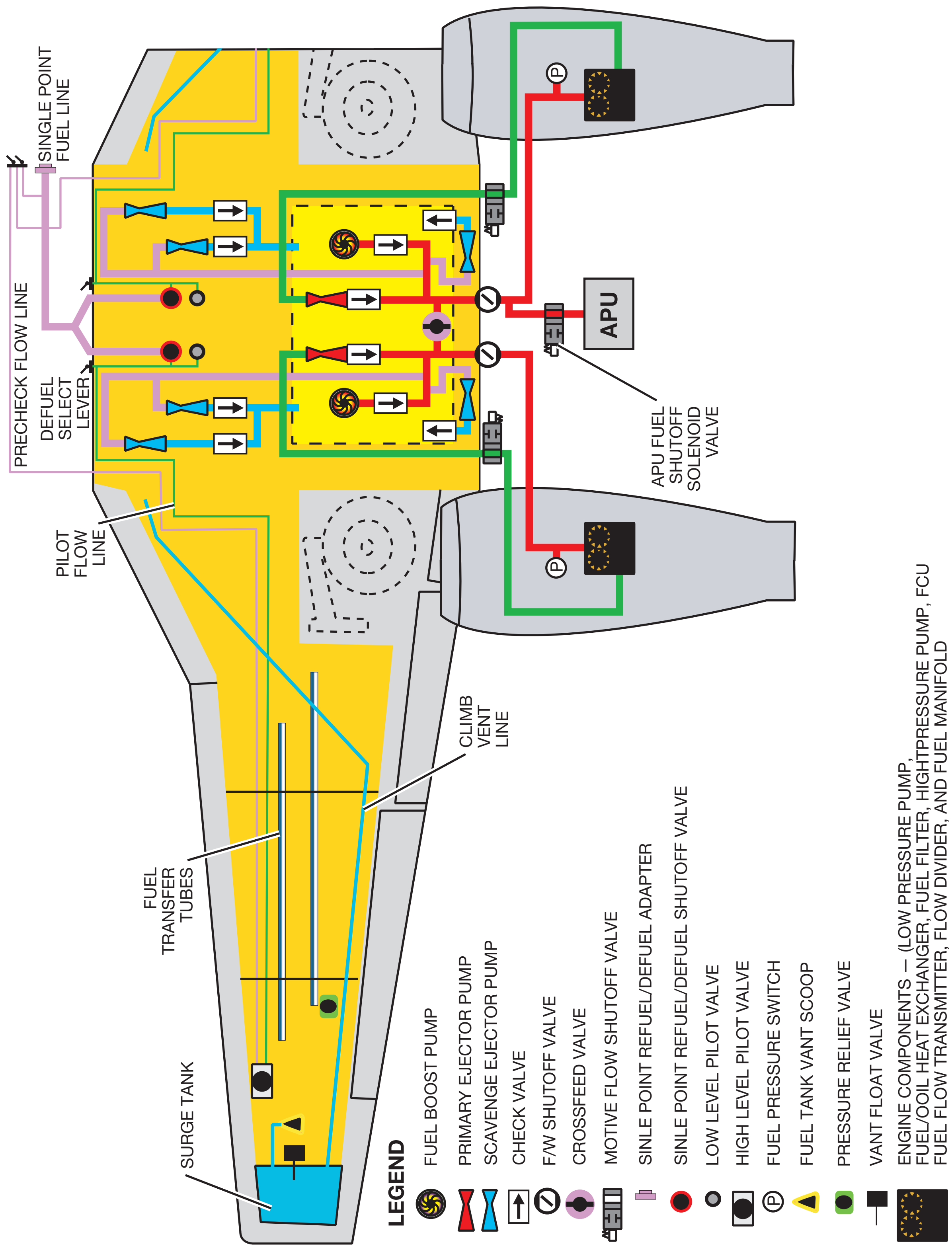


Figure 28-1. Fuel Distribution System Schematic

GENERAL

The sections contained in this chapter include:

- Fuel storage
- Single-point refuel/defuel
- Fuel distribution
- Fuel crossfeed
- Fuel indicating

The storage section covers:

- Ventilation systems
- Cell and tank interconnectors
- Overwing filler necks and caps
- Reservoir feed pumping systems
- Reservoirs within the tanks which are not a part of the distribution system (vent tank), etc.

The distribution section contains general coverage of the portion of the system used to distribute fuel from the filler connector to the storage system (Figure 28-1). It also covers the portion from the storage system, including the powerplant fuel quick-disconnect and single-point refueling/ defueling system, as well as the crossfeed system. Items such as plumbing, pumps, valves, controls, etc, are included.

The indicating section contains pictorial and general coverage of that portion of the system used to indicate the quantity and temperature of the fuel. This does not include engine fuel flow or pressure.

NOTES

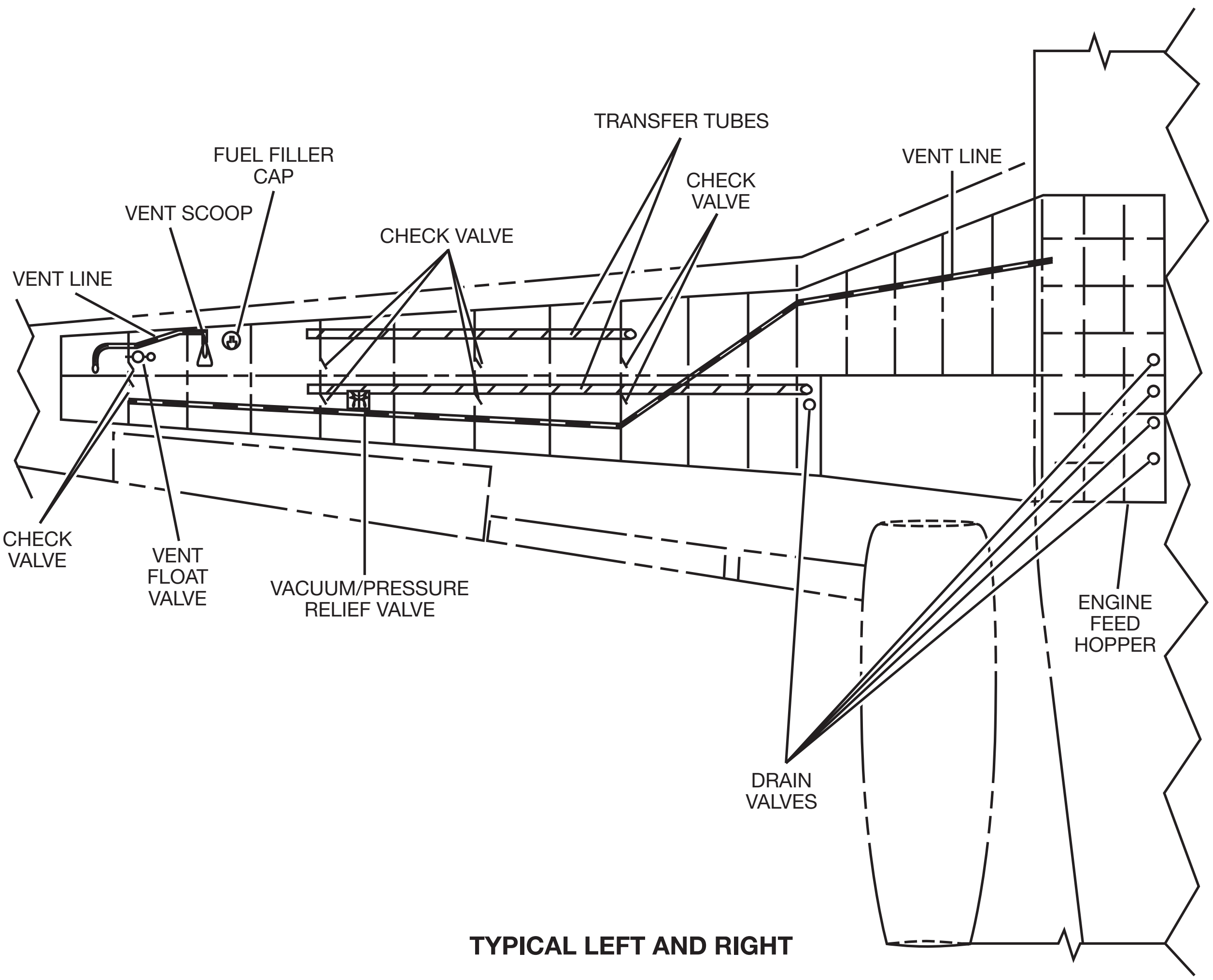


Figure 28-2. Wing Fuel Tank and Vent System

FUEL STORAGE

DESCRIPTION

The airplane has one integral fuel tank in each wing (Figures 28-2 and 28-3). Each wing fuel tank has a usable fuel capacity of approximately 503 gallons. The tank cavity extends from BL 0.00 outboard to WS 284.52 and is bounded by the forward and aft wing spars, except where it is interrupted by the main wheel well structure from WS 34.00 to WS 94.50. Lightening holes and stringer cutouts permit movement of fuel within the wings. Flapper-type check valves are in the rib assemblies in the outboard wing to prevent a rapid shift of fuel to the outboard section of the wing when the airplane is in a wing low attitude.

The engine feed hopper is a functional part of the wing fuel tank and is between BL 0.00 and WS 11.50 ribs and between the rear spar and FS 346.00 closeout. It is sealed except for vent opening at the top, in order to maintain a full hopper under low fuel conditions. It has flapper-type check valves that allow for gravity fuel flow into the hopper. The components that supply fuel to the engines, are within the hopper.

Fay, surface, and fillet sealing metal-to-metal joints and coating rivet heads with sealant, form the liquid tight fuel tanks. The interior

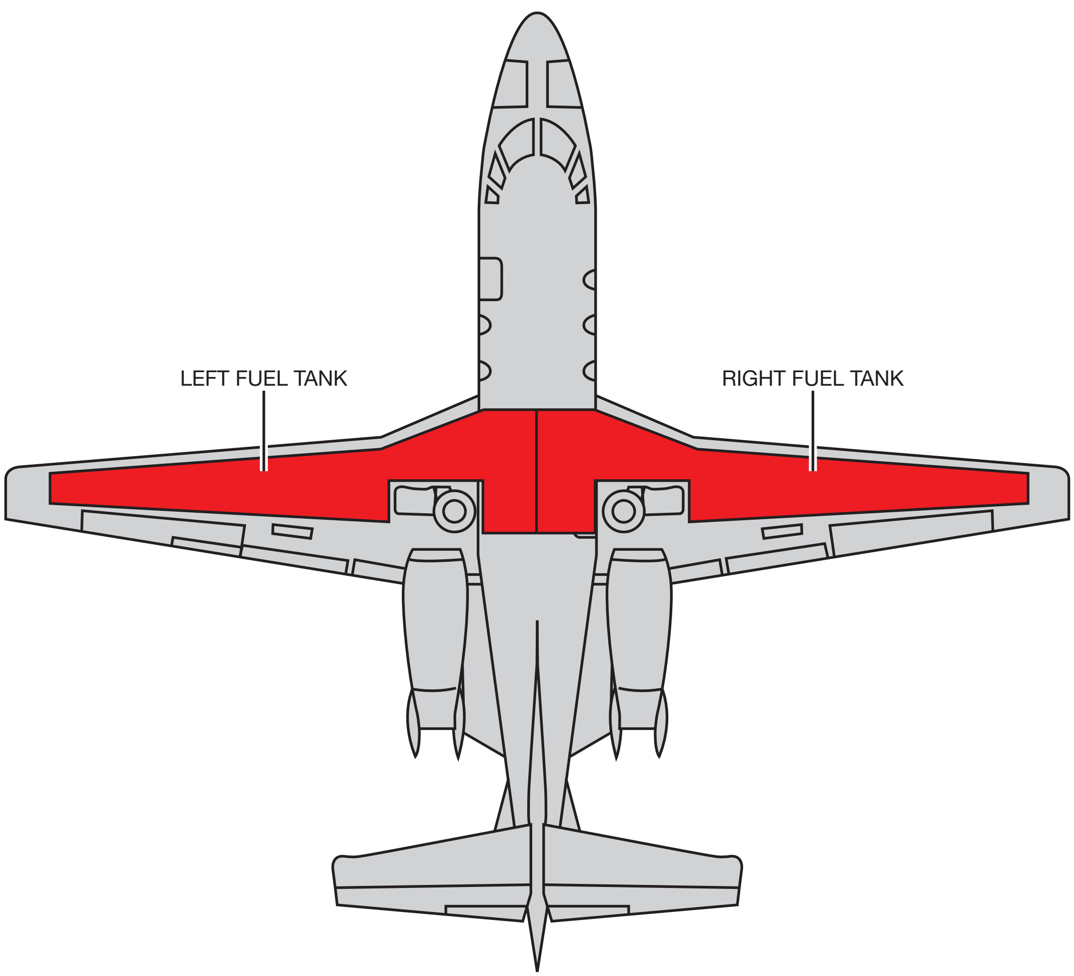


Figure 28-3. Citation XL/XLS/XLS+ Fuel Tank Location

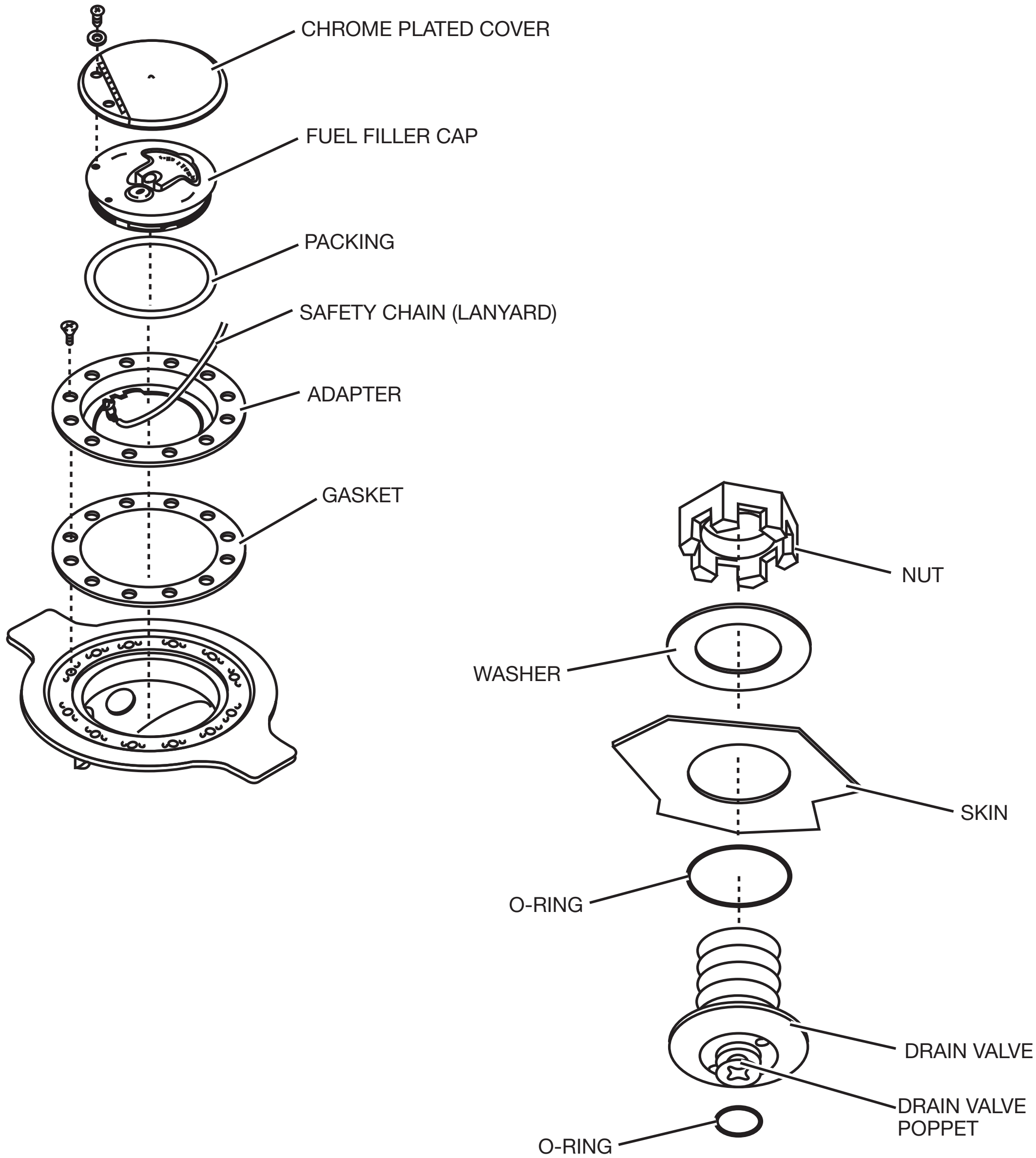


Figure 28-4. Drain Valve and Filler Cap Installations

surfaces of the tanks are chemically treated then coated with epoxy for corrosion resistance.

The right and left wing fuel tanks are interconnected by a crossfeed line which is opened or closed, by one electrically operated, crossfeed valve in the left wing center tank. The crossfeed valve is normally closed.

The surge tank (from WS 284.52 to WS 303.020) is the most outboard bay in the wing tank, but since it acts only as a fuel collector, it is not considered part of the tank capacity.

COMPONENTS

Tank Drain Valves

Five drain valves are in the lower surface of each wing. The valves are tool-operated, poppet type, that are semi-flush externally mounted. The valves allow the draining of sediment, moisture, and/or residual fuel from the tanks.

The spring loaded poppet is housed in the drain valve body (Figure 28-4). The poppet is spring loaded in the closed position. The valve is sealed by a packing on the poppet valve and another between the valve and the airplane skin. A slot in the end of the poppet allows for screwdriver operation of the valve to the OPEN position. A nut inside the fuel tank secures the valve to the skin.

NOTE

The drain valve poppet O-ring can be changed with fuel in the tank.

To remove the drain valve poppet O-ring, use a Phillips screwdriver, and turn the poppet clockwise until it drops down, exposing the O-ring. Then remove and replace the O-ring. After replacing the O-ring, turn the poppet counterclockwise while pushing upward, to re-install the poppet into the drain valve.

WARNING

Continuing to turn drain valve poppet counterclockwise past the locked position will result in the valve being locked in the open position.

Fuel Tank Filler

One flush mounted fuel filler cap and its adapter is on the upper surface of each wing near the outboard end. The fuel filler cap and adapters are used for fuel servicing when the single point refuel/defuel system is not used. The fuel filler cap and adapter includes:

- A key locking type fuel filler cap
- Adapter
- A safety chain (or lanyard) to attach the cap to the adapter

When fuel is filled through the fuel filler cap, the location of the fuel filler cap and the fuel filler standpipe control the full fuel tank level of each wing. Fuel will flow out of the fuel filler cap once the tank is full, assuring the standpipe expansion space cannot be filled with fuel.

Identical fuel filler cap and adapters are used on each wing. Each cap is recessed and marked to indicate open and closed positions. To remove the cap, lift the hinged cover (attached to the cap) to access the cap tab. Using the key provided (key marked with the word “FUEL”), unlock and rotate the cap tab counterclockwise. The cap may then be lifted off. To install the cap, reverse the procedure.

Locking fuel filler caps are provided in keyed-alike pairs. The keys are identified with the word FUEL and can be removed from the cap when cap is unlocked. Periodic lubrication of the lock is necessary for proper key operation. A bright chrome plated cap cover sits over the cap to protect the lock from weather.

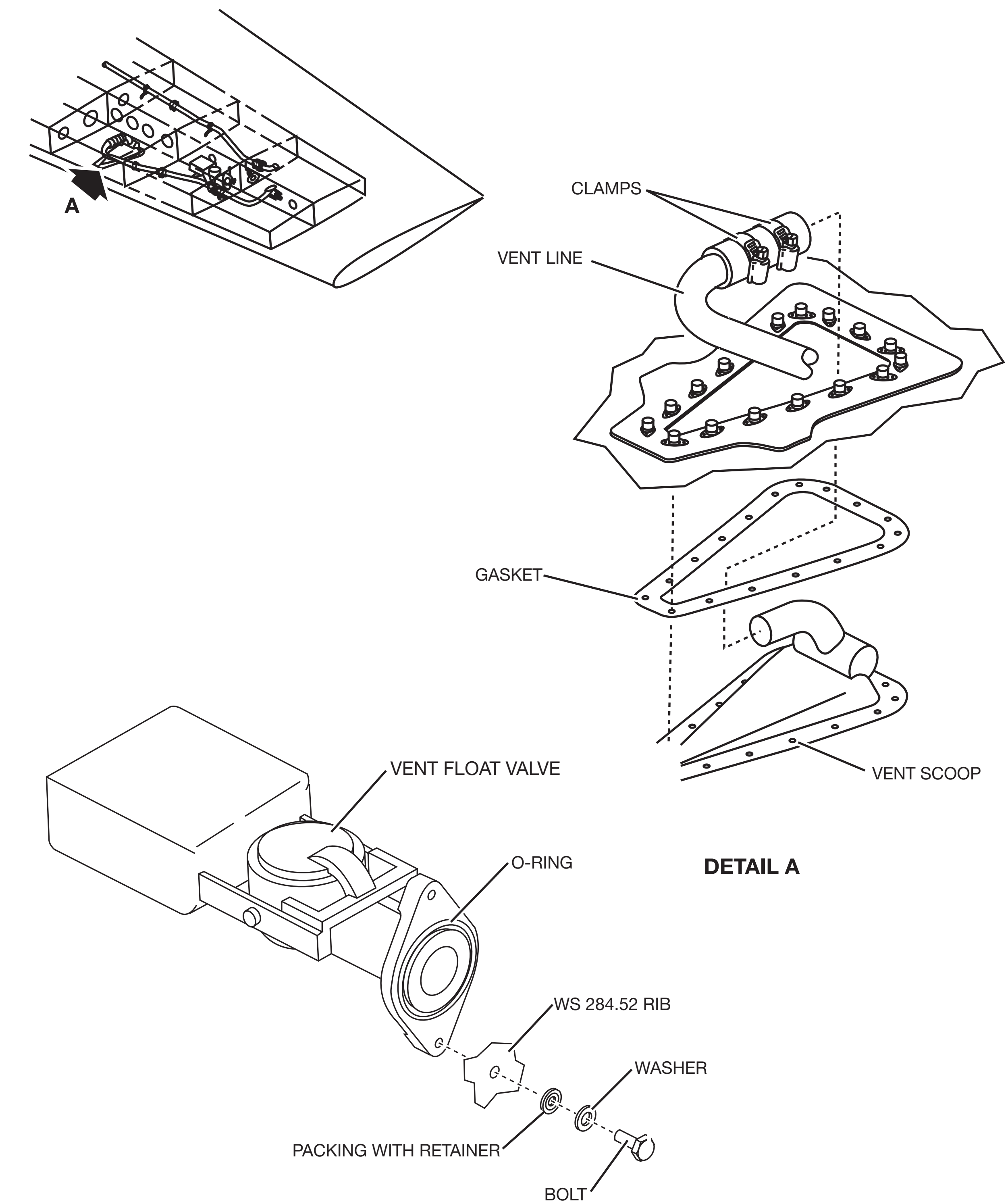


Figure 28-5. Fuel Tank Vent Components

Vent System

A ventilation system is in each wing to maintain positive internal tank pressures within the structural limitations of the wing.

The system consists of (Figure 28-5):

- Vent Line
- Surge (Vent) Tank
- Vent scoop assembly
- Relief valve
- Relief valve stand pipe
- Vent Float Valve
- Flapper-type Check Valves

Vent Line

The vent line extends from the surge tank to the sump area. The inboard end of the line is open and provides an entry for air if the check valves and float valves fail in the closed position. If the airplane is parked on a sloping ramp, such that a vent float valve is closed, fuel expansion will force fuel through the open end of the vent tube and out the vent scoop, thus preventing pressure buildup.

Surge Tank

The surge tank is semi-isolated from the remainder of the wing fuel tank, and does not normally contain fuel. The surge tank functions as a fuel collector for relatively small amounts of fuel that can become trapped in the climb vent line during flight maneuvers or climb attitudes (or during thermal expansion of the fuel). The surge tank is vented to the atmosphere by a vent scoop on the lower wing surface. The vent scoop connects to the surge tank with an open ended tube at a high point in the surge tank. This prevents fuel from siphoning overboard. It also prevents fuel from spilling overboard during wing low conditions of flight or uncoordinated turns.

Relief Valve

The relief valve is a combination positive/negative relief fuel valve in each wing (at WS 221.82). The relief valve protects the fuel tanks from over pressurization—either positive or negative—when pressure refueling, or as vent backup in case of a vent system failure. The relief valve uses surrounding internal fuel tank pressure to open itself when the internal fuel tank pressure has reached a preset level above or below ambient air pressure.

Vent Float Valve

The vent float valve allows air to either enter or leave the fuel cell. It is the primary vent for level attitudes, including for refueling and defueling. The valve is float-actuated so that whenever fuel moves to the wing tip for any reason, the valve closes preventing fuel flow into the surge tank.

NOTE

A wing vent system pressure leak check and/or wing tank leak test must be performed after any major maintenance of the wing vent system, or when wing fuel tank is completed or when proper operation of the vent system is suspect.

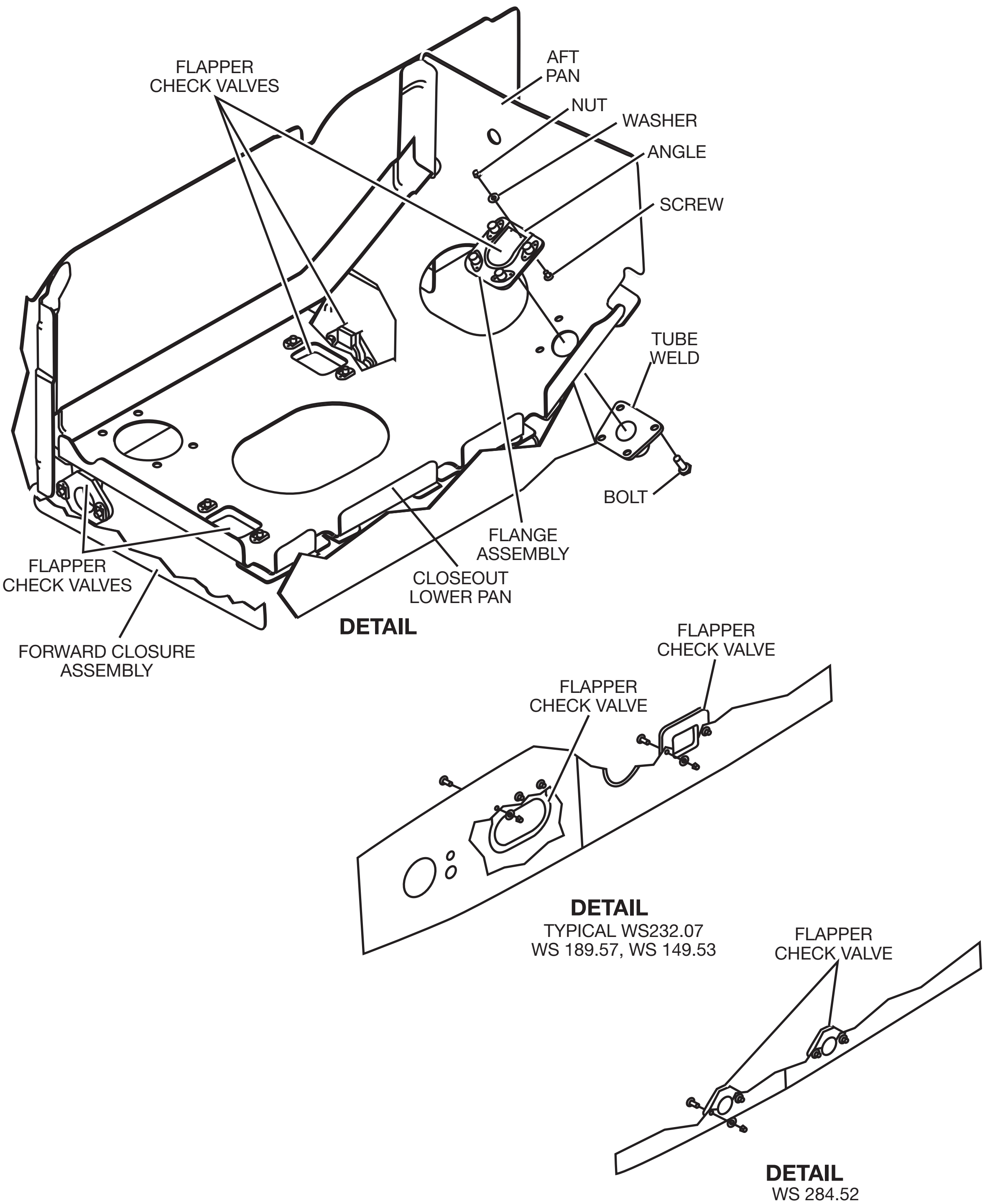


Figure 28-6. Flapper–Type Check Valves

Flapper-type Check Valves

Flapper-type check valves are as follows:

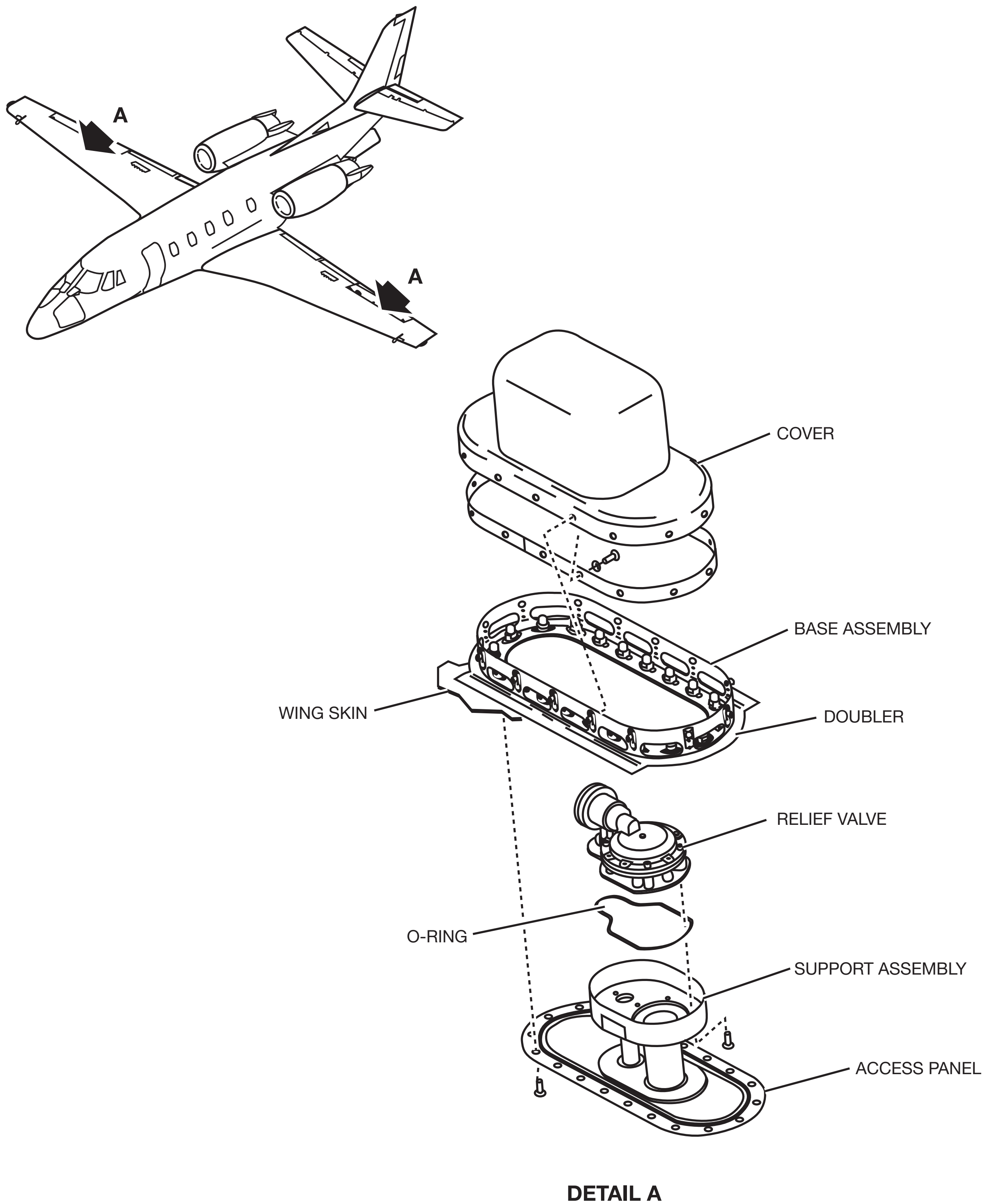
- Thirteen flapper-type check valves are in the fuel tanks of each wing.
- Two check valves are in the rib at WS 284.52 leading into the surge tank area of the wing.
- Two check valves each are in ribs at WS 232.07, WS 189.57, and WS 149.57.
- The remaining five check valves are in the wing sump area (Figure 28-6).
- Two of the sump area check valves are in the engine feed hopper closeout assemblies at FS 346.00 and FS 359.00.
- The three remaining check valves are on the engine feed hopper; two on the lower pan and one on the aft pan at the flange assembly attached to the tube extending from the aft scavenge ejector.

Depending on the location of the check valve to be removed, gain access through either the access plates/panels on the bottom of each wing, or on the aft access door on the engine feed hopper, or on the access panel that supports the boost fuel pump.

WARNING

Do not apply sealer to flapper-type check valves or to flapper-type check valve seating surfaces.

NOTES



DETAIL A

Figure 28-7. Relief Valve Installation

Relief Valve

There is a combination positive/negative (pressure/vacuum) relief valve in the left and right wing fuel panel (on the wing access panel at WS 221.82). The relief valve protects the airplane from overpressurization of the wing fuel tanks (either positive or negative) during pressure/single-point refueling. This relief valve is also used as a backup in case of fuel system vent failure (Figure 28-7).

The relief valve is a fast acting valve that opens to relieve overpressure of the airplane wing fuel tanks (positive or negative) in the event that a malfunction of the single-point refuel/defuel operation occurs. Specifically, if the automatic refuel/defuel shutoff system fails or if the fuel vent system is unable to adequately relieve internal pressure, the valve relieves the overpressure. The valve uses surrounding internal fuel tank pressure to open itself when the internal fuel tank pressure has reached a preset level (above or below ambient air pressure). After the pressure equalizes, the relief valve automatically resets itself.

The relief valves are functionally tested for proper operation and to make sure they open at specific positive/negative pressures before they are initially installed on the airplane.

NOTES

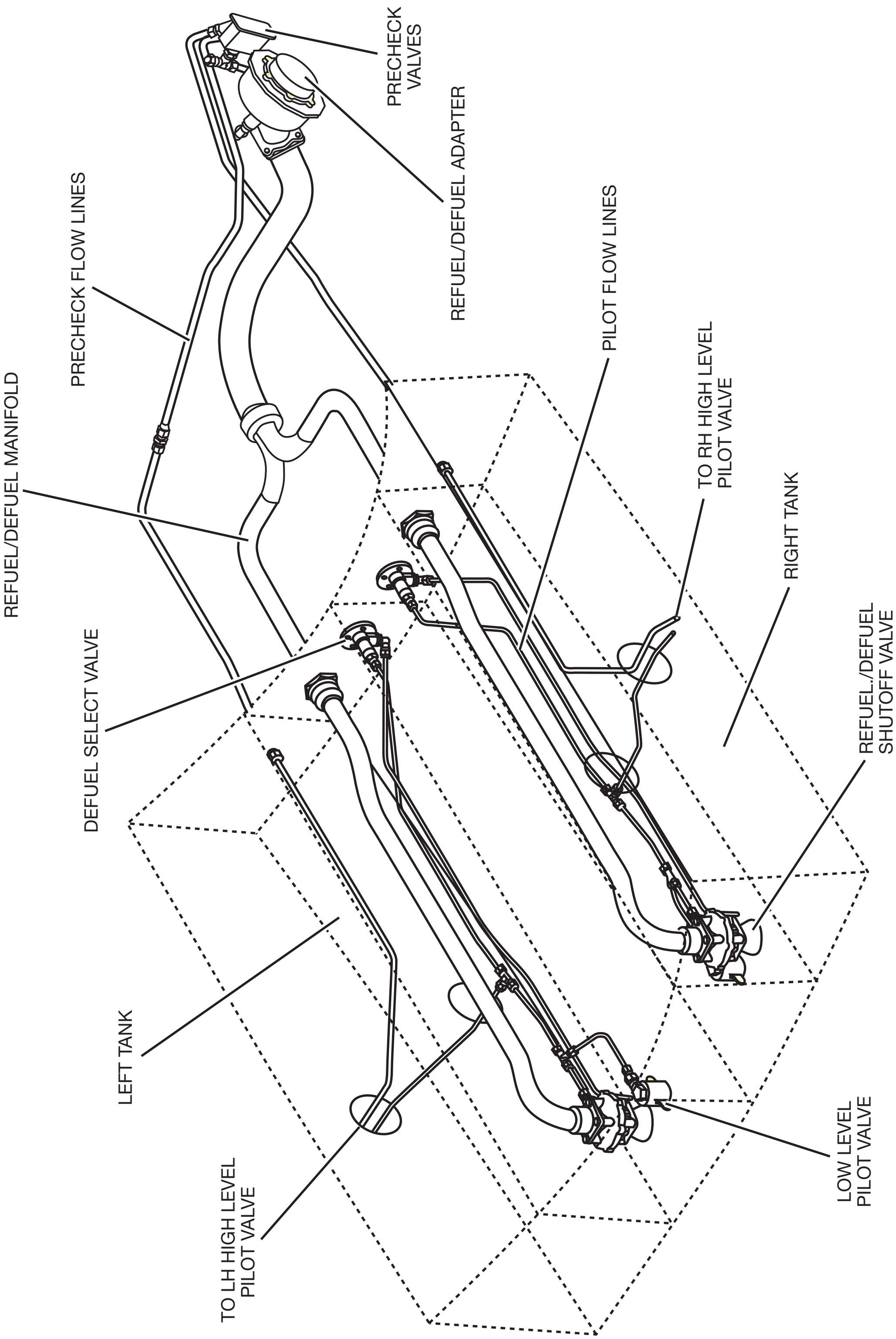


Figure 28-8. Single Point Refueling/Defueling System

SINGLE POINT REFUEL/DEFUEL SYSTEM

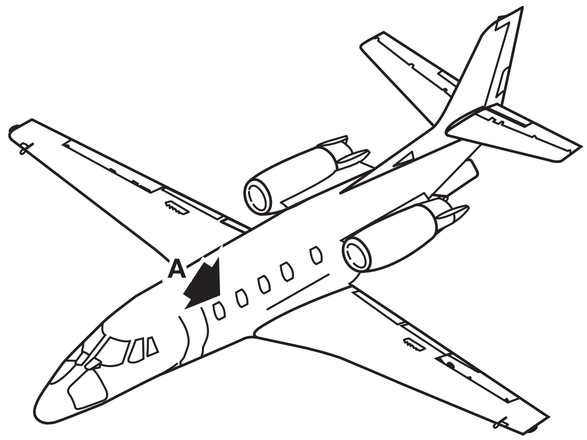
The single point refuel/defuel system (sometimes identified as pressure refueling) is used to pressure refuel and defuel the left and right wing fuel tanks from a single refuel/defuel receptacle (Figure 28-8).

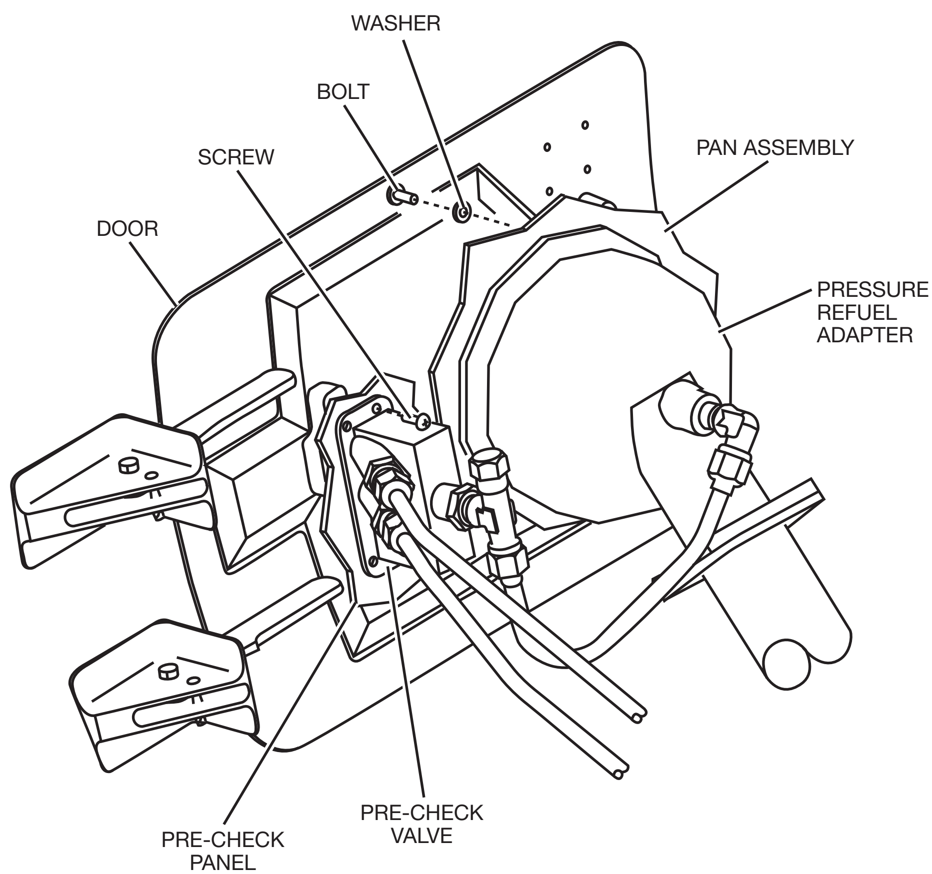
Advantages of single point refueling/defueling are:

- Less time spent refueling or defueling
- Fuel contamination
- Airplane skin damage
- Static electricity hazards
- Fuel contact with personnel

The single point refuel/defuel system is independent of the airplane fuel system. Refueling and defueling operations are accomplished at the pressure refuel adapter (receptacle) in the single point refuel/defuel compartment.

NOTES





DETAIL A

Figure 28-9. Refuel/Defuel Compartment Components

DESCRIPTION

The major components of the single point refuel/defuel system are:

- Single point refuel/defuel compartment
- Precheck panel
- Pressure refuel adapter housing
- Refuel/defuel shutoff valves
- Low level pilot valves
- High level pilot valves
- Refuel select valves

COMPONENTS

Refuel/Defuel Compartment

The single point refuel/defuel compartment is forward of the right wing on the fuselage. It contains the pressure refuel adapter and the precheck panel (Figure 28-9).

Precheck Panel

There is one precheck panel in the single point refuel/defuel compartment. There are two precheck valves on the panel. The precheck panel has two levers that the operator uses to control precheck flow to each tank. The flow comes from the auxiliary port on the pressure refuel adapter housing and flows to the selected high level pilot valve precheck port.

Refuel Adapter Housing

The pressure refuel adapter housing consists of an adapter and housing for the adapter, that connects the refueling equipment to the airplane. The adapter contains a spring loaded coupling valve so that no fuel will be lost in the coupling process. The housing has a port in it to supply precheck flow to the precheck valve.

Refuel/Defuel Shutoff Valves

Two refuel/defuel shutoff valves—one in each wing tank—shut off fuel flow during refueling or defueling. The spring-loaded refuel/defuel shutoff valves open by either positive refuel or negative defuel pressure to allow refueling/defueling through the same valve. Part of the refuel flow is bypassed to the pilot line. During refueling, when the pilot port flow is cut off, the increased back pressure closes the respective valve. During defueling, when the pilot port is opened to tank pressure, the respective valve closes.

NOTES

Figure 28-10. Single Point Refuel/Defuel Panel

Low Level Pilot Valve

A low level pilot valve is at the low point each wing tank (two valves per airplane) (Figure 28-10). This float-operated valve is in a bracket, attached to the bottom part of the refuel/defuel shutoff valve. Defueling is enabled when the fuel level lifts the float and blocks off the pilot line port. Defueling terminates when the fuel level lowers to the point where the float drops, opening the pilot port to tank pressure. The valve has a ball check that closes under refuel pressure.

NOTES

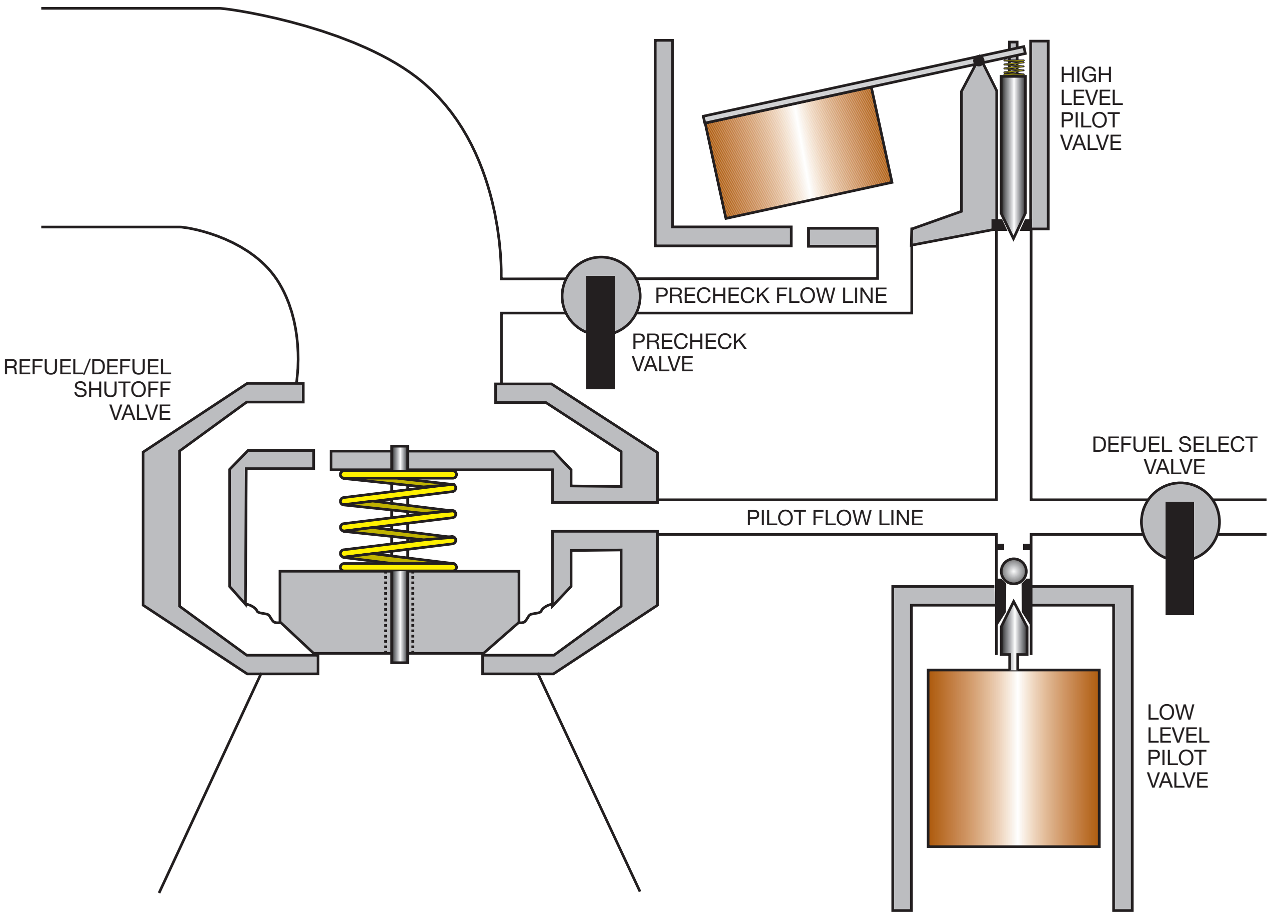


Figure 28-11. Single Point System Operation

High Level Pilot Valve

In each tank, a high level pilot valve is attached to the forward side of the main spar (at the full fuel level). This valve is operated by a float and needle valve and has a pilot (sensing) and a precheck port (Figure 28-11). The pilot port is connected to the refuel/defuel shutoff valve and the precheck port is connected to the precheck panel. The high level pilot valve shuts off the corresponding pilot flow (closing the refuel/defuel shutoff valve) when the precheck flow or the full tank fuel level fills the float bowl. The valve has a floating needle valve that closes under negative defuel pressure. The high level pilot valve operates in the following instances:

- During single-point pressure refueling, incoming fuel fills the wing tanks. When fuel reaches the high level pilot valve's float chamber, it closes the pilot port causing a pressure build-up in the pilot line. If the pressure build-up closes the refuel/defuel shutoff valve for the respective wing.
- During a refuel precheck, fuel directed through the precheck port (of the high level pilot valve) fills the float chamber, simulating a full wing tank—regardless of the actual fuel level in the tank. When the float actuated needle valve closes the pilot port, a pressure build-up in the pilot line closes the refuel/defuel shutoff valve.

Defuel Select Valves

There are two defuel select valves, one for each wing tank, are on the front spar. When either the left or right tank defuel select valve is closed, the associated refuel/defuel shutoff valve activates defueling of the tank. When either of the defuel select valves are open, the corresponding refuel/defuel shutoff valve deactivates. This is accomplished by relieving the refuel/defuel shutoff valve pilot port, thus not allowing a negative pressure to unseat the defuel valve poppet.

NOTES

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OPERATION

Refueling

To accomplish single-point pressure refueling, connect the refuel equipment to the airplane pressure refuel adapter (receptacle) in the single point refuel/defuel compartment. Fuel is delivered to both wings or to each wing independently. Opening the precheck valve for a wing prevents refueling that respective wing.

Prior to beginning a refueling operation, a precheck of the system is accomplished at the precheck panel, adjacent to the pressure refuel adapter. A precheck of the system ensures proper operation of the refueling components, including automatic refuel shutoff.

To direct fuel to the precheck port of a wing's high level pilot valve, open the left or right precheck valve. The fuel fills the float bowl faster than it can flow out, regardless of the fuel level in the tank. When the high level pilot valve float becomes buoyant, the float operated needle valve seats to close off the pilot flow in the wing tank. The fuel pressure in the pilot line closes the refuel/defuel shutoff valve as it would if the tank were full. Close the precheck valves to continue refueling.

CAUTION

If refuel flow does not stop during the precheck, refueling must be immediately terminated.

Pressure limits are shown on a placard at the single point pressure refuel adapter (receptacle).

Minimize duration of wing precheck operation when the wing tanks are full; extended precheck flow could cause tank overflow.

At the start of fuel flow, fuel is directed through a common manifold to each wing tank's refuel/defuel shutoff valve. Fuel pressure opens the spring-loaded refuel/defuel shutoff valves, delivering most of the fuel to the wing tanks. A small quantity is bypassed to the high level pilot valve. As the fuel level reaches the high level pilot valve, a float operated needle valve seats to close off the pilot flow. This builds pressure on the back side of the refuel/defuel shutoff valve. The resulting force imbalance closes the refuel/defuel shutoff valve and discontinues fuel flow. When one wing fuel tank is full and the flow has shut off, the opposite wing receives the full refueling flow until it is also full.

Defueling

For single-point defueling, connect the refueling equipment to the pressure refuel adapter. The manual defuel select valve (for a tank not requiring defueling) must be open. When either defuel select valve opens, the corresponding refuel/defuel shutoff valve deactivates. Relieve the pilot port of the refuel/defuel shutoff valve to keep negative pressure from unseating the pressure valve poppet. With the single-point refuel/defuel equipment, application of negative pressure causes the selected wing tank refuel/defuel shutoff valve to open. Fuel is drawn from the tank, through the open refuel/defuel shutoff valve, into a storage reservoir. Defueling terminates when the fuel level lowers to the point where the low level pilot valve float drops—opening the pilot port to tank pressure and causing the refuel/defuel shutoff valve to close.

CAUTION

Defueling requires equipment with adequate suction and hose stiffness.

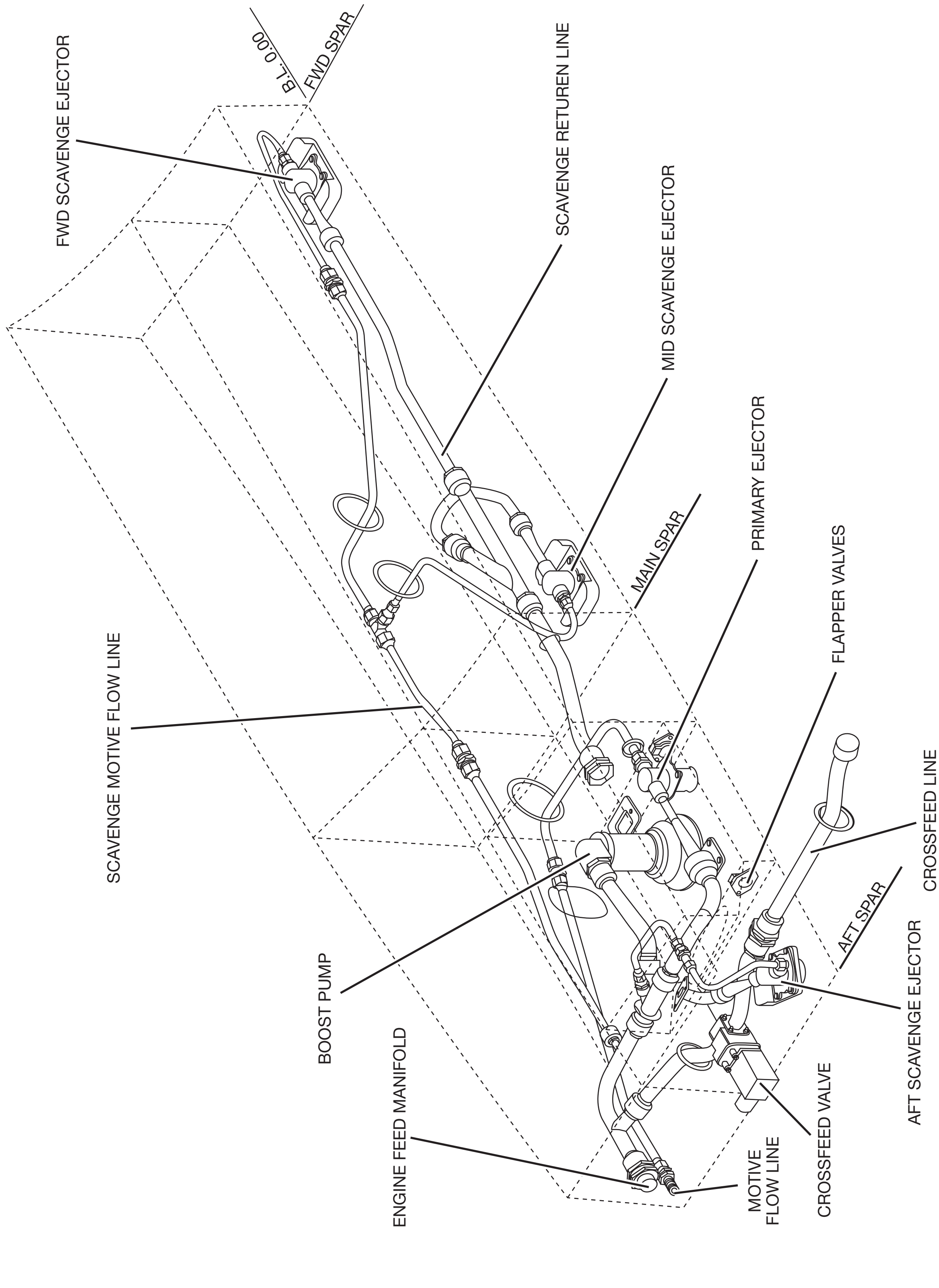


Figure 28-12. Fuel Supply and Crossfeed Components

FUEL DISTRIBUTION

NORMAL ENGINE FEED SYSTEM

Description

The fuel distribution system is divided into the normal engine feed system and the engine crossfeed system. The fuel system has the following capabilities:

- Supplying each engine from its respective tank
- Supplying either engine from the opposite tank
- Supplying both engines from the same tank
- Transferring fuel from one tank to the other

The fuel scavenge components supply fuel from the wing tank to the engine feed hopper (Figure 28-12).

The normal fuel feed system for each wing consists of:

- Four ejector type pumps
- One primary (motive flow)
- Three scavenge transfer pumps
- Electric boost pump
- Engine fuel firewall shutoff valve
- Flow check valves

The engine crossfeed system includes:

- Crossfeed valve
- Motive flow shutoff (solenoid) valve
- Crossfeed line

During operation, the crossfeed system obtains pressure from the electric boost pump of the tank selected.

NOTES

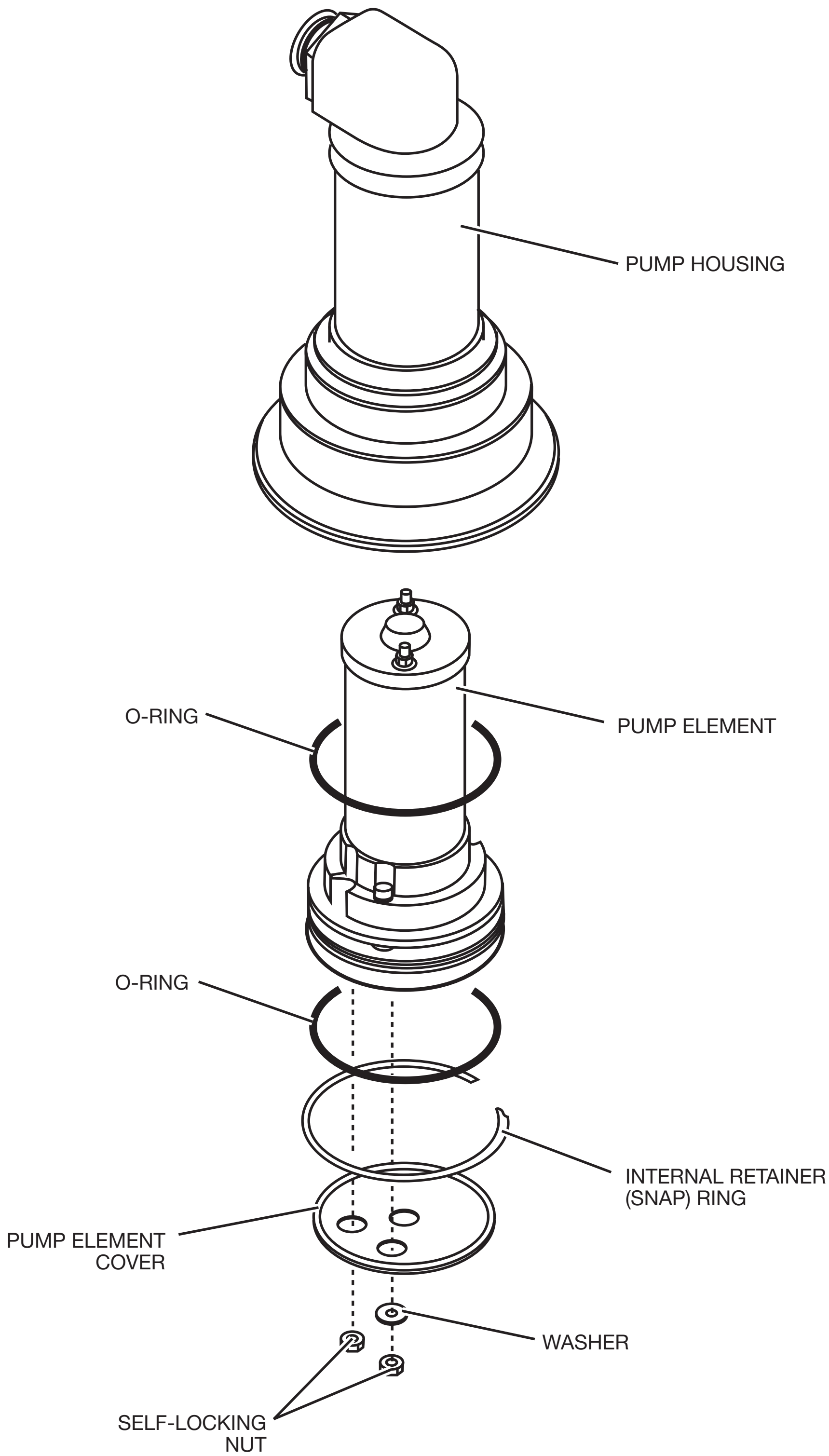


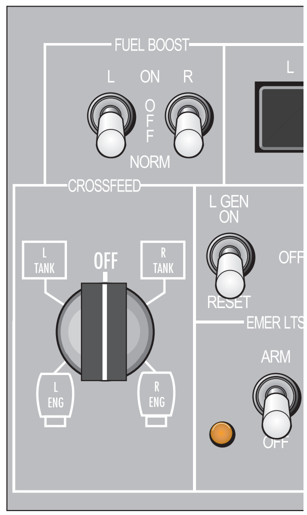
Figure 28-13. Electric Boost Pump

Components

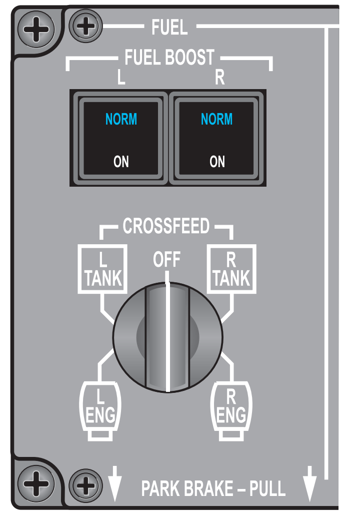
Electric Boost Pump

There is one boost pump in the engine feed hopper of each wing tank (Figure 28-13). Each electric boost pump is accessed through access panels on the bottom surface of each wing. There is also an access door panel on wing rib WS 11.50 that allows access into the engine feed hopper (through the fuel access hole). The pump is a fully submerged canister type with a field replaceable centrifugal pumping cartridge element. It is driven by an integral 28 VDC motor. The boost pump supplies fuel to its respective engine during engine start, crossfeed, APU only operation (normally in the right wing tank), and when there is a primary ejector pump failure.

NOTES



XL/XLS



XLS+

Figure 28-14. Fuel Boost Switch Panels

FUEL BOOST PUMP ON L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White	SIPI		

The amber message is displayed when the fuel boost pump is on, fuel pressure is low, and the throttle is not in cutoff. Once the amber message is displayed, it will remain latched until the fuel pressure becomes normal and the fuel boost pump is off. This message is inhibited during start and when the engine is not running.

The white message is displayed when the fuel boost pump is selected on, APU running, or not turned on by low fuel pressure.

When the boost pump is on, the EICAS receives the same 28V signal which drives the pump, and it posts the message. When the pump is off, the EICAS reads a ground through the resistance of the pump. For I/Os for throttle in cutoff and low fuel pressure, see the FUEL PRESSURE LOW message.

FUEL BOOST PUMP ON L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White	SIPI		

The white message is displayed when the fuel boost pump is selected on, APU running, or not turned on by low fuel pressure. Refer to amber EICAS message for details.

L/R FUEL BOOST
Steady illumination indicates the respective boost pump is receiving power. Steady illumination occurs during normal operations. These operations include:
1) Manual selection ON
2) Automatic activation during engine start, or
3) Crossfeed operations.

Flashing illumination occurs when the boost pump is activated because of low fuel pressure. All automatic activations require the FUEL BOOST switch to be in the NORM position.

XL/XLS ANNUNCIATOR

XLS+ CAS MESSAGES

Figure 28-15. Fuel Boost Pump Messages

There is a boost pump switch for each of the boost pumps (Figure 28-14). The switch controls the boost pump through its three positions: ON, OFF, and NORM. Operation in each switch position is as follows:

ON—The boost pump operates, regardless of other switches or sensors. The L(R) FUEL BOOST light is illuminated (Figure 28-15).

OFF (XL/XLS)—The boost pump does not operate regardless of other sensors.

NORM—The boost pump does not normally operate. Fuel flow for the engine and scavenge ejectors is provided by the primary ejector at engine speeds of idle and above. The boost pump automatically operates in the following circumstances:

- During engine start until N2 reaches the starter cutout speed of 44.7 percent.
- During auxiliary power unit start. or operation when the right engine is not operating.
- When low fuel pressure is detected by the fuel pressure switch, and the throttle is above the cutoff position. Operation continues until the switch is cycled to OFF and back to NORM.
- During crossfeeding, the boost pump operates for a tank that is selected as the feed tank.

NOTE

The cartridge element for the boost pump motor and impeller can be replaced without tank entry or defueling.

NOTE

The O-ring on the pump element may cause resistance when attempting to remove pump element. Rotate the pump element while applying downward pressure to free pump element from pump housing.

NOTES

Figure 28-16. Ejector Pumps

Primary Ejector Pump

There is one primary ejector pump in the engine feed hopper of each wing tank. The primary ejector pump operates with the scavenge ejector pumps as a matched pumping system for supplying the engine with a continuous supply of fuel at the required pressure and flow rate (Figure 28-16). The primary ejector pump is powered by high-pressure 425 to 725 psig motive flow from the engine driven fuel pump. This also provides low-pressure motive flow for the three scavenge ejector pumps. There is a check valve in the pump discharge fitting to prevent backflow through the pump.

Scavenge Ejector Pumps

Each wing tank has three scavenge ejector pumps in the sump area. They are ejector-type pumps that operate continuously and utilize motive flow from the primary ejector pump's discharge flow. The forward scavenge ejector pump is just aft of the forward spar. The mid scavenge ejector pump is just forward of the main spar. The aft scavenge ejector pump is just forward of the aft spar. Since the scavenge ejector pumps are strategically located they provide a continuous flow of fuel to the engine feed hopper, keeping it full in all normal flight attitudes.

The scavenge ejector inlets and feed hopper gravity inlets are protected by large area screens of wire mesh that minimize contamination reaching the hopper and fuel system components.

Check Valves

There are three check valves in the fuel flow lines of each wing. Two of the check valves are downstream from the forward and mid scavenge ejector pumps, and the remaining check valve is downstream from the primary ejector pump. The check valves for the forward and mid scavenge ejectors are in the scavenge return manifold, at the coupling fitting of the flow inlet.

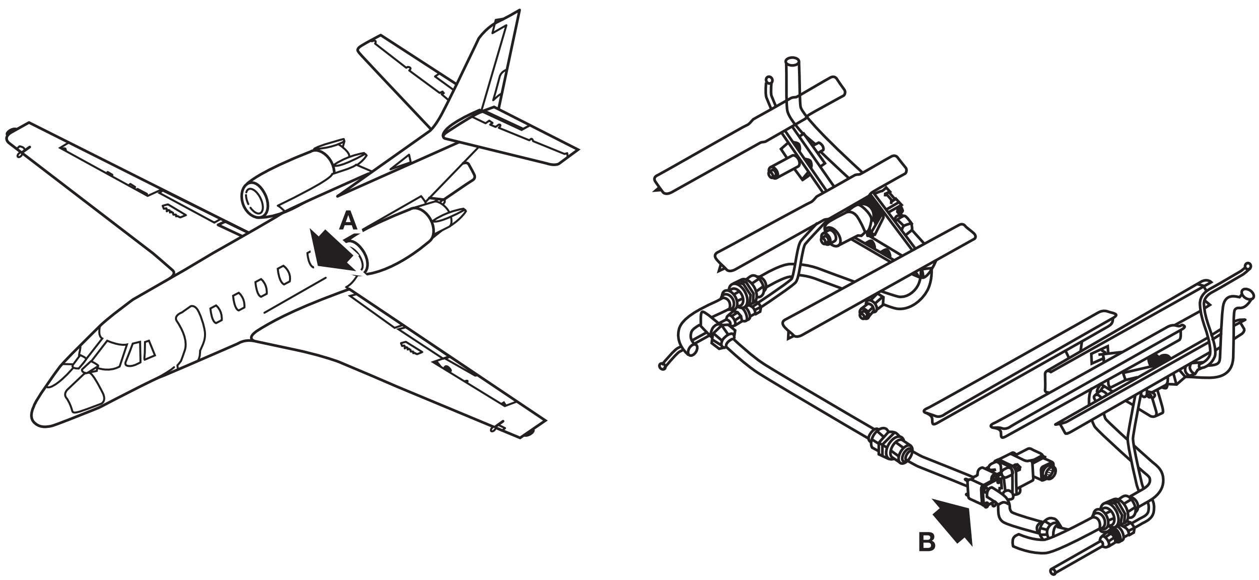
The scavenge return manifold is above the mid scavenge ejector pump. The check valve for the primary ejector pump is in the fuel flow tube assembly connected by coupling to the discharge port of the primary ejector pump.

The check valves prevent fuel pressure from reversing and entering the two scavenge ejector pumps and the primary ejector pump.

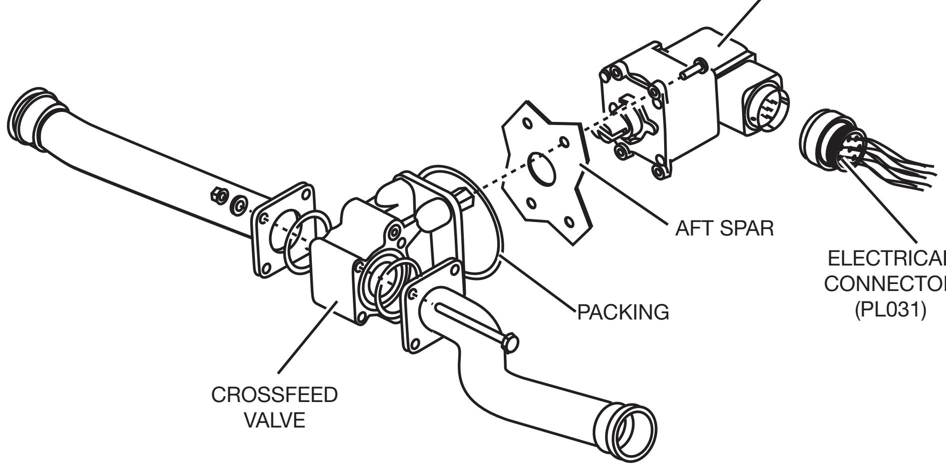
NOTE

The check valve can only be installed with proper directional fuel flow.

NOTES



DETAIL



DETAIL

Figure 28-17. Crossfeed Valve

Operation

The fuel system supplies each engine from its respective tank. It can supply either engine from the opposite tank or both engines from the same tank.

In normal operation, each engine receives fuel from its respective tank. During engine start, the electric boost pump supplies fuel to the engine. When the engine starts, high pressure fuel (motive flow) from the engine-driven fuel pump operates the primary ejector fuel pump which supplies fuel to the engine. Boost pump activation is controlled by the engine start relay circuitry when the FUEL BOOST switch on the left instrument panel is in the NORM position.

CROSSFEED SYSTEM

Description

The engine crossfeed system allows either or both engines (and the auxiliary power unit) to be fed from the primary ejector and/or auxiliary boost pumps in either tank. Crossfeed components include:

- A crossfeed valve
- Motive flow shutoff valve
- Associated plumbing

Components

Crossfeed Valve

The crossfeed valve is a motor-operated ball valve, driven open and closed during crossfeed (Figure 28-17). The valve moves from a fully open to a fully closed position, and vice versa, in 0.5 to 1.0 second. One valve, in the plumbing, connects the left and right engine feed manifolds. The two piece assembly is on the aft wing spar, with the valve inside the engine feed hopper and the motor actuator portion on the outside (in the dry bay area). This permits actuator replacement without disturbing the valve and plumbing connections and without requiring tank entry or defueling. External marking on the valve includes the nameplate and relief flow direction marking.

NOTE

The crossfeed valve actuator assembly can be removed without defueling the airplane. If fuel leakage occurs during removal of actuator assembly, then defuel airplane and replace the complete crossfeed valve.

NOTES

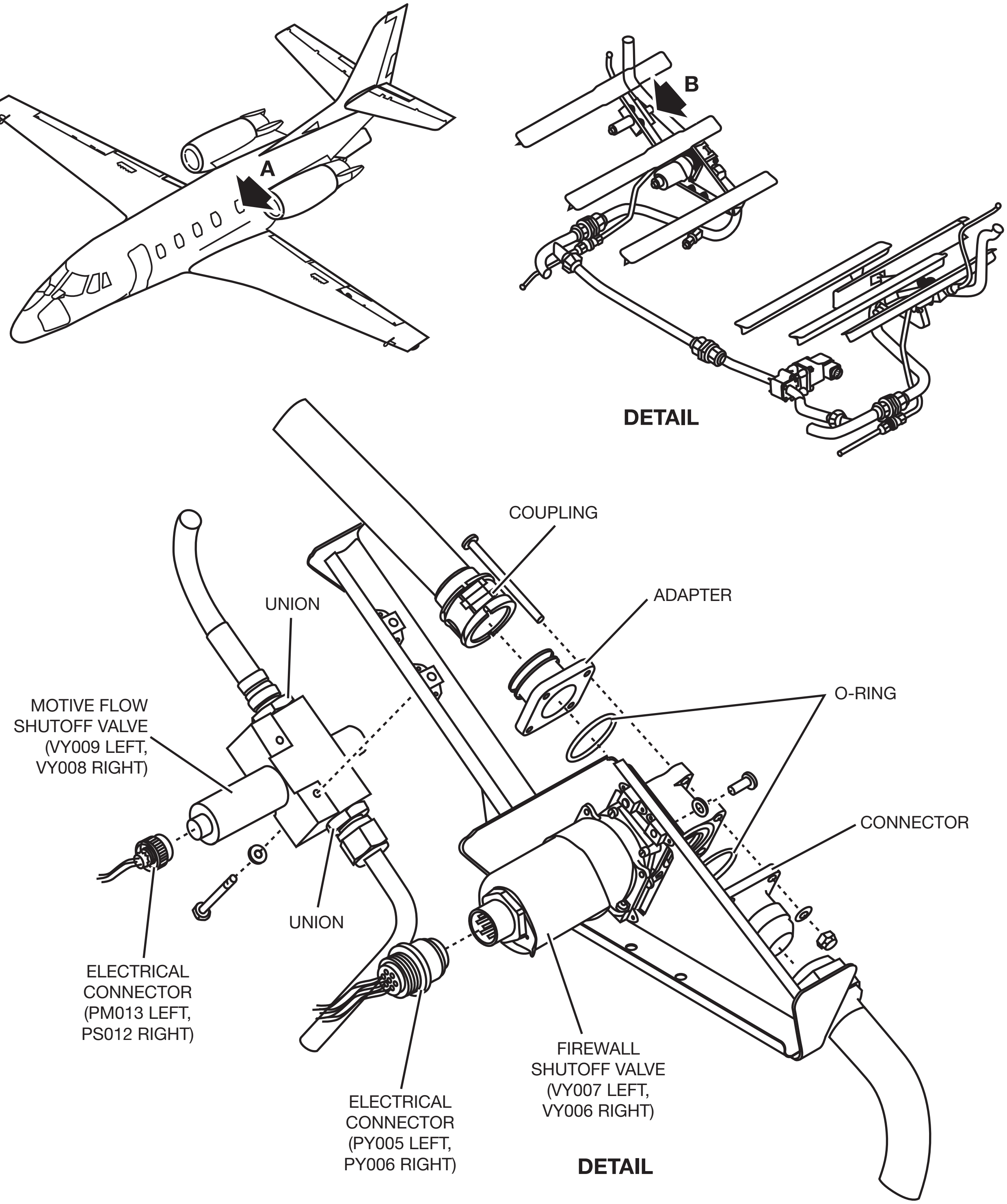


Figure 28-18. Firewall and Motive Flow Shutoff Valves

Motive Flow Shutoff Valve

There is a motive flow shutoff valve in each motive flow line to shut off the primary ejector motive flow that leads to the non-feeding tank when crossfeed is selected (Figure 28-18). It is a normally open, electrically operated solenoid valve.

NOTES

FUEL XFEED
Annunciator illuminates steady if fuel crossfeed is selected and the fuel crossfeed valve is open. Annunciator flashes and MASTER CAUTION illuminates steady if fuel crossfeed is selected off and the fuel crossfeed valve is not closed.

XL/XLS ANNUNCATOR

FUEL CROSS FEED

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*10 Second
White			

Fuel Cross Feed operation - When the fuel selector is selected to the left tank or right tank, the normal operation is to increase the fuel pressure in the tank you are cross feeding from, then open the fuel cross feed valve, and reduce the fuel pressure in the tank you are not cross feeding from.

The white message is displayed when the fuel cross feed valve is commanded open from the cockpit crossfeed switch. **The amber message is displayed when the fuel cross feed valve is not in agreement with the selected crossfeed switch position.** The white message has the standard debounce, and the amber message has a 10 second debounce.

When fuel cross feed is not selected, a ground is sent to the EICAS system from the switch in the cockpit. When cross feed is selected, an open is sent to the EICAS system. When the cross feed valve is either open or closed, one of two switches in the valve sends a 28 Volt signal to the EICAS. When the valve is neither open or closed, neither switch is made and both inputs are open.

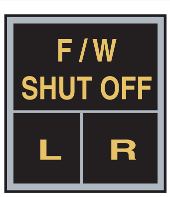
FUEL CROSS FEED

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	
White	SIPI		*Standard

The white message is displayed when the fuel cross feed valve is commanded open from the cockpit crossfeed switch. The white message has the standard debounce, and the amber message has a 10 second debounce. Refer to amber EICAS message for details.

XLS+ CAS MESSAGES

Figure 28-19. Fuel Crossfeed Messages



L/R FW SHUTOFF
Flashes to indicate the respective fuel and hydraulic firewall shutoff valves have closed and the generator field relay has tripped. This annunciation occurs after the engine fire switchlight has been pressed. All three conditions are required for the light to illuminate.

XL/XLS ANNUNCATOR

FIREWALL SHUTOFF L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	2 Second
White			

The advisory white message indicates normal operation. Refer to amber EICAS message for details.

XLS+ CAS MESSAGE

Figure 28-20. Fuel Firewall Shutoff Messages

Crossfeed Operation

To initiate the crossfeed mode, the crossfeed switch is positioned to either the right or left tank position. When the crossfeed is selected, the boost pump in the tank selected is energized and the crossfeed valve receives power and opens. Three seconds later, the motive flow valve, on the engine receiving crossfeed fuel, closes, and the FUEL XFEED annunciator (XL/XLS) or FUEL CROSSFEED CAS message (XLS+) illuminates (Figure 28-19). To terminate crossfeed operation, the crossfeed valve is placed in the OFF position. When crossfeed is turned off, the motive flow valve opens and three seconds later, the crossfeed valve closes and the boost pump shuts off. The FUEL XFEED annunciator (XL/XLS) or FUEL CROSSFEED CAS message (XLS+) then extinguishes.

Engine Fuel Firewall Shutoff Valve

The engine fuel firewall shutoff valve is behind the aft wing spar in the wing fairing area on each fuel supply line. The firewall shutoff valve is an electric motor that is operated ball valve assembly. The ball valve shuts off the fuel flow to the engine in the event of an engine fire. It is activated opened or closed by pressing the respective LH–RH ENG FIRE switchlight below the glareshield on the firetray, to the left and right of the annunciator panel. External marking on the valve includes the nameplate, valve assembly/rubber cure date, and relief flow direction marking.

The valve moves from fully opened to fully closed or from fully closed to fully opened in a maximum of one second. The operational check of the firewall shutoff valve is performed during the testing of the fire extinguisher system. When the valve is fully closed, an input is sent to the annunciator panel to indicate that it is closed. As soon as both the fuel and hydraulic firewall shutoff valves fully close on either the right or left side of the aircraft, the F/W SHUT OFF L or R annunciator (XL/XLS) or FIREWALL SHUTOFF L or R CAS message (XLS+) illuminates respectively (Figure 28-20).

NOTE

Position firewall shutoff valve so that relief flow direction arrow is pointing toward the fuel flow line extending from the aft spar (fuel tank). The arrow denotes direction of pressure relief, not direction of fuel flow.

NOTES

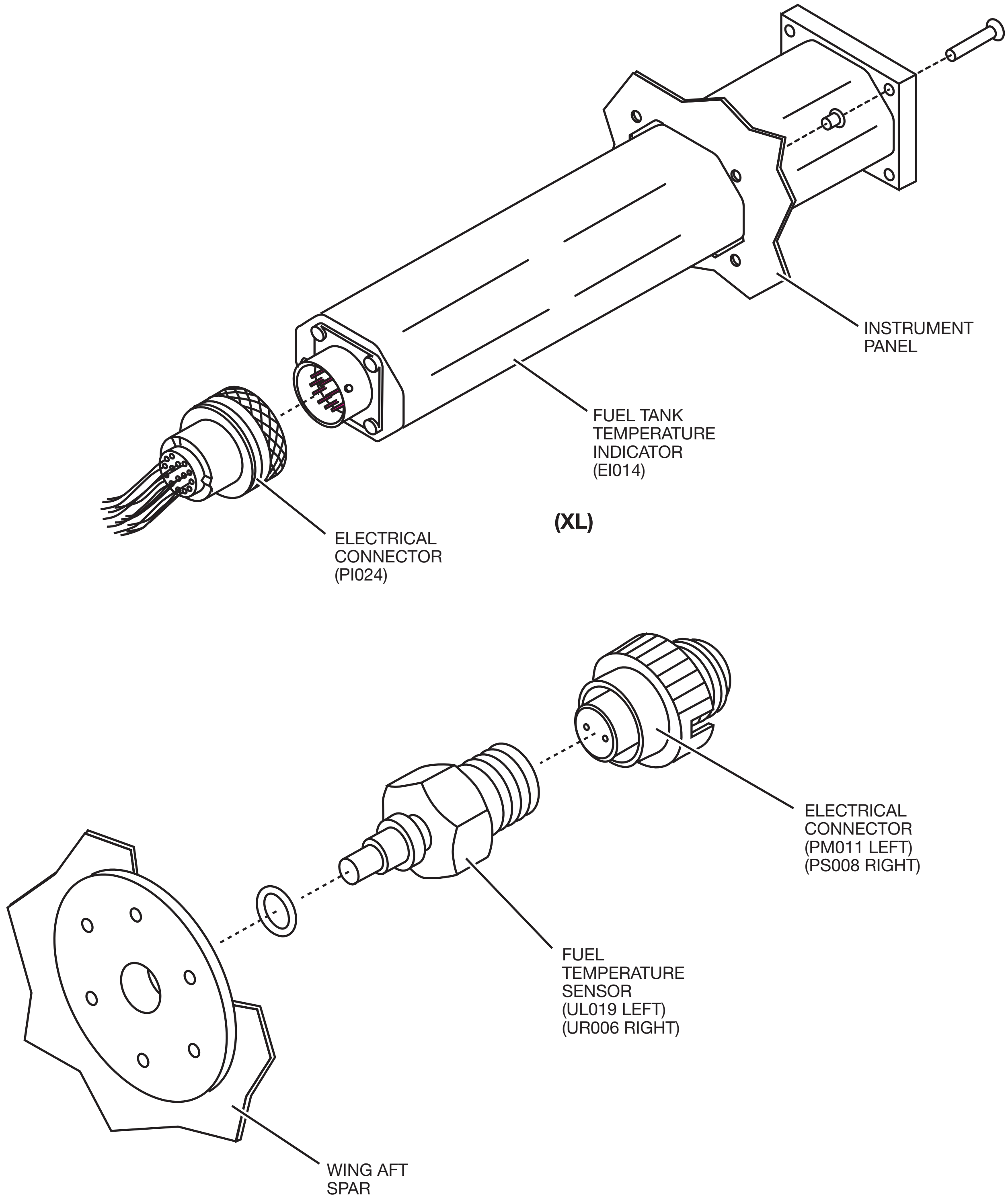


Figure 28-21. Fuel Temperature Components

INDICATING

DESCRIPTION

The fuel indicating section consists of the fuel quantity gauging system, which includes:

- Fuel system components that indicate quantity
- Temperature
- Pressure of fuel

Also included are pressure warning systems for the pumping systems in the wing fuel tanks.

The fuel quantity gauging system consists of:

- Fuel quantity system components
- Associated fuel system indicators
- Probes, switches, and annunciators

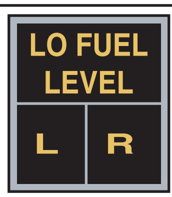
The low fuel warning system consists of two float switches and two indicating annunciator lights.

COMPONENTS

Fuel Temperature

The fuel temperature of the left and right fuel tanks is measured by a fuel temperature sensor installed through the aft spar, one on each side of center, with its temperature bulb extending into the tank area. The temperature reading is sent to the fuel temperature (FUEL TEMP) indicator (Figure 28-21) on the center instrument panel, where a temperature readout is displayed for the left and right fuel tanks. The fuel temperature indicator uses a dual liquid crystal display to indicate the left and right fuel tank temperatures. The range of the indicator is –60°C to 70°C, with a tolerance of ±3°C.

NOTES



L/R LOW FUEL LEVEL
Annunciator flashes to indicate low fuel quantity in the respective tank. (360 lbs ± 20 lbs). Measured by a float switch, activates MASTER CAUTION lights.

Limitation: The respective fuel boost pump must be turned ON.

XL/XLS ANNUNCATOR

FUEL LEVEL LOW L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*34 Second

This message is displayed when the fuel level in the fuel tank is low as determined by a float switch. When the fuel level is less than approximately 360 lbs, the float switch sends a ground signal to the EICAS system, which displays the message. When the fuel level is greater than 360 lbs, the switch sends an open to the EICAS system, which removes the message. The message has a 34 second debounce on, and a 32 second debounce off.

There are dual paths for presentation of a low fuel condition on the XLS+. In addition to the CAS message, the fuel quantity display on the MFD will turn amber and flash for ten seconds for indication of a low fuel condition. This is a Level A independent path that does not go thru the DCU.

XLS+ CAS MESSAGE

Figure 28-22. Fuel Level Low Messages

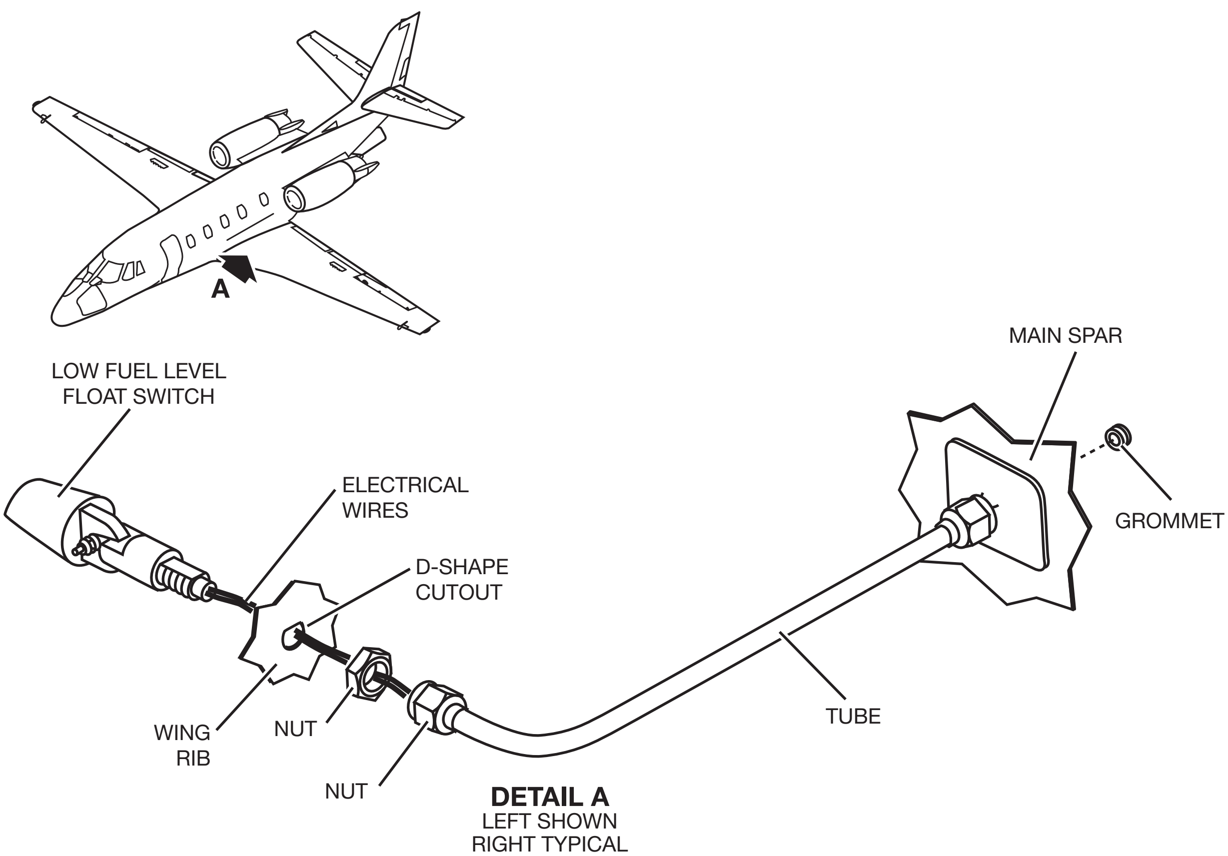


Figure 28-23. Low Fuel Level Float Switch

Low Fuel Warning

The low fuel warning system includes two low fuel level float switches, and a L–R LO FUEL LEVEL annunciator (XL/XLS) or FUEL LEVEL LOW CAS message (XLS+). One low fuel level float switch is in each wing and has its own annunciator (Figures 28-22 and 28-23). The float portion of the switch is on the inboard side of WS 34.00 in the wing fuel tank, and the electrical switch portion extends through the wing rib and into the wheel well.

When electrical power is applied to the airplane fuel indicators, the low fuel warning system becomes operational. When the fuel level in the wing fuel tank decreases to a level allowing the float to lower and actuate the electrical switch (less than 360 ± 20 pounds for 30 seconds), the amber L–R LO FUEL LEVEL annunciator (XL/XLS) or FUEL LEVEL LOW CAS message (XLS+) for that respective wing fuel tank illuminates.

NOTES

FUEL PRESSURE LOW L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
	ESDI	SIPI	

The message is displayed when the fuel pressure is low, and the respective engine is running. For the purposes of this message, engine running begins when the start contactor disengages and ends when the throttle is put into cutoff.

For I/O definition of engine start, see: Start Contactor in the power distribution system section. When the fuel pressure is low, a pressure switch provides a ground signal to the EICAS system, which posts the message. When the pressure is normal, the switch sends an open signal to the EICAS, which removes the message. Fuel cutoff is a switch in the throttle quadrant which detects if the throttle is in cutoff. When it is in cutoff, a ground is provided to the EICAS system. When it is not in cutoff, an open signal is provided.

L/R LO FUEL PRESS
Annunciator illuminates steady if the fuel system has low pressure prior to engine starts while the aircraft is on the ground. Annunciator flashes and MASTER CAUTION illuminates steady, if the fuel system has low pressure after both engines are started with aircraft on the ground or in flight and the throttle is out of cutoff. Activates MASTER CAUTION lights.

XL/XLS ANNUNCATOR

XLS+ CAS MESSAGE

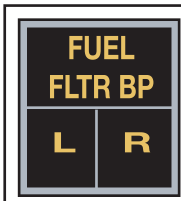
Figure 28-24. Fuel Pressure Low Messages

FUEL FILTER BYPASS L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
	*ESDI	SIPI	

This message is displayed when the fuel filter impending bypass is true. This message has two different sets of inputs that can trigger the message. A configuration strap is used to tell the DCU which set of inputs to use. The two sets of inputs are either the impeding/actual fuel bypass switches or the differential pressure transducers.

With the fuel bypass configuration strap pin grounded, the impeding and actual fuel bypass switches are used to trigger the message, They measure pressure across the fuel filter. The impending fuel bypass is set to trip at 14 +/- 2 PSID (14 PSI = 44.34 mV) and is the trigger for the CAS message, while the actual bypass is set to trip at 26 +/- 2 PSID (26PSI = 78.06 mV) and is provided for fault monitoring only (no CAS message). The typical pressure drop across the fuel filter is approximately 1.2 PSID. The fuel filter pressure relief valve will open at 32 +/- 2 PSID as measured across the fuel filter.

Without the fuel bypass configuration strap pin grounded, the differential pressure transducer is used to trigger the message. The DCU transmits differential fuel pressure, corrected for sensor excitation voltage error and filtered per PWC requirements, to the FADEC via GPBUS-5 label 346 at a 10 Hz update rate.



L/R FUEL FLTR BP
Annunciator flashes if the respective engine fuel filter bypass switch has activated to indicate an impending bypass condition due to possible filter blockage. Activates MASTER CAUTION lights.

XL/XLS ANNUNCATOR

XLS+ CAS MESSAGE

Figure 28-25. Fuel Filter Bypass Messages

Fuel Pressure

A fuel pressure switch is in the nacelle on the engine fuel supply line for each engine for sensing fuel pressure. It actuates at 5.3 psig with decreasing pressure and deactivates by 7.5 psig with increasing pressure. Actuating the switch causes the amber L–R LO FUEL PRESS annunciator (XL/XLS) or FUEL PRESSURE LOW CAS message (XLS+) to illuminate and the boost pump to operate (Figure 28-24).

Fuel Filter Bypass

An amber L–R FUEL FLTR BP annunciator (XL/XLS) or FUEL FILTER BYPASS CAS message (XLS+) advises of an impending bypass of the engine fuel filter (Figure 28-25).

NOTES

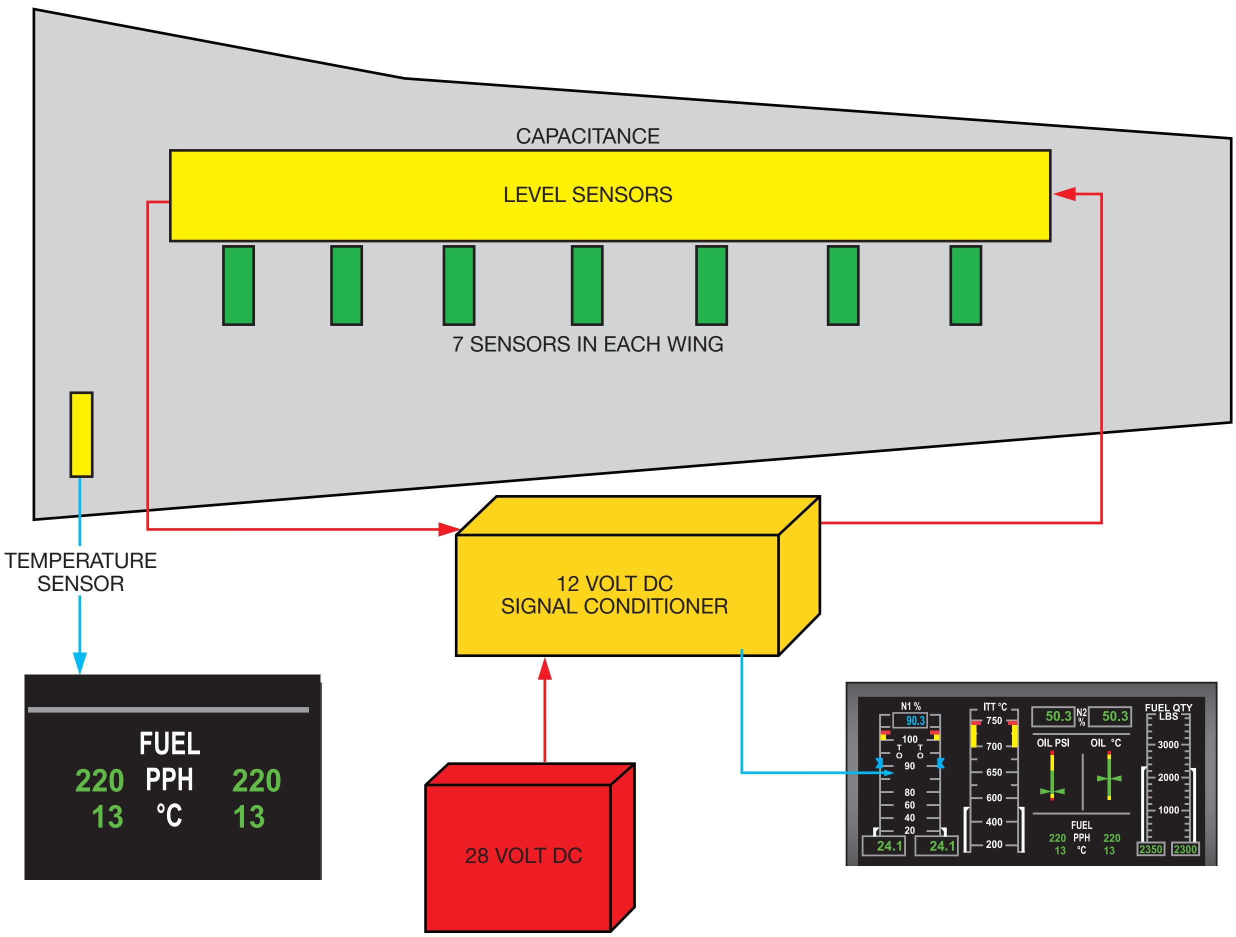


Figure 28-26. Fuel Quantity Indicating System

FUEL QUANTITY

Description

The fuel quantity indicating system is a capacitance system that includes:

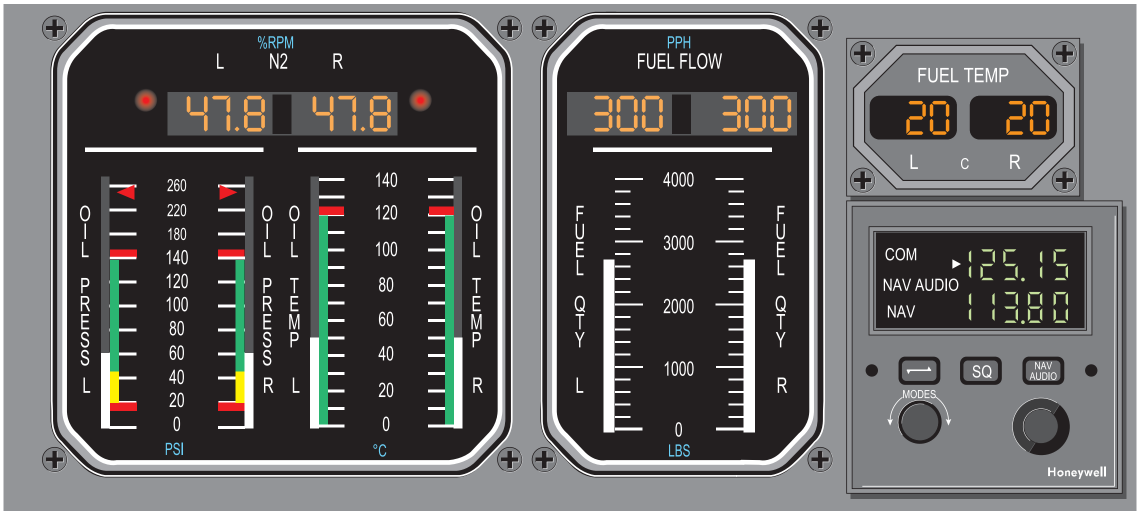
- Dual linear fuel quantity indicator
- Microprocessor base dual channel fuel quantity signal conditioner with self-test and monitoring features
- Seven fuel probes (sensing units) per wing

The fuel probes are constructed of two concentric tubes. They are nonadjustable, and function at a particular buttock line/wing station location. The fuel probes are bracket mounted to the wing ribs, perpendicular to wing datum, inside each wing fuel tank.

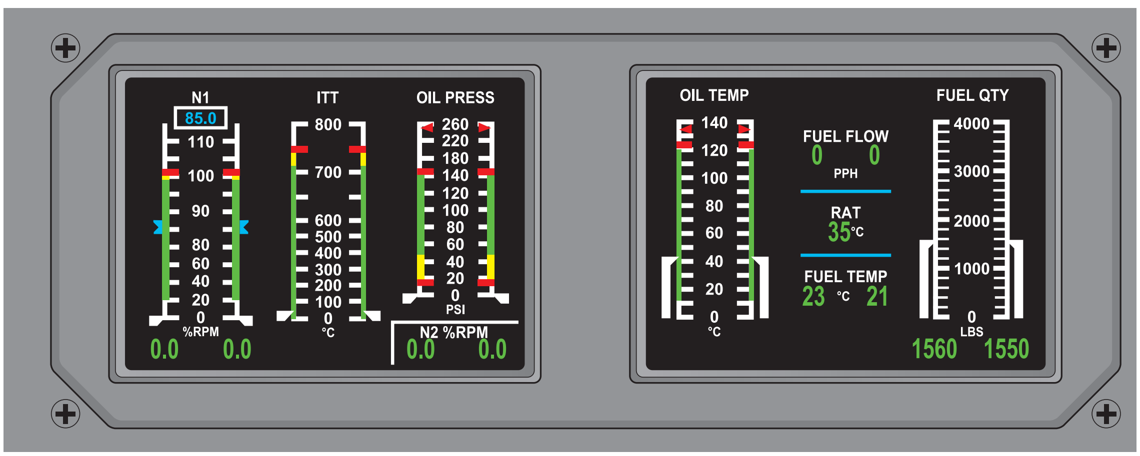
The fuel quantity signal conditioner is horizontally mounted in the pressurized cabin of the airplane in the pilot side console. The fuel quantity signal conditioner is a microprocessor based unit that has a channel for the left wing fuel system and a channel for the right wing fuel system (Figure 28-26). It interfaces with all the wing fuel probes and the fuel quantity/fuel flow indicator on the center instrument panel.

The fuel quantity/fuel flow indicator is on the center instrument panel. It indicates actual usable fuel remaining in the right and left wing fuel tanks and the fuel flow rate.

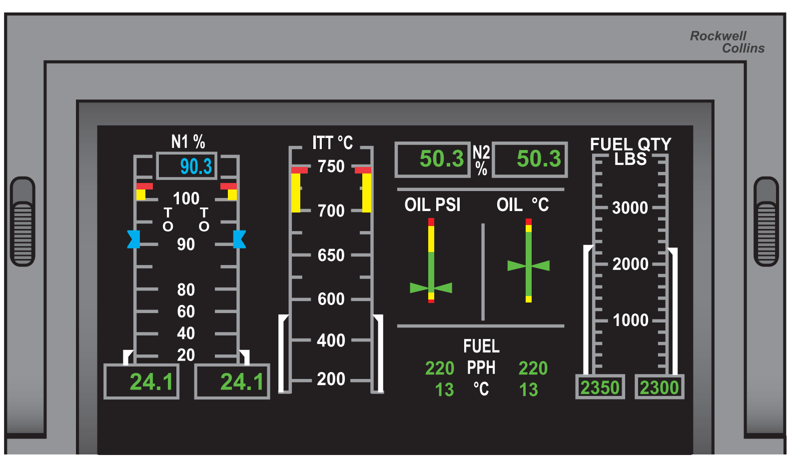
NOTES



UNITS 5001–5268



UNITS 5269–6000



UNITS 6001 AND SUBSEQUENT

Figure 28-27. Fuel Quantity Indications

Operation

The fuel probes are located so that accurate indications for the fuel volume are maintained during both level and unlevel flight attitudes. Each fuel probe has an integral electronic module that converts the capacitance of the probe to a current signal (Figures 28-27 and 28-28).

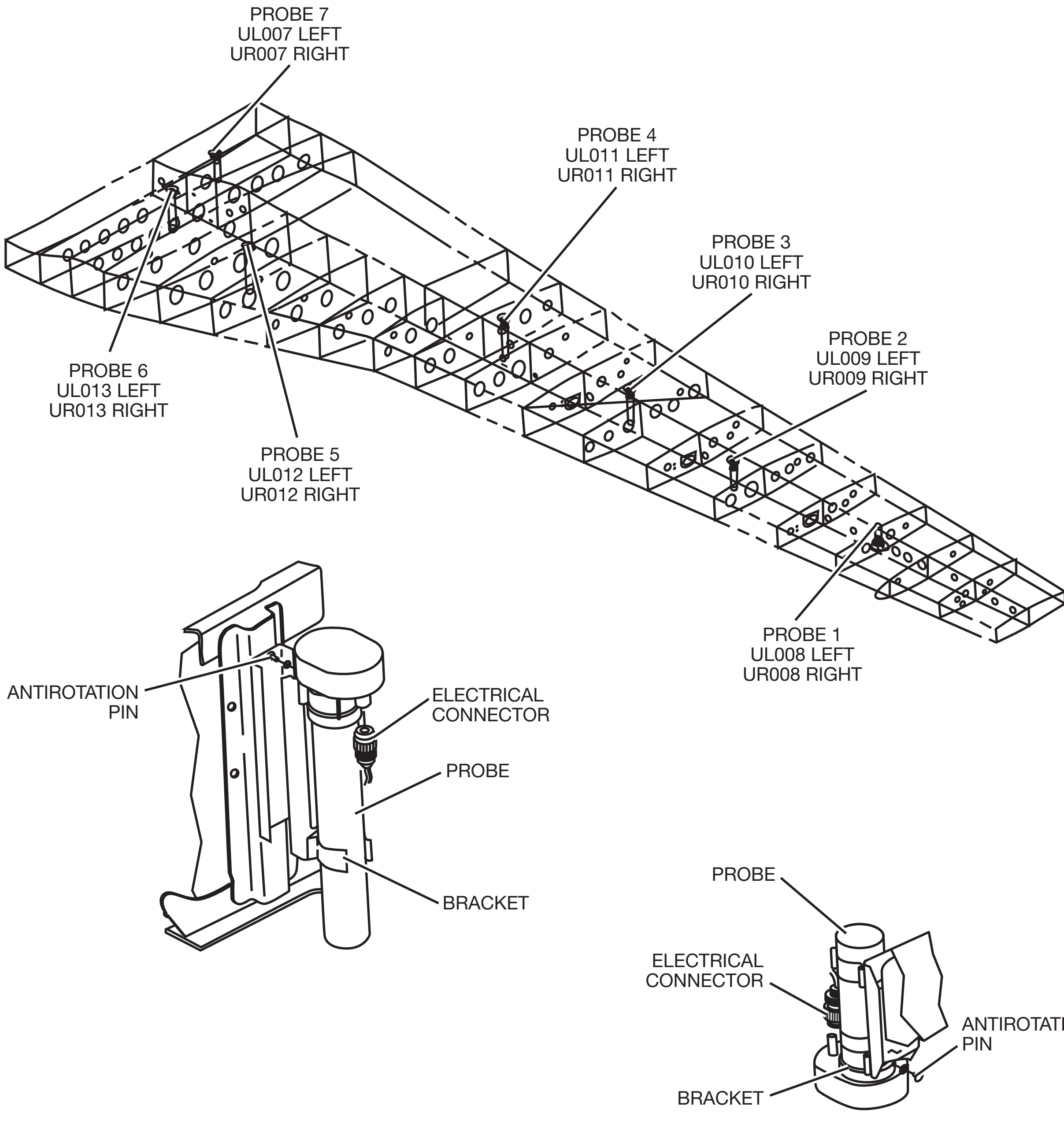


Figure 28-28. Fuel Probes

Table 28-1. BIT FAULT DESCRIPTION

FAULT DESCRIPTION	FUEL GAUGE ANNUNCIATOR	IND 0	IND 1	IND 2
NONE	OFF	OFF	OFF	OFF
SIGNAL CONDITIONER	ON	ON	ON	ON
PROBE #1	ON	ON	OFF	OFF
PROBE #2	ON	OFF	ON	OFF
PROBE #3	ON	ON	ON	OFF
PROBE #4	ON	OFF	OFF	ON
PROBE #5	ON	ON	OFF	ON
PROBE #6	ON	OFF	ON	ON
PROBE #7	ON	FLASH	FLASH	FLASH
SIGNAL CONDITIONER	ON	OFF	OFF	OFF

The fuel quantity signal conditioner provides two output signals for use by the left and right FUEL QTY indication on the fuel quantity/fuel flow indicator (Figure 28-28). The electrical output voltage signal from the fuel quantity signal conditioner consists of a voltage ranging from 0 to 5.7 VDC, where zero (0) VDC represents zero pounds of fuel and 5.7 VDC represents 3800 pounds of fuel.

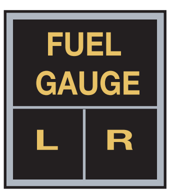
The fuel quantity/fuel flow indicator receives a voltage signal from the fuel quantity signal conditioner and converts it into a linear scale indication (its left and right FUEL QTY indication).

A built-in test (BIT) function of the fuel quantity signal converter checks each fuel probe signal for validity. A failure, and its type of failure, is annunciated on the fuel quantity signal conditioner by three light emitting diodes (LED). A detected failure also illuminates the L and/or R FUEL GAUGE annunciators (XL/XLS) or FUEL GAUGE CAS messages (XLS+) (Figure 28-29). Fault handling also checks for circuit faults in the fuel quantity signal converter, and for faults in the fuel probes (Table 28-1). If a failure is detected, the channel discrete BIT fault output will be turned ON, and the BIT status LEDs will display a pattern that identifies the failure. The BIT fault out remains on and the appropriate BIT status LED pattern continues to display until power is removed from the fuel quantity signal converter.

CAUTION

The fuel probe mounting bracket utilizes a protruding fastener (rivet) to align with the fuel probe antirotation hole. The protruding fastener in the fuel probe mounting bracket must mate with the hole in the fuel probe for proper installation.

NOTES



L/R FUEL GAUGE
Annunciator flashes to indicate the respective fuel quantity gauging system has detected a fault. Activates MASTER CAUTION lights.

Note: Record the signal generator BITE indications prior to securing electrical power.

XL/XLS ANNUNCATOR

FUEL GAUGE L-R			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
This message is displayed when there is a fault in the fuel quantity indicating system, as determined by the fuel quantity signal conditioner. When the signal conditioner detects a failure, it sends a ground signal to the EICAS system, which posts the message. When the signal conditioner is in normal operation, it sends an open to the EICAS, which removes the message.			

XLS+ CAS MESSAGE

Figure 28-29. Fuel Gauge Messages

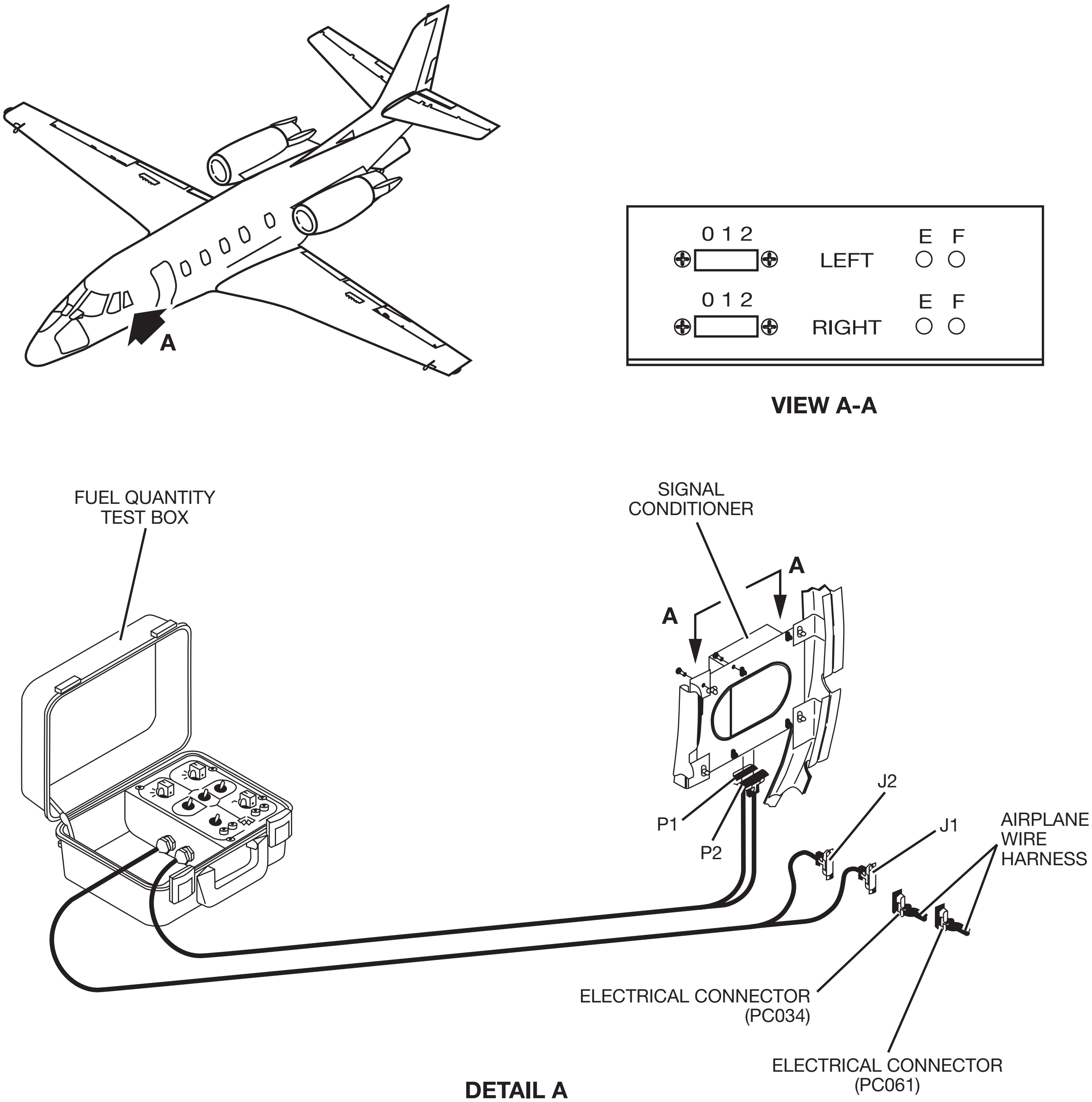


Figure 28-30. Fuel Quantity Test Box Connection

Diagnostics

Fuel Quantity Indicating Troubleshooting

Troubleshooting the fuel quantity indicating system (quantity capacitance indicating) primarily uses the LED indication on the fuel quantity signal conditioner (Figure 28-30), that is generated during its BIT and performing the system functional tests.

The design of the fuel tank units, with no moving parts, are very reliable and relatively maintenance free. The fuel quantity/fuel flow indicator and the electrical interconnect cable are more likely to sustain a malfunction than the wing fuel tank units.

NOTES

QUESTIONS

1. In the event of a loss of main DC power while operating in crossfeed:
   A. The crossfeed valve fails closed.
   B. Crossfeed continues.
   C. The LO FUEL PRESS L or R annunciator illuminates.
   D. The motive flow valve for the receiving side fails open and X-feed terminates.

2. During initial engine starting, the primary source of fuel pressure to the engine-driven pump is:
   A. Motive flow fuel pressure.
   B. Primary ejector pump pressure.
   C. Respective side electric boost pump pressure.
   D. Suction pressure from the engine driven pump.

3. The primary ejector fuel pump:
   A. Provides motive flow fuel pressure.
   B. Provides head pressure to the engine-driven fuel pump.
   C. Provides high pressure, low volume fuel to the engine-driven fuel pump.
   D. Is located in the surge tank.

4. During initial engine start, the electric boost pump is activated when the:
   A. Start button is depressed
   B. Throttle is advanced from cutoff to idle
   C. Placing the boost pump switch to ON
   D. Fuel low pressure switch

5. During over-the-wing fueling:
   A. Fill the wing tanks until fuel fills the standpipe.
   B. It is not necessary to ground the refueling apparatus.
   C. Fill the wing tanks until fuel reaches the bottom of the standpipe.
   D. None of the above.

6. Select the correct choice regarding single point pressure refueling:
   A. Immediately after fuel flow has stabilized, perform a precheck test.
   B. A fuel flow precheck test is not required if a partial load of fuel is desired.
   C. Extreme care must be observed when attaching the fueling nozzle in order not to spill fuel.
   D. The refueling/defueling compartment is located directly forward of the left wing.

7. Opening a defuel select lever:
   A. Allows defueling the corresponding wing tank
   B. Prevents defueling the opposite wing tank
   C. Prevents refueling the corresponding wing tank
   D. Prevents defueling the corresponding wing tank

8. With total loss of DC power, the motorized fuel crossfeed valve will:
   A. Fail in the OPEN position
   B. Fail in the CLOSED position
   C. Fail in its present position
   D. Return to a RESET position

9. With the BOOST PUMP switch in the NORMAL position, the boost pump:

A. Automatically activates during crossfeed
B. Only activates during fuel crossfeed
C. Runs continuously
D. Only activates during engine start

10. To crossfeed fuel on the ground:

A. A GPU is required for power, because the aircraft battery switch must be OFF
B. The aircraft battery switch must be in the ON position
C. Is impossible
D. One engine must be operating

11. (XL/XLS) If the right boost pump switch, in the cockpit, is in the OFF position and the pilot attempts to start the right engine:

A. Right boost pump would come on when the right throttle is taken out of cutoff and the right engine would start
B. Right boost pump automatically comes on when the right start button is depressed because a low fuel pressure condition exists during start
C. Right boost pump would not come on, causing a hung start due to a lack of fuel
D. Right boost pump would come on automatically

12. During an engine start, the fuel pressure switch opens at:

A. 7 psi, causing the LO FUEL PRESS annunciator to illuminate
B. 7 psi, causing the LO FUEL PRESS annunciator to extinguish
C. 5 psi, causing the LO FUEL PRESS annunciator to illuminate
D. 5 psi, causing the LO FUEL PRESS annunciator to extinguish

13. During crossfeed operation, right tank to left engine:

A. Crossfeed valve motors open, right boost pump comes on, and left motive flow shutoff valve closes
B. Crossfeed valve motors open, right boost pump comes on, and left motive flow shutoff valve opens
C. Crossfeed valve motors open, left boost pump comes on, and left motive flow shutoff valve closes
D. Crossfeed valve motors open, right boost pump comes on, and right motive flow shutoff valve closes

14. After selecting crossfeed OFF, the white FUEL XFEED annunciator remains illuminated and begins to flash after 10 seconds. This would be an indication of:

A. Normal system operation.
B. Crossfeed valve is not fully open.
C. Boost pump switches are in the OFF position. The boost pump switches must be in the NORM position in order to crossfeed.
D. Crossfeed valve did not close.

15. When defueling a Citation Excel using the single point system:

A. Defuel levers must be in their normal stowed, vertical position
B. One defuel lever must be up and one down, to defuel both tanks
C. Both defuel levers must be placed in the up or horizontal position
D. Precheck levers must be actuated to the defuel position

16. Minimum recommended fuel load for running engines at full power, with any interior removed:
    A. 3,000 pounds total
    B. Any quantity
    C. 1,000 pounds total
    D. 2,000 pounds total

17. (XL/XLS) With the battery switch on, engines not operating, boost pump switches in the OFF position, and throttles out of cutoff:
    A. Boost pumps come on automatically due to low fuel pressure
    B. Boost pumps do not operate
    C. Boost pumps come on because one of the functions of the cutoff switch is to turn the boost pumps on
    D. (In this configuration) Boost pumps ONLY come on when the boost pump switches are selected to the ON position

18. The fuel probes in each wing of the aircraft:
    A. Can be interchanged with its counterpart in the opposite wing
    B. Can be installed correctly with either end up
    C. Are wired in series
    D. All of the above

19. The signal conditioner for the fuel indicating system is :
    A. In the sump area with the boost pump
    B. In the pilots left sidewall, aft of the circuit breakers
    C. In the wing, mounted in each fuel probe
    D. Inside the fuel quantity indicator

20. What action should be taken if the FUEL GAUGE annunciator illuminates?
    A. The bite lights, in the conditioner must be checked prior to turning off the battery switch.
    B. The fuel quantity indicator must be replaced, due to time life limits.
    C. The fuel quantity must be recalibrated.
    D. No action needed. The system is operating normally.

21. (XL/XLS) During an engine start, which of the following would indicate a failure of the electric fuel boost pump?
    A. FUEL BOOST ON annunciator would illuminate
    B. LO FUEL PRESS annunciator remains illuminated
    C. Fuel flow would decrease
    D. Engine would shut down

22. The primary purpose of the scavenge ejector pumps are to:
    A. Transfer fuel from wing to wing
    B. Backup the primary ejectors in case of failure
    C. Supply fuel to the sump area where the boost pump and primary ejector are located
    D. To supply high pressure fuel to drive the primary ejector

23. The primary ejector pump is:
    A. Electric-driven pump, located in the sump area of each wing
    B. Fuel driven pump located in the sump area of each wing
    C. Supply fuel to the sump area where the boost pump and primary ejector are located
    D. Fuel driven pump used to prime the electric driven boost pump

24. With the battery switch in BATT, engines running, and boost pump switches in the NORM position:

A. Boost pumps run continually
B. Boost pumps only run while crossfeed is selected
C. Boost pumps automatically come on if low fuel pressure occurs
D. None of the above

25. The LO FUEL LEVEL annunciator illuminates:

A. When 1 hour of fuel remaining is sensed
B. Immediately after 360 pounds is indicated
C. To indicate 360 pounds after a 30 second time delay
D. Using a switch located in the hopper tank

CHAPTER 29
HYDRAULIC POWER

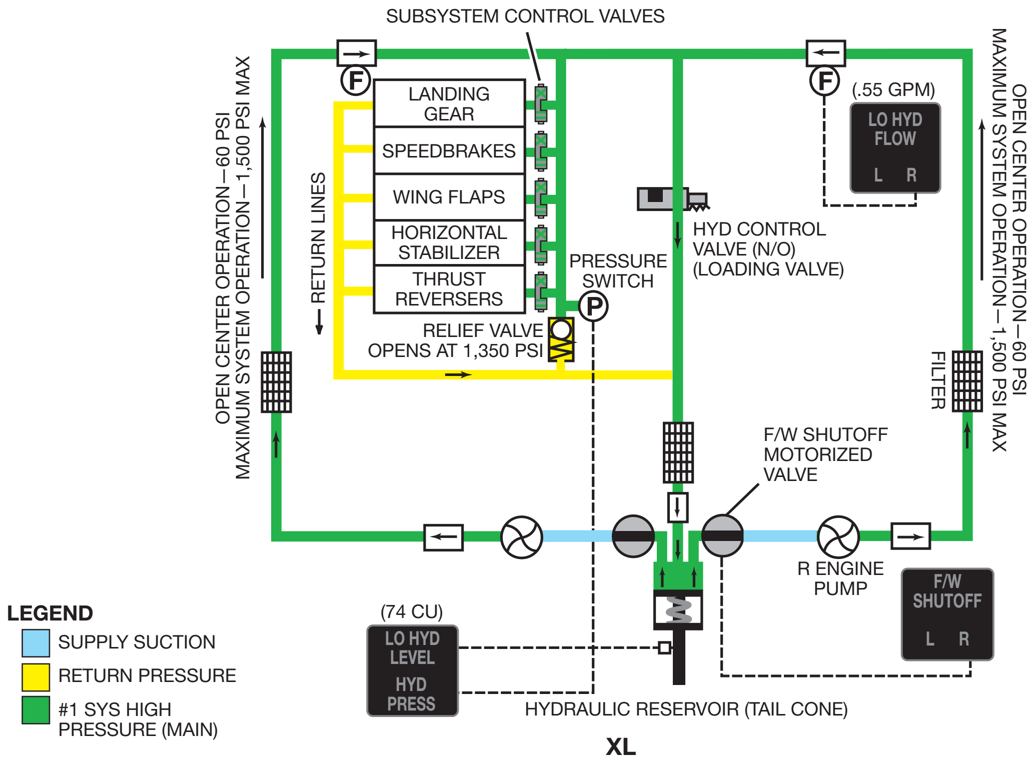
CONTENTS

ILLUSTRATIONS

CHAPTER 29
HYDRAULIC POWER

INTRODUCTION

This chapter presents the hydraulic system for the 560XL/XLS/XLS+ Citation aircraft with special emphasis given to components and their operation. General maintenance considerations arc included, with an introduction to functional and operational checks. References for this chapter and further specific information can be found in Chapters 5—“Time Limits/Maintenance Checks,” Chapter 12—“Servicing,” and Chapter 29—“Hydraulic Power,” of the *Aircraft Maintenance Manual (AMM)*.



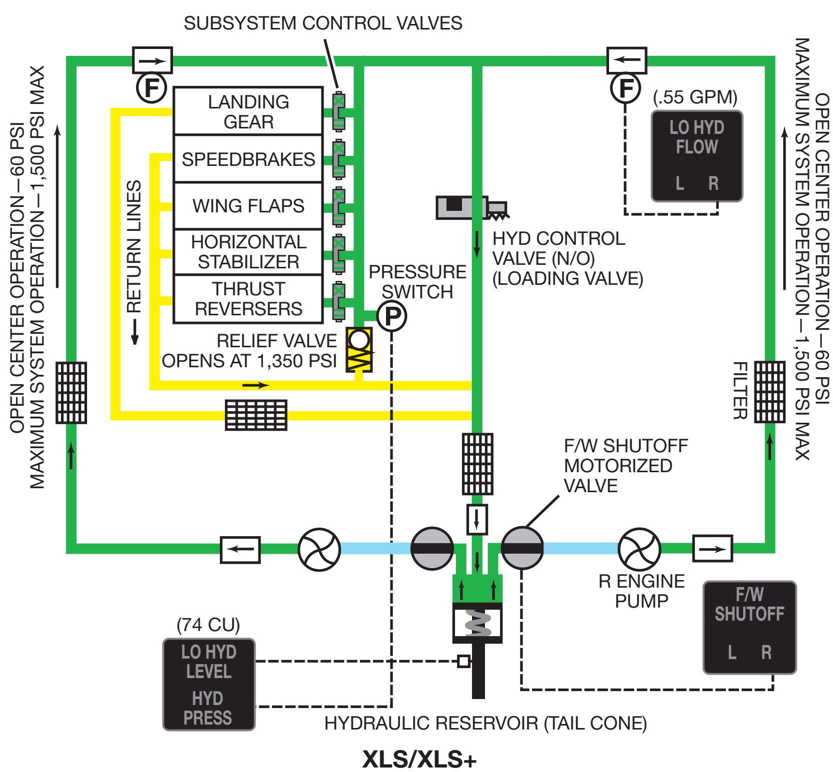


Figure 29-1. Hydraulic System Schematic—Open Center XLS/XLS+

GENERAL

The hydraulic power system operates the landing gear, speedbrakes, and thrust reversers, in addition to the flaps and horizontal stabilizer actuator.

The system includes:

- Hydraulic reservoir
- Firewall shutoff valves
- Hydraulic pumps
- Panel and filter components

The hydraulic system is classified as "open-center" because fluid continually circulates between the hydraulic pumps and the reservoir at approximately 60 psi, when there is no demand on the system (Figure 29-1). When a demand is made for system pressure by initiating operation of a subsystem, a bypass valve closes causing the pressure to increase. Pressure is determined by the system relief valve and does not exceed 1,500 psi. The system remains pressurized until the subsystem being actuated completes its cycle. It then depressurizes as the bypass valve opens. A separate independent system is employed for the main wheel antiskid/power brake system.

CAUTION

Phosphate ester base hydraulic fluid is used in the main hydraulic power system and the antiskid/ power brake system, which requires additional safety precautions to be followed and adhered to when accomplishing work on the systems. Long exposure to phosphate ester base hydraulic fluid can cause skin chapping and dehydration. Eye contact with phosphate ester base hydraulic fluid can cause extreme tearing and a burning sensation.

NOTES

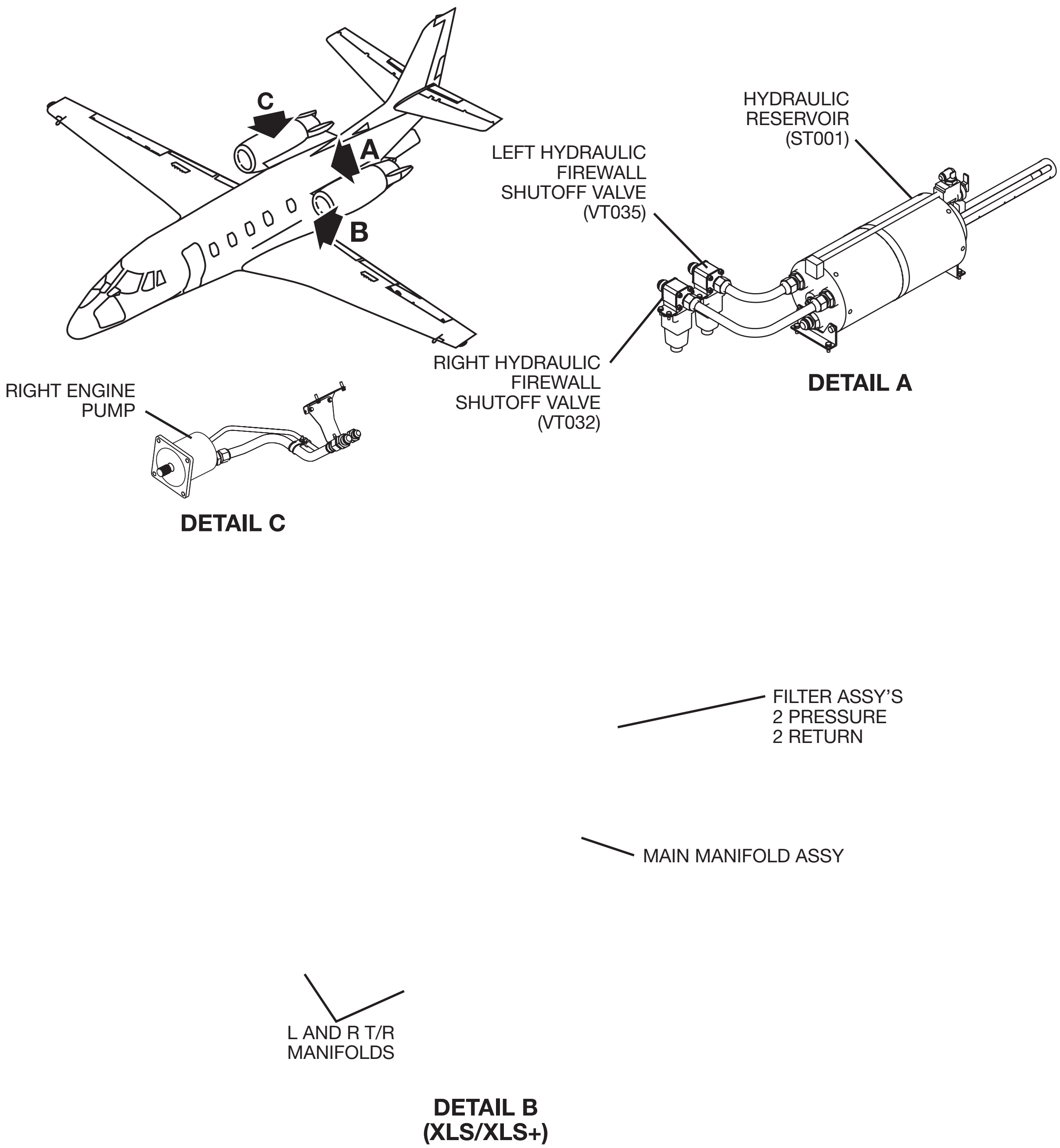


Figure 29-2. Hydraulic System Components

MAIN HYDRAULIC SYSTEM

DESCRIPTION

This section provides maintenance information on the portion of the hydraulic system that is used to store and deliver hydraulic fluid to using systems.

The main hydraulic system is an open center type system (Figure 29-2). Two engine-driven pumps (one on each engine) supply a continuous flow of hydraulic fluid, as long as the engine(s) is operating. A reservoir (ST001) stores fluid for the various hydraulically operated components. The reservoir is a bootstrap type and performs in a manner similar to a hydraulic accumulator by maintaining potential pressure on the system. A solenoid operated load valve (VY044) controls the open center operation of the system. In a no-pressure-demand condition, the load valve is de-energized open, allowing the fluid to free-flow from pressure to return. In a pressure demand condition, the load valve is energized closed and pressure is routed to a selected system/component. A relief valve limits the hydraulic system pressure to 1,500 psi. The relief valve is on the hydraulic sub-panel. The load valve is installed on the hydraulic filter panel.

Other main system components include:

- Two pressure filters
- One in the left engine pump pressure line
- One in the right pump pressure line

A third return filter is in the return line to the hydraulic reservoir. Two firewall hydraulic shutoff valves (VT032 left and VT035 right) are motorized electrically closed or opened. Either shutoff valve may be closed during an engine fire, stopping the flow of fluid to the engine pump selected. Ground service connections are at the tailcone lower exterior surface.

A flow switch check valve (SY001 left and SY002 right) is incorporated in each pressure line from the engine hydraulic pumps. The check valve prevents fluid flow from one engine pump to the other. The flow switches provide an indication on the annunciator panel (UF002) when low or no-flow occurs from the respective engine pump.

OPERATION

Hydraulic fluid flow is provided by two engine-driven hydraulic pumps. Hydraulic pressure is provided by the closing (energizing) of the load (open center) valve upon demand, during the:

- Operation of the landing gear extension/retraction
- Flap extension/retraction
- Speedbrake extension/retraction
- Operation of the thrust reverser

A pressure relief valve limits the pressure in the selected system to the maximum system operating pressure. The pressure relief valve begins opening at 1,350 psi and is fully opened at 1,500 psi (maximum hydraulic system pressure). In a no-demand condition, the load valve is open (deenergized) and fluid flows from pressure to return. A hydraulic reservoir provides storage for fluid not required by hydraulic actuated systems. Fluid flows from the reservoir to (and through) the left and right engine-driven pumps. The fluid returns to the reservoir through a return line or by return flow from an operating system component. The hydraulic reservoir is pressurized whenever the engine-driven pumps are operating or when an external hydraulic service unit is connected and operating.

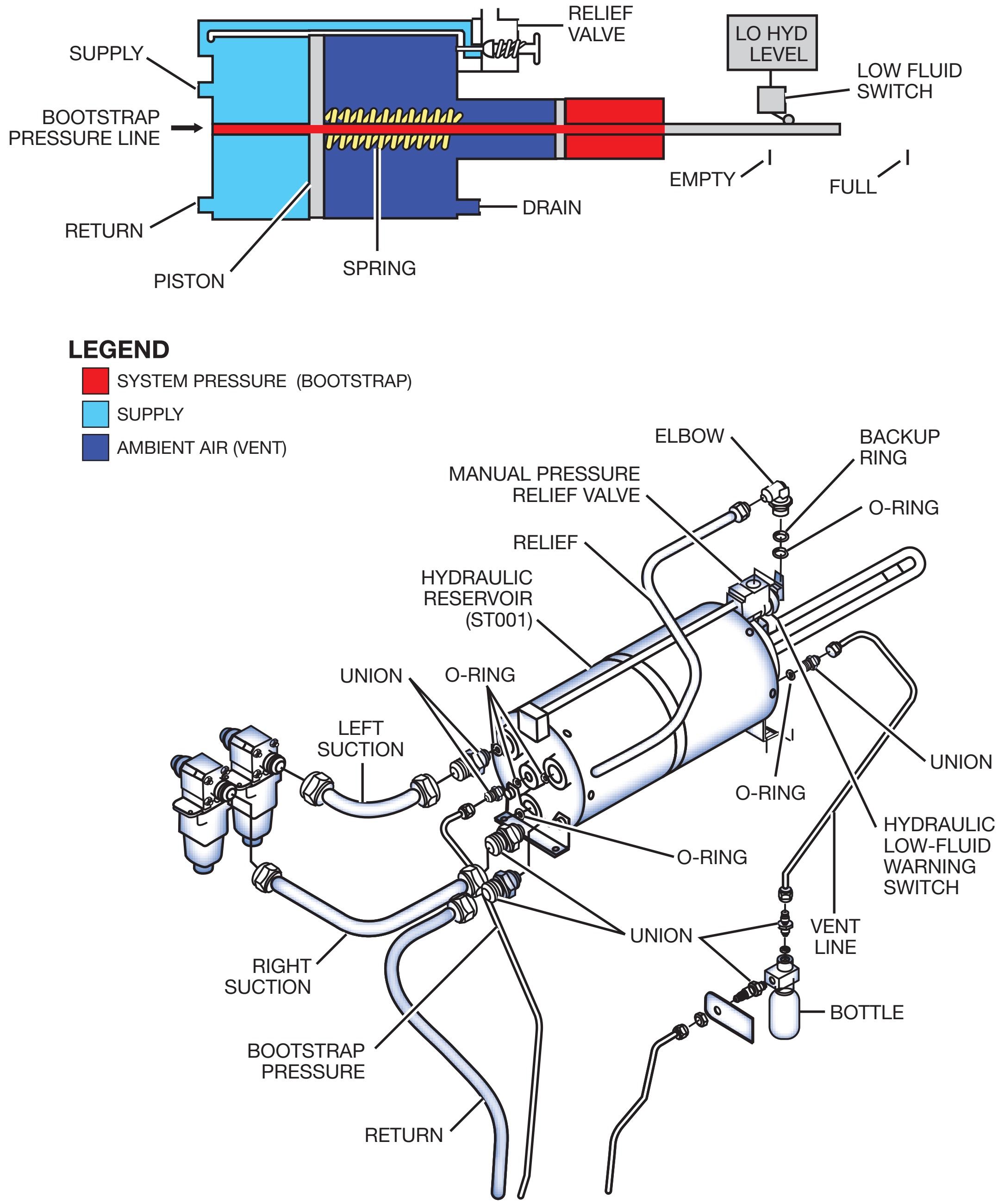


Figure 29-3. Hydraulic Reservoir

COMPONENTS

Hydraulic System Reservoir

Maintenance of the hydraulic reservoir (ST001) is mainly removal, installation, inspection, and replacing hydraulic low-fluid warning switch. If internal leakage or leaking of the reservoir case is apparent, the reservoir requires replacement. Special tools are required to successfully disassemble and assemble the reservoir. Observe phosphate ester base hydraulic fluid precautions during maintenance of the reservoir.

The reservoir is self-pressurizing with hydraulic system pressure up to 1,500 psi that pushes on a small diameter piston which is connected to a large diameter surface in the fluid reservoir (Figure 29-3). The area of the large surface is approximately 120 times the area of the small piston, to maintain 15 to 16 psi on the fluid in the reservoir. The large surface is also spring-loaded to maintain 2.7 to 4.0 psi on the fluid reservoir and the hydraulic system when the engine-driven pumps are not operating.

There is a pressure relief valve in the low-pressure area of the fluid reservoir. The valve starts to open at 40 psi and is fully open at 60 psi. When the fluid reservoir is filled to capacity of 360 cubic inches, the relief valve is opened mechanically to drain excess fluid. The relief valve may be operated manually to bleed off air and relieve pressure, prior to working on the hydraulic system. The entrapped air is the first to be expelled.

NOTES

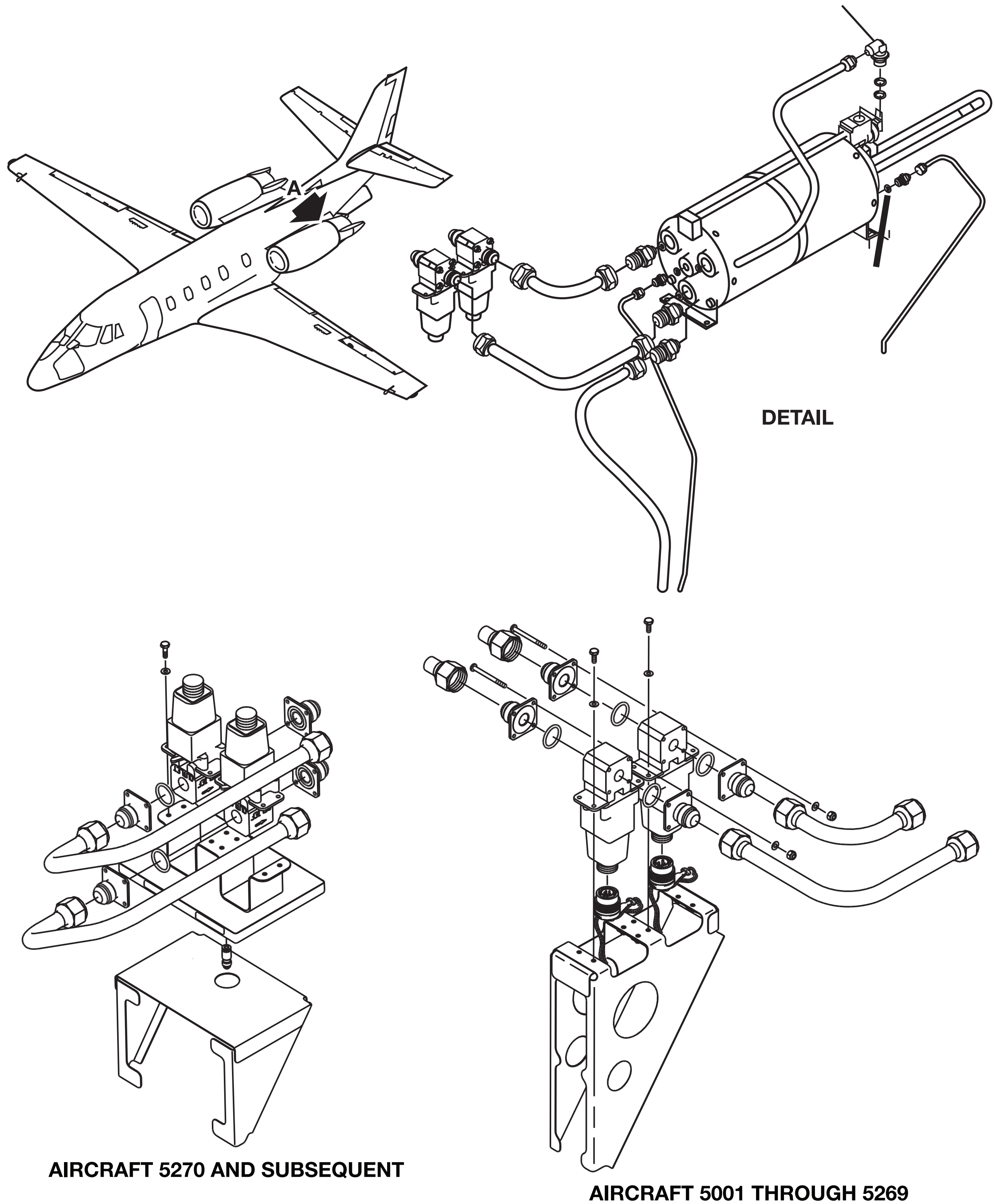


Figure 29-4. Hydraulic Firewall Shutoff Valves

Hydraulic Firewall Shutoff Valve

Description

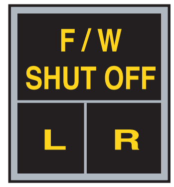
The hydraulic firewall shutoff valve is on the suction side of each engine-driven pump (right side of hydraulic reservoir and aft of the aft engine carry-thru) (Figure 29-4). The hydraulic firewall shutoff valves are operated by an electric motor. In the closed position, a thermal relief valve opens at 75 psi to relieve trapped fluid between the valve and pump.

The hydraulic firewall shutoff valves are controlled by the LH–RH ENGINE FIRE switchlights. When the hydraulic firewall shutoff valve is in the closed position (in conjunction with the fuel firewall shutoff valves—VY007 left and VY006 right), the L–R F/W SHUTOFF annunciator illuminates (XL/XLS) (Figure 29-5). On the XLS+, the white FIREWALL SHUTOFF L–R CAS message illuminates to indicate that both fuel and hydraulic firewall shutoff valves are closed on their respective sides. If one valve should open the message turns amber after 2 seconds (Figure 29-5).

A pointer on the valve assembly indicates the position of the valve.

Maintenance

Maintenance on the hydraulic firewall shutoff valves (VT032 left and VT035 right) consists of removal, installation and inspection. If a malfunction—such as failure to operate, or leaks in closed position—occurs, the assembly shall be returned to manufacturer for overhaul. Observe phosphate ester base hydraulic fluid precautions during maintenance on the suction shutoff valve.



L/R FW SHUTOFF
Flashes to indicate the respective fuel and hydraulic firewall shutoff valves have closed and the generator field relay has tripped. This annunciation occurs after the engine fire switchlight has been pressed. All three conditions are required for the light to illuminate.

XL/XLS ANNUNCIATOR

FIREWALL SHUTOFF L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	2 Second
White			Standard

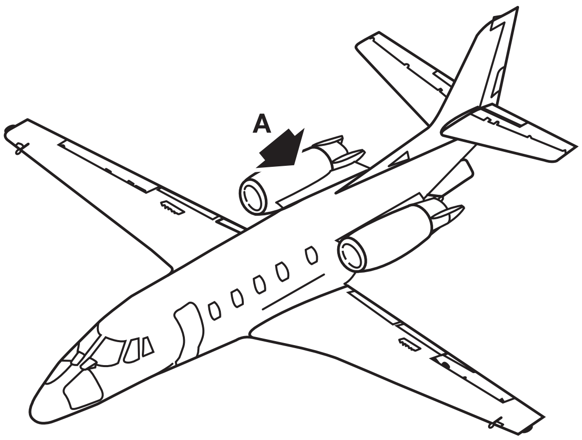
The advisory white message indicates normal operation while the amber message indicates abnormal operation. Normal operation for firewall shutoff is both fuel and hydraulic shutoff valves closed when the ENGINE FIRE switches are selected.

When both fuel and hydraulic shutoff's on one side become closed, the white message for the respective side will be displayed. If one valve should open the message will turn amber after 2 seconds. The 2 second delay allows for both valves to open when commanded without triggering an amber message.

When the firewall shutoffs are closed, a switch in the valve sends a 28 Volt signal to the EICAS system. When the valve is not closed, the switch sends an open signal to the EICAS system.

XLS+ CAS MESSAGE

Figure 29-5. Firewall Shutoff Indications



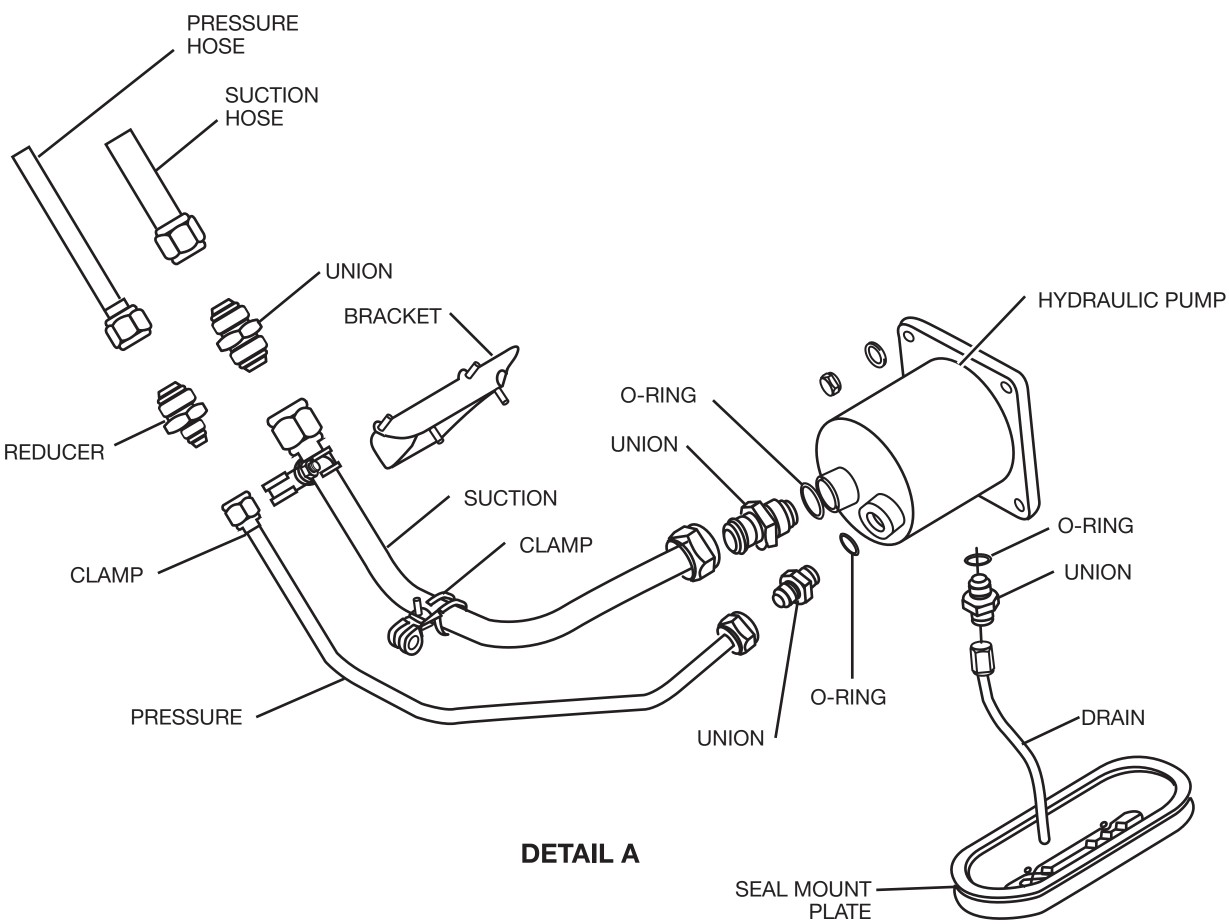


Figure 29-6. Hydraulic Pump

Hydraulic Pump

Description

The hydraulic pumps—one on each engine—are constant displacement-type pumps, driven by a splined shaft that is on the engine accessory gearbox (Figure 29-6). Butyl or silicone rubber impregnated White Night fire resistant sleeving is utilized around the hoses connecting the pump to the nacelle firewall fittings.

Maintenance

Maintenance of the hydraulic pump is removal, installation and inspection. The pump is designed to be able to operate for an indefinite period of time without actually pumping hydraulic fluid before it fails. This condition exists when an engine is shut down due to an engine fire or fire warning (firewall shutoff valve closed) and when the engine windmills after shutdown. Observe phosphate ester base hydraulic fluid precautions during maintenance on the hydraulic pumps.

NOTE

No lubrication is required on the hydraulic pump spline prior to pump installation onto the engine.

NOTES

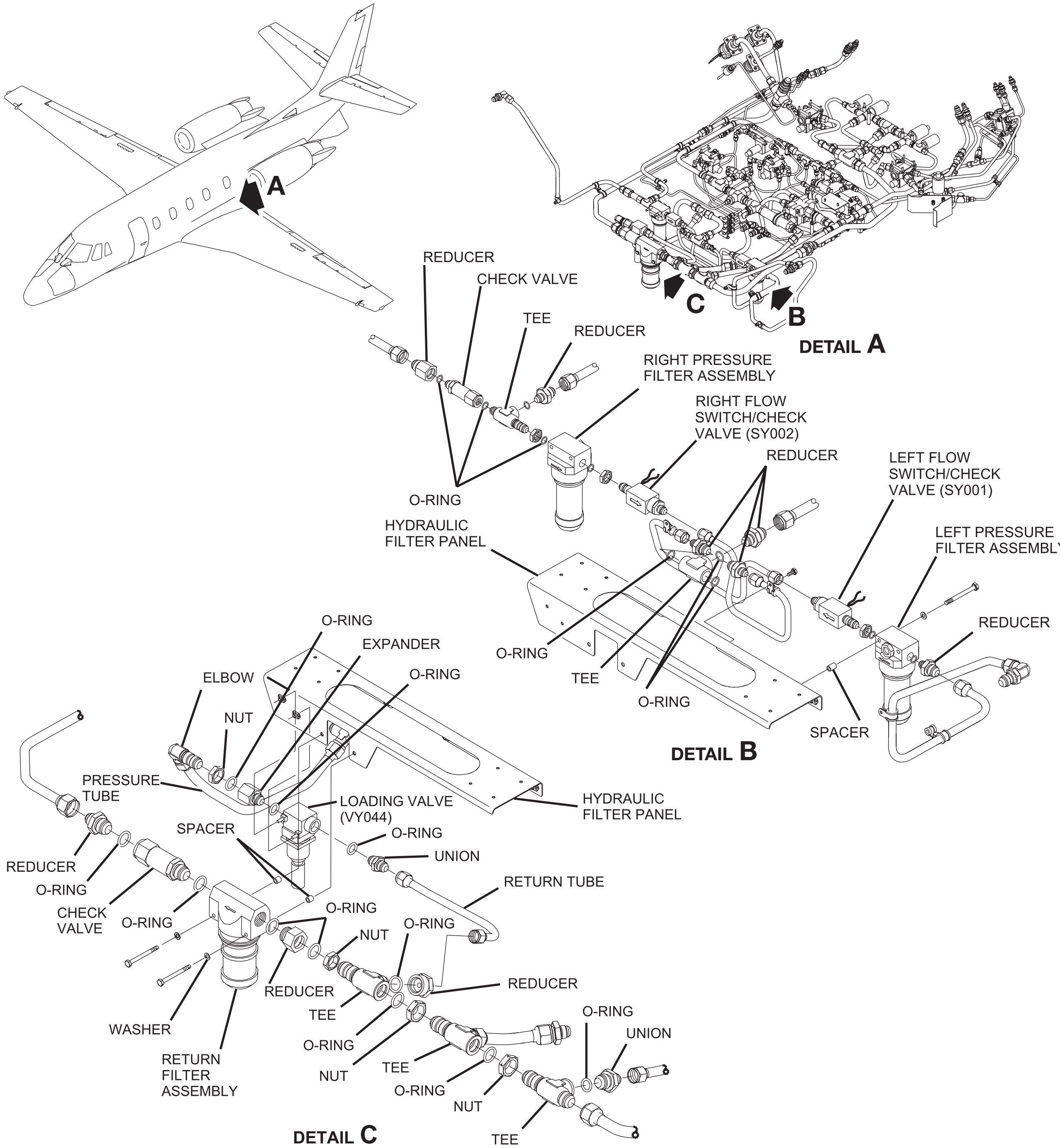


Figure 29-7. Hydraulic Filters

Hydraulic Filter Panel

The hydraulic filter panel provides support for:

- Pressure line filters
- Flow switch check valves
- Return line filter and the load valve (Figure 29-7)

Maintenance on the hydraulic filter panel components consists of removal, installation, inspection and filter element replacement. The filter assemblies and the flow switch check valves have a flow arrow cast in the body of the component that aid in installation. Observe phosphate ester base hydraulic fluid precautions during maintenance on the hydraulic filter panel components.

Filter Assemblies

A filter is in the pressure line from each engine-driven pump. These filters have a 3 gallons per minute (GPM) (11.4 liters per minute) nominal capacity, a 5 micron nominal rating and a 15 micron absolute rating. A bypass valve opens with a pressure differential of 100 psi. The filters use a disposable element.

A filter is installed in the return line leading to the fluid reservoir. This filter has a 12 GPM (45.4 liters per minute) capacity, a 5 micron nominal rating and a 15 micron absolute rating. A bypass valve opens with a pressure differential of 100 psi.

NOTE

XLS/XLS+ incorporate an additional return filter for the landing gear controls.

Open Center Load Valve

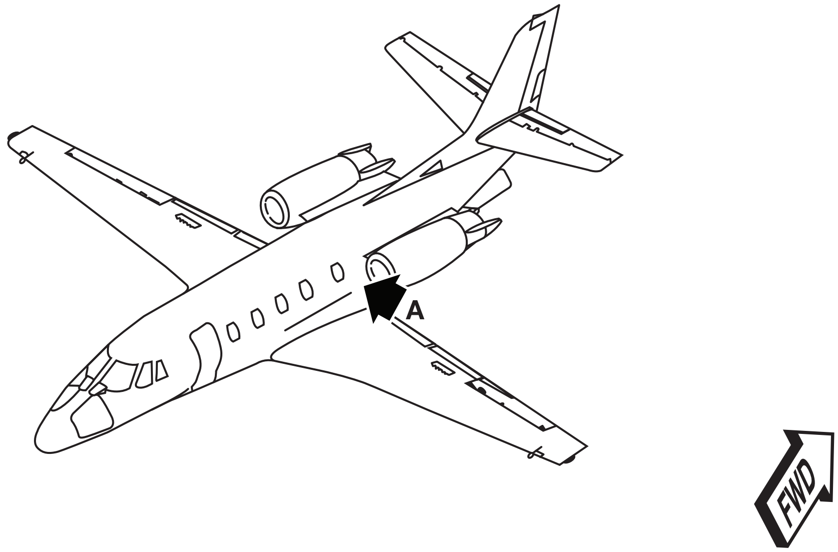
A solenoid operated open center load valve is on the center part of the hydraulic filter panel assembly. The load valve connects the hydraulic system pressure line to the system return line. It controls the open-center operation of the system. In a no-pressure-demand condition, the load valve is de-energized "open," allowing the fluid to free-flow from pressure to return. In a pressure demand condition, the load valve is energized "closed" and pressure is routed to a selected system/component.

Flow Switch Check Valves

The left and right flow switch check valves perform two functions: to prevent hydraulic flow from one engine-mounted pump to the other and to alert the flight crew when the left and/or right pump flow is low or no flow.

The check valve portion of the flow switch check valve is spring loaded "closed." The spring determines the pressure (flow) required to off-seat a poppet, allowing fluid to flow through the unit. A permanent-type magnet is attached to the poppet and moves with it. The electrical switch portion of the flow-switch check valve is a single-pole single-throw reed-type switch, secured in place with epoxy adhesive potting (not repairable). The left and right flow-switch check valve operation is the same.

As hydraulic fluid flow moves the poppet from the seated position, the attached magnet passes by the switch, opening its contacts. With the switch open, the respective L—R LO HYD FLOW annunciator extinguishes (XL/XLS) or amber HYDRAULIC FLOW LOW L–R CAS message (XLS+) (Figure 29-8). As the fluid flow decreases, the poppet moves toward the seated position and the magnet moves away from the switch: the switch closes. With the switch closed, the applicable L—R LO HYD FLOW annunciator illuminates.



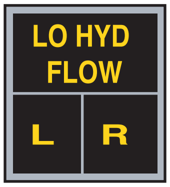
FILTER ASSY'S
2 PRESSURE
2 RETURN

MAIN MANIFOLD ASSY

L AND R T/R
MANIFOLDS

DETAIL A

Figure 29-9. Hydraulic Panel Components



L/R LOW HYDRAULIC FLOW
Annunciator illuminates steady to advise the crew that L and/or R engine-driven hydraulic pump flow rate is below normal. After five seconds it will begin flashing and illuminate MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

HYDRAULIC FLOW LOW L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	*5 Second
	*ESDI	SIPI	

This message is displayed when the hydraulic flow is low after engine start. The message has a 5 second debounce on, and a 3 second debounce off. On the output of each engine driven pump, there is a flow sensitive switch, which sends a ground to the EICAS system when the flow is low, which displays the message after 5 seconds. When the flow is normal, the switch provides an open signal, which removes the message after 3 seconds.

XLS+ CAS MESSAGE

Figure 29-8. Hydraulic Flow Indications

Flow switch check valve operation is as follows:

- On an increasing fluid flow of 1.33 gallons per minute (GPM) (503 liters per minute) maximum, the switch opens.
- On a decreasing flow of 0.35 to 0.55 GPM (1.32 to 2.08 liters per minute) minimum, the switch closes.

Hydraulic Panel Assembly

The panel provides a support for the relief valve and pressure switch. The panel also supports (Figure 29-9):

- Speedbrake control components
- Flap control components
- Landing gear control components.

The speedbrake, flaps and landing gear control components are described and maintained as outlined in their respective chapters.

The hydraulic panel assembly is a removable panel and may be removed as a unit without removing the components.

Observe phosphate ester base hydraulic fluid safety and technical precautions while performing maintenance on the hydraulic panel or it's components.

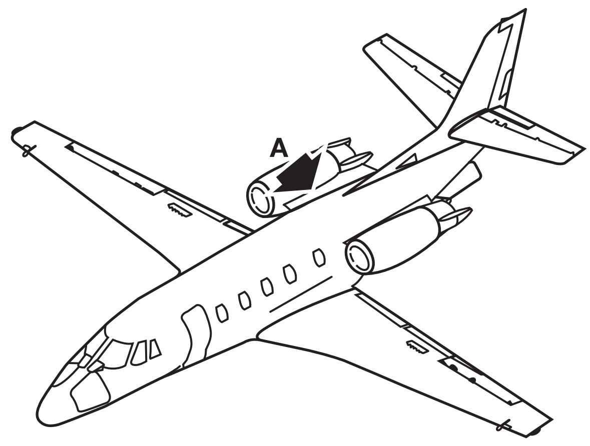
Hydraulic Panel Assembly

Components are secured to the panel with screws (in matching nutplates) on the panel. Clamps with bolts and washers are used to secure necessary lines, fittings and components that do not incorporate mounting provisions to the panel.

Relief Valve

The relief valve is on the left forward portion of the hydraulic panel assembly and secured to the panel with a clamp. The relief valve cracks open at 1,350 psi and is fully open at 1,500 psi. The relief valve is incorporated into the hydraulic manifold on the XLS/XLS+.

NOTES



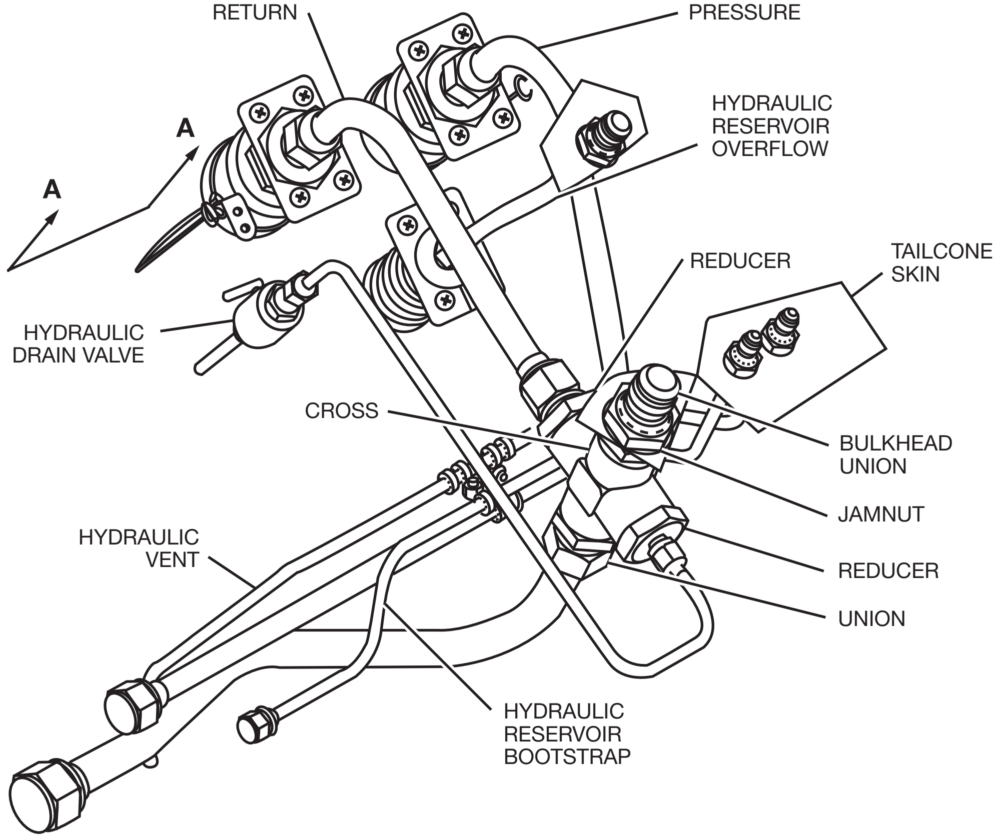


Figure 29-10. Ground Service Connections

GROUND POWER CONNECTION

DESCRIPTION

The ground power connectors for the hydraulic system are on the lower right side of the airplane at FS 424.50 (Figure 29-10). A fiberglass door with drain tube covers the quick-disconnects when not in use.

The panel includes:

- Pressurization connection for the hydraulic system
- Return connection from the hydraulic system
- Return connection from the hydraulic reservoir relief valve
- Drain valve for the hydraulic reservoir
- Vent line for the hydraulic reservoir

CAUTION

Ensure that the airplane ground suction source quick-connect fitting is securely connected to the service cart return line. Failure to do so could cause damage to the reservoir by over pressurizing it.

CONTROLS AND INDICATIONS

Hydraulic Pressure Indicating System

The purpose of the hydraulic pressure indicating system is to inform the flight or maintenance crew that the hydraulic system is pressurized during:

- Landing gear actuation
- Flap actuation
- Speed brake actuation
- Thrust reverser operation

When the hydraulic system is in a no pressure condition (load valve open to return) the pressure indicating switch (SY032) is "open." The HYD PRESS annunciator (XL/XLS) or white HYDRAULIC PRESSURE CAS message (XLS+) extinguishes (Figure 29-11). When the landing gear, flaps, speedbrake or thrust reverser is actuated, the load valve also closes, and pressure is built-up to operate the selected system. As the pressure increases toward 1,500 psi (maximum system pressure), the pressure switch closes at 185 psi maximum and completes the electrical circuit to illuminate the HYD PRESS annunciator (XL/XLS) or white HYDRAULIC PRESSURE CAS message (XLS+). After the selected hydraulic system completes actuation, the load valve opens—bypassing pressure to return. As the pressure decreases, the pressure switch opens at 155 ± 5 psi minimum, extinguishing the HYD PRESS annunciator. If the hydraulic system remains

HYDRAULIC PRESSURE
ON GROUND—Annunciator illuminates steady with no illumination of master caution to indicate the hydraulic system is pressurized.
IN FLIGHT—Annunciator illuminates steady with no illumination of master caution to indicate the hydraulic system is pressurized. If still on after 40 seconds, annunciator begins to flash and activates MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

HYDRAULIC PRESSURE

Color	Inhibited By		Debounce
Amber	*LOPI	*TOPI	*40 Second
White			Standard

This message is displayed when hydraulic pressure is in the hydraulic system. The message changes to amber if there is pressure for more than 40 seconds in the air. There is a hydraulic pressure switch which provides a ground to the EICAS system when the pressure is above 185 PSI, which displays the message. When the pressure drops below 155 PSI, the switch opens and the message is removed.

* The white message does not have TOPI or LOPI, the amber message has TOPI and LOPI.

XLS+ CAS MESSAGE

Figure 29-11. Hydraulic Pressure Indications

pressurized for more than 40 seconds, the HYD PRESS annunciator (XL/XLS) begins to flash or the white HYDRAULIC PRESSURE CAS message (XLS+) turns amber and causes the MASTER CAUTION switchlight to illuminate.

DIAGNOSTICS

Hydraulic System reservoir Fluid Level

The system hydraulic-reservoir (ST001) has a visual indicator at one end to indicate quantity of fluid. The indicator scale is visible from the right side of the indicator. The scale is marked, identified EMPTY, LOW, REFILL, FULL and OVERFULL. The piston extension top is painted red for improved visibility of the fluid level scale. A remote warning system, consisting of an electrical switch attached to the reservoir assembly and an annunciator, alerts the flight crew when the fluid level is low.

The reservoir is serviced utilizing either an external hydraulic service unit or a hand pump. The hydraulic reservoir fluid capacity is measured by volume of fluid. A visual indicator on the end of the reservoir is scaled at EMPTY (5 cubic inches—82 ml.), LOW (74 cubic inches—1213 ml.), REFILL (175 cubic inches—2868 ml.), FULL (215 cubic inches—3523 ml.), and OVERFULL (360 cubic inches—5899 ml.).

The low-fluid warning switch alerts the flight or maintenance crew when the volume of hydraulic fluid in the reservoir (ST001) is at approximately 74 cubic inches (1213 ml.). The hydraulic reservoir visual indicator is between the EMPTY mark and the REFILL mark. The warning switch actuator rides on the visual indicator rod and is held “open,” breaking the electrical circuit to the LO HYD LEVEL annunciator (XL/XLS) or amber HYDRAULIC FLUID LEVEL LOW CAS message (XLS+) (Figure 29-12). If the fluid volume in the reservoir reduces to approximately 74 cubic inches (1213 ml.), the visual indicator rod passes by the switch actuator, allowing the switch to close. This action completes the electrical circuit to the LO HYD LEVEL annunciator (XL/XLS) or amber HYDRAULIC FLUID LEVEL LOW CAS message (XLS). When the reservoir is replenished with fluid by servicing, the annunciator extinguishes.

LOW HYD LEVEL
Annunciator flashes to indicate that the hydraulic reservoir level is low (fluid quantity is 74 cu. in. or below). Activates MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

HYDRAULIC FLUID LEVEL LOW			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the hydraulic fluid level in the reservoir is low. There is a mechanical switch on the reservoir which provides a ground signal to the EICAS when the fluid level is low. When the EICAS receives the ground, it posts the message. When the fluid level is normal, an open is sent to EICAS, which removes the message.			

XLS+ CAS MESSAGE

Figure 29-12. Hydraulic Fluid Level Indications

Recommended External Leakage Limits

Dynamic Seals

Dynamic seals are those which contact sliding or rotating parts such as actuator shaft seals, control valve shaft seals, etc.

Actuate the component through several full travel cycles to exercise the seal prior to performing the check. This is particularly important during extremely cold weather since seal resilience and, therefore, seal capability are reduced under such conditions. Also, sufficient actuation to warm up the system fluid is often beneficial in cold weather.

The following recommended limits apply with the unit under full or partial system pressure.

- After overhaul limit: One drop in five minutes, maximum.
- In-service limit: One drop per minute or one drop in twenty five complete cycles, maximum.

Static Seals

Static seals are those at parting surfaces, boss seals under tube fittings, static gland seals, etc.

The following recommended limits apply with the unit under full or partial pressure:

- Seepage, causing no perceptible dripping, is acceptable.
- Dripping leaks from accessible static seals are cause for seal replacement.
- Dripping leaks from inaccessible seals that cannot be reduced to one drop in ten minutes are cause for unit removal.

Hydraulic Reservoir External Relief Valve

The following recommended limits apply with the unit under full or partial system pressure.

- Static external relief valve leakage shall be zero.
- Dynamic external relief valve leakage shall not exceed one drop per 50 cycles of the relief valve poppet, with 0 to 100 psi across the seal.

Maintenance Practices

Phosphate Ester Safety Precautions

CAUTION

Observe the following safety precautions when working on systems containing phosphate ester-based fluid. Long exposure to phosphate ester-based fluids can cause skin dehydration and chapping.

- Wash hands thoroughly with soap and water before starting work.
- Apply panthoderm cream or equivalent (silicone hand cream) to hands, wrists, and forearms at the beginning of the work period. Rub cream under the fingernails and into the creases of the skin.
- Apply kerodex or equivalent frequently during the work period. Reapply the panthoderm cream only after the skin has been cleansed by washing.
- Wear goggles when pressure-testing components or systems and any time there is possibility of fluid splashing into the eyes.
- If fluid splashes into the eyes, treat eyes immediately by irrigating thoroughly with clear, cold, water.
- Wash hands, wrists, and forearms with soap and hot water whenever they have been in contact with fluid.
- If clothing becomes soaked with fluid, remove it as soon as possible; thoroughly wash skin, and put on clean clothing.

NOTES

Technical Precautions

CAUTION

Observe the following technical precautions when working on the hydraulic systems. Phosphate ester based fluids adversely affects a wide range of materials, including rubber, copper, various plastics, and paints.

Ensure that the fluid does not come into contact with any part of the airplane outside of the hydraulic system. Keep spillage to an absolute minimum. Place rags under fittings before disconnecting lines. Clean up spilled hydraulic fluid immediately to prevent entry into adjacent areas of the airplane and to prevent future false hydraulic leak reports.

- When lines are disconnected and/or components are removed, provide suitable protection by use of caps or covers to prevent foreign material from entering the lines or components.
- When electrical connectors are disconnected, install caps or other suitable protectors to prevent entry of hydraulic fluid, moisture, and foreign objects.
- Always check position and angle of all fittings removed from components to ensure placement and alignment on installation or replacement components.
- When washing metal parts before assembly, use only naphtha, Federal Specification P-D-680 (Type 1) or a high flash stoddard solvent, and ensure that all traces of the solvent are removed before assembly.
- Use only clean phosphate ester-based fluid for flushing or testing hydraulic components.
- Use only clean phosphate ester-based fluid when filling the reservoir.

QUESTIONS

1. If DC power is lost to the hydraulic system, the loading valve:

A. Fails to the closed position
B. Is not affected
C. Fails to the open position
D. None of the above

2. The hydraulic system provides pressure to operate:

A. Landing gear and speedbrakes only
B. Antiskid brakes, landing gear, and flaps
C. Speedbrakes, landing gear, thrust reversers, horizontal stabilizer, and flaps
D. Speedbrakes, landing gear, and wheel brakes

3. Low reservoir fluid level is indicated by illumination of the:

A. LO HYD LEVEL annunciator
B. HYD PRESS annunciator
C. L/R LO HYD LEVEL annunciator
D. L/R LO HYD FLOW annunciator

4. Hydraulic system operation is indicated by illumination of:

A. LO HYD LEVEL annunciator
B. HYD PRESS annunciator
C. L/R LO HYD LEVEL annunciator
D. L/R LO HYD FLOW annunciator

5. The correct statement concerning the hydraulic system is:

A. The HYD PRESS annunciator illuminates anytime an engine-driven pump is operating.
B. The HYD PRESS annunciator illuminating while the gear is extending may indicate a failed hydraulic pump.
C. The LO HYD FLOW L/R annunciator illuminates whenever reservoir fluid level is low.
D. A L or R LO HYD FLOW annunciator may indicate a failed hydraulic pump.

6. The white HYDRAULIC PRESSURE CAS message is normal anytime a hydraulic system is in operation. If this light begins to flash, it indicates:

A. Hydraulic system has been pressurized for more than 40 seconds
B. Hydraulic pumps are overheating
C. Hydraulic system has failed
D. Landing gear must be lowered by the emergency system

7. Illumination of the HYD PRESS light indicates:

A. Hydraulic load valve has energized closed
B. Fluid is circulating between the hydraulic pumps and the reservoir at approximately 60 psi
C. Hydraulic pressure is available to the aircraft brake system
D. Hydraulic reservoir is pressurized at 2.7 to 4.0 psi

8. The hydraulic firewall shutoff valve is:

A. Energized closed, deenergized open
B. Energized open, deenergized closed
C. Open and closed electrically, when the guarded, red fire switch is pushed
D. Automatically closes when the HYD FLOW low light illuminates to prevent cavitation of the hydraulic pump

9. Leakage around hydraulic fittings and nuts that have static seals can be:

A. Corrected by lightly tapping on the affected part
B. Corrected by increasing the torque to 50% above specified limit
C. corrected by applying correct torque and/or replacing the seal
D. Acceptable as long as the leak does not exceed one drop per minute or one drop in five complete cycles

10. Before adding fluid to the hydraulic reservoir, verify that the:

A. Speedbrakes and flaps are retracted
B. Landing gear is extended
C. Thrust reversers are stowed
D. All of the above

11. The hydraulic fluid used to service the 560XL aircraft is:

A. Mineral based
B. Phosphate ester based
C. Compatible to Mil-H-5606
D. Designed to be used as a leak detector in the aircraft fuel system

12. When servicing the hydraulic system, after installing a new hydraulic load valve, you should:

A. Not operate the hydraulic system because the system is self bleeding
B. Cycle fluid through the system 2 to 5 minutes to bleed air from the system
C. Pump the brake pedals 12 times with the battery switch in BATT
D. Inspect and clean the hydraulic return and pressure filters

CHAPTER 30
ICE AND RAIN PROTECTION

CONTENTS

ILLUSTRATIONS

CHAPTER 30
ICE AND RAIN PROTECTION

INTRODUCTION

This chapter presents the ice and rain protection systems found in the Citation 560XL/XLS/XLS+ aircraft, and has been divided into seven sections. These sections are engine anti-ice, wing anti-ice, tail deice, windshield rain removal, windshield anti-ice, pitot/static anti-ice, and the heated drains. General maintenance considerations are included in each section along with a description of components and their operation. References for this chapter and further specific information can be found in Chapters 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," Chapter 30—"Ice and Rain Protection," and Chapter 36—"Pneumatics," of the *Aircraft Maintenance Manual (AMM)*.

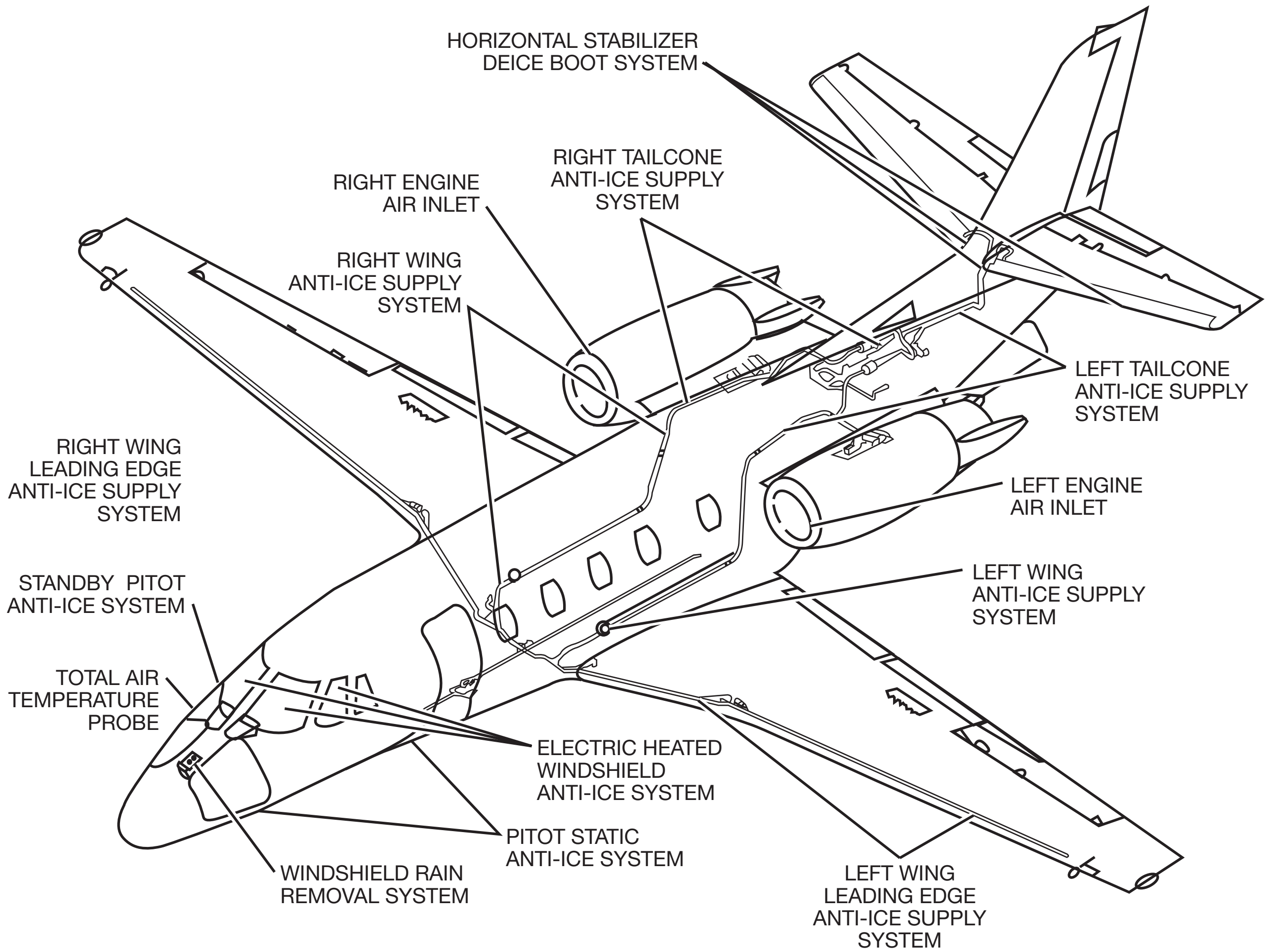


Figure 30-1. Ice and Rain Protection Systems

GENERAL

This chapter describes the systems and components which prevent or dislodge ice formation on various exterior areas of the aircraft. Preventing ice formation is identified herein as anti-ice and dislodging ice formation is identified as deice.

Areas protected from the formation of ice by anti-ice systems are:

- Inboard/outboard wing leading edge
- Engine air intake nacelles
- Pitot/static ports
- AOA vane
- Overboard water drain lines

These areas have anti-ice systems which either heat the area with hot engine bleed air or electrical heating elements (Figures 30-1 and 30-2).

The horizontal stabilizer is protected by pneumatic boots which periodically inflate to dislodge or break up accumulated ice.

The windshield anti-ice system includes electrically heated glass windshields and forward side windows, combined with a forced air windshield moisture/rain removal system.

The pitot static anti-ice systems are comprised of electrically heated pitot tubes and electrically heated static ports.

Ice is detected by visual verification of ice being present. The wing inspection lights and the ice detection lights facilitate in verifying ice is present.

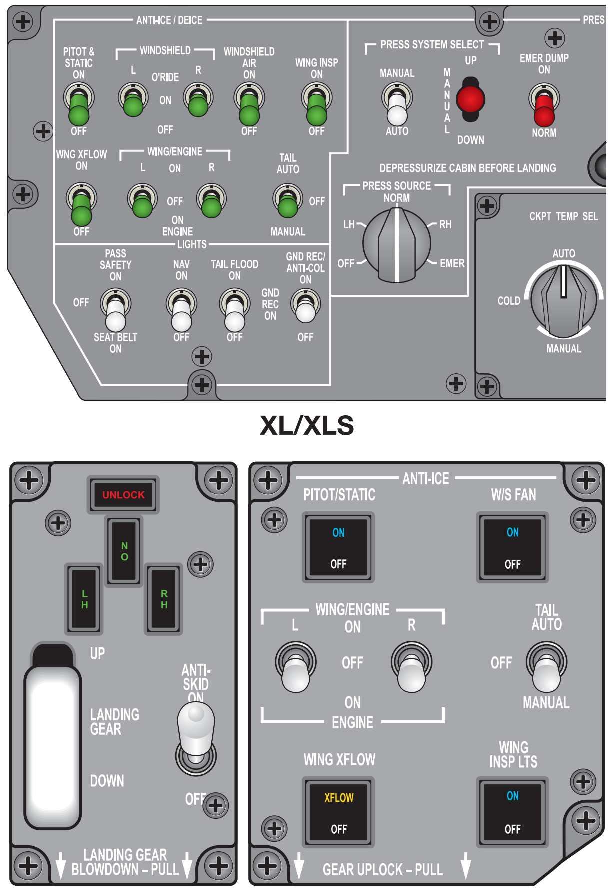


Figure 30-2. ANTI-ICE Switch Panels

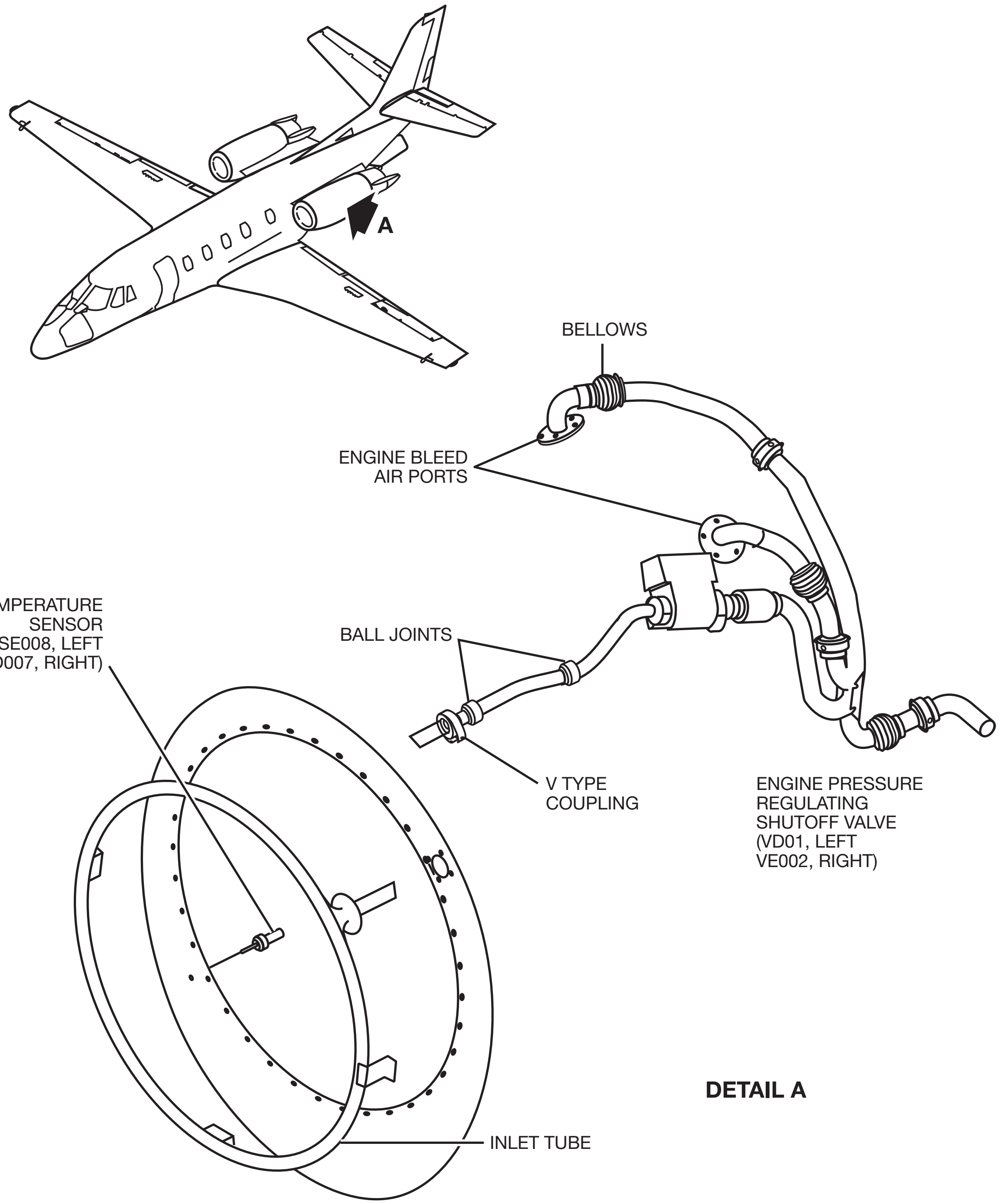


Figure 30-3. Engine Air Inlet Anti-ice Components

ENGINE AIR INLET ANTI-ICE

DESCRIPTION

The bleed air for anti-icing the engine inlets is taken from the engine being anti-iced. There are no provisions for cross feeding the two systems. In the event of an engine failure, the failed engine is no longer anti-iced. Engine anti-ice also includes continuous ignition to prevent engine flame out, and stator anti-icing. Bleed air is extracted from the engines and supplied directly to the engine inlet pressure regulating shutoff valves. The temperature of the air supplied to the engine inlets is controlled only by throttle settings. The engine-inlet bleed air is then routed directly to the engine inlet assemblies. After passing through the inlet assemblies, the air passes over the under temperature switches. In the event that the engine anti-ice systems are on and an engine inlet has cooled below a safe level, the under temperature switches annunciates the cold condition on the anti-ice panel.

The engine inlet assembly contains a forward bulkhead that creates a plenum behind the forward surface of each engine (Figure 30-3). Inside this plenum, there is a circular piccolo tube that fits just behind the forward surface of the inlet. The bleed air enters the piccolo tube at the top of the engine, impinges on the forward surface, then travels aft in the plenum, and then exhausts outside the engine inlets.

Stainless steel tubing is used to transfer bleed air from the engine to the air inlet duct anti-ice system. The engine stator anti-ice system is part of the engine installation, except for the electrical connection which powers a control valve.

There are engine inlet pressure regulating shutoff valves on the engines. The valves are poppet type valves constructed of stainless steel. The two three-position WING/ENG ANTI-ICE switches activate the pressure regulating shutoff valves. They are electrically actuated, but pneumatically powered.

NOTES

L/R ENG ANTI-ICE
Steady illumination indicates the system is warming up. Flashing illumination indicates the system has not warmed up properly. A 4-minute and 45-second warm-up period is required before the light begins flashing. If the system warms up but later becomes inoperative, the annunciator flashes immediately. Causes for a flashing light include the loss of stator vane heat or the engine nacelle is too cold. This annunciator also flashes if engine anti-ice is selected OFF and the stator vane heating valve does not close. The engine anti-ice monitoring sensors are enabled when wing anti-ice is selected ON.

If the annunciator is flashing, the MASTER CAUTION lights will activate.

XL/XLS ANNUNCIATOR

ENGINE ANTI-ICE COLD L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	5 Seconds
White	ESDI	SIPI	1 Second

In air operation - the white message is displayed when anti-ice is selected on, and the surface is not warmed up yet. If, after 285 seconds of cold, the white message becomes amber. The amber message also can come up if the surface has warmed up and then cooled off again. Once the amber message is shown, it remains for 5 seconds after the condition is removed.

On ground operation - the white message is displayed when anti-ice is selected on, until the surface becomes warm, then it goes out. There is no 285 second timer on the ground. The amber message also can come up if the surface has warmed up and then cooled off again.

The amber message can also be displayed, on ground or in air, if the fan/stator anti-ice valve is not in the correct position for more than 5 seconds.

ANTI-ICE on is: respective engine side anti-ice selected on or engine/wing anti-ice turned on. For I/O definition of engine/wing anti-ice, see WING ANTI-ICE COLD L-R.

Amber message logic is the following with a 5 second debounce on and off:

- ANTI-ICE on **AND**
- In air **AND**
- NOT engine shutdown **AND**
- Surface cold more than 285 seconds

OR

- ANTI-ICE on **AND**
- Surface cold **AND**
- NOT engine shutdown **AND**
- The surface was warm at least once since being selected on

OR

- NOT engine shutdown **AND**
- Engine fan/stator anti-ice valve is not in correct position

White message logic is the following for more than 1 second:

- ANTI-ICE on **AND**
- NOT amber message **AND**
- Surface cold
- NOT engine shutdown **AND**
- In air **AND**

OR

- ANTI-ICE on **AND**
- NOT amber message **AND**
- The surface was cold when selected on **AND**
- The surface has remained cold since selecting on
- NOT engine shutdown **AND**
- On ground **AND**

Engine cold is ground for cold, open for warm. Eng A/I On is ground for engine anti-ice selected on, open for off. F/S Valve Clsd is ground for valve closed, open for valve open. The valve is open to provide anti-icing to the fan and stator.

XLS+ CAS MESSAGE

Figure 30-4. Engine Anti-Ice Indications

The engine inlet under temperature switches are in the engine inlet forward bulkhead, extending into the plenum. These switches monitor the temperature of the air in the plenum. When the temperature in the plenum is less than 60°F, an undertemperature condition is indicated by the illumination of the L–R ENG ANTI-ICE annunciator (XL/XLS) or ENGINE ANTI-ICE COLD L–R CAS message (XLS+) (Figure 30-4). The L–R ENG ANTI-ICE annunciator also illuminates if the stator bleed-air solenoid valve fails to open when engine anti-ice is selected.

OPERATION

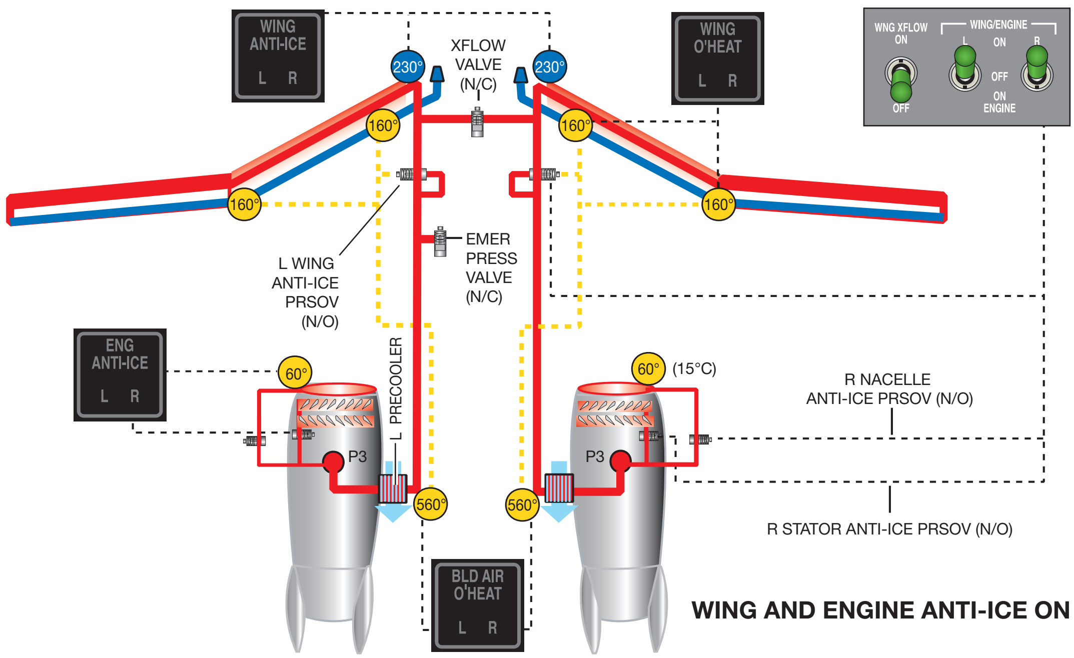
In the absence of electrical power the pressure regulating shutoff valves are driven to the "open" position by upstream pressure. When electrical power is applied, the upstream pressure is used to shut the valve. The pressure regulating shutoff valves control the airflow pressure downstream of the valve to 16 psig, ± 3 psig. The pressure regulating valves (in combination with the airflow restriction of the inlet assembly) effectively regulate the air flow of the engine inlet anti-ice system.

When either of the WING/ENGINE ANTI-ICE switches are in the ENGINE or WING/ENGINE ON position, electrical power is removed from the engine inlet pressure. This regulates shutoff valves, allowing bleed air to flow to the engine inlet anti-ice assemblies. With both switches in the OFF position, electrical power is applied to the inlet pressure regulating the shutoff valves to shut off bleed air flow to the engine inlet assemblies. In addition, in the OFF position, the under temperature warning system is disabled.

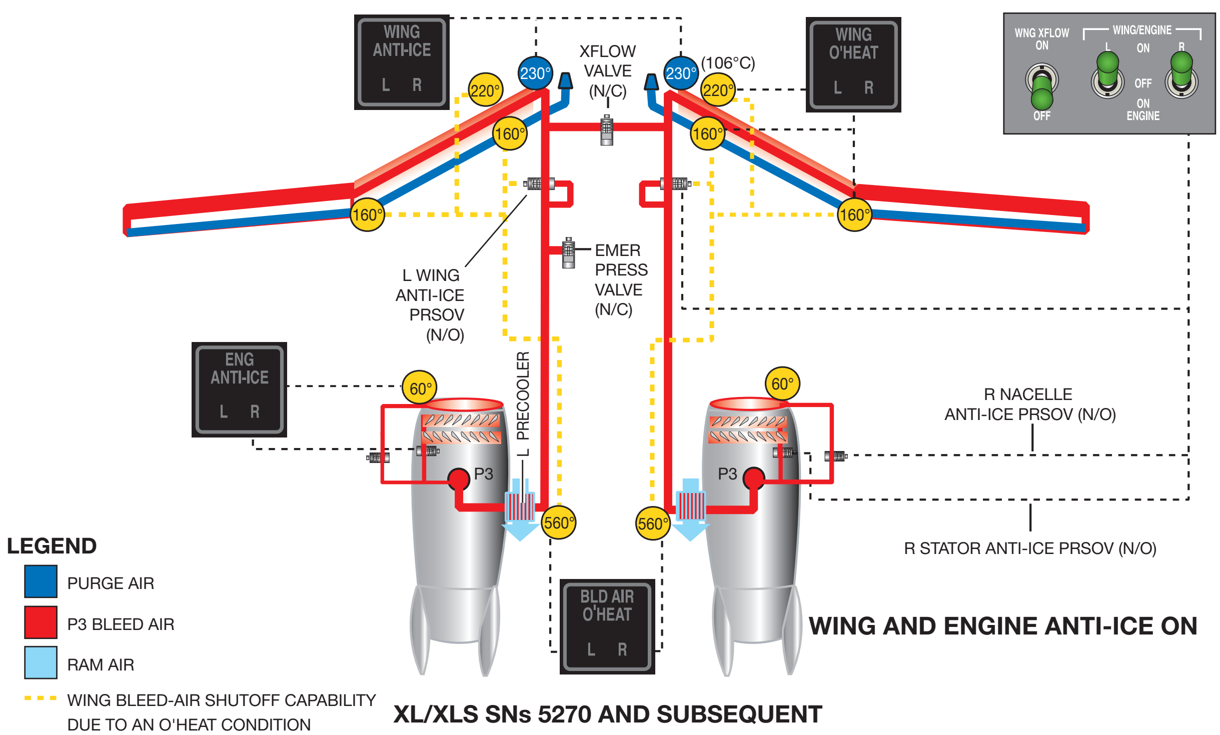
NOTE

Allow time for inlet temperature sensor to heat after turning on system. L–R ENG ANTI-ICE then extinguishes.

NOTES



XL SNs 5001 THROUGH 5269



XL/XLS SNs 5270 AND SUBSEQUENT

Figure 30-5. Wing/Engine Anti-Ice Schematic

WING LEADING EDGE BLEED AIR ANTI-ICE

DESCRIPTION

A bleed-air heated-wing anti-ice panel assembly is on the left and right wing leading edges to prevent ice build-up forward of the engines. Systems are typical for both wings.

Bleed air is extracted from the engines and used as a source of heat to keep wing leading edges and engine inlets clear of ice (Figure 30-5). Hot bleed air is sprayed onto the inside surface of both wing leading edges and engine inlets to maintain the temperature of surfaces above freezing while in flight. If surfaces cool below a safe level during flight, the condition is indicated by illumination of the L–R WING ANTI-ICE annunciator (XL/XLS) or amber WING ANTI-ICE COLD L–R CAS message (XLS+) (Figure 30-6). Throttle settings control the temperature of the air supplied to the engine inlets.

The wing leading edge assembly includes an aluminum outer skin and diffuser that is screwed to front spar of each wing. Inside the leading edge outer skin there is a heatshield covered with neoprene coated cloth. There is a piccolo tube between the outer skin and diffuser secured with a clamp at WS 101.07. The inner liner assembly and outer skin are bonded together to form a single wing leading edge assembly. Two scoops on the lower surface of the wing near the outboard aft edge allow bleed air to vent overboard.

Wing overtemperature switches, inboard wing/fuselage overheat and inboard wing overheat switch, are set to close when the temperature in the wing leading edge cavity exceeds 160°F (71°C). A closed wing overheat switch causes the L–R WING O'HEAT annunciator (XL/XLS) or amber WING ANTI-ICE OVERTEMP L–R CAS message to illuminate and the respective wing pressure regulating shutoff valve closes, shutting off bleed air to the overheated wing panel (Figure 30-7).

L/R WING ANTI-ICE
Steady illumination, ground or inflight, indicates that wing anti-ice has been selected ON and the surface is warming up.
Flashing illumination indicates the surface is too cold. A 4-minute and 45-second warm-up period is required before the light begins flashing. If the surface reaches operating temperature, but later becomes too cold, the light flashes immediately. The undertemperature sensors are enabled when wing anti-ice is selected ON.

XL/XLS ANNUNCIATOR

WING ANTI-ICE COLD L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
White			

ANTI-ICE on is: X-flow selected on OR respective side wing anti-ice selected on.

Amber message logic is:
- ANTI-ICE on **AND**
- In air **AND**
- Bleed air cold more than 285 seconds

OR
- ANTI-ICE on **AND**
- Bleed air cold **AND**
- The surface was warm at least once since being selected on

OR either of the above was true in the last 5 seconds.

XLS+ CAS MESSAGE

Figure 30-6. Wing Anti-Ice Indications

The XLS/XLS+ has an additional 220°F overheat switch in each wing inboard leading edge, which causes the L–R WING O'HEAT annunciator (XL/XLS) or amber WING ANTI-ICE OVERTEMP L–R CAS message (XLS+) to illuminate and closes the respective left or right wing pressure regulating shut off valve.

The undertemperature switches monitor the temperature of bleed air entering the wing leading edge anti-ice panel assemblies. When bleed air passing over the under temperature switch is less than 230°F (110°C), the switch closes and the appropriate L–R WING ANTI-ICE annunciator (XL/XLS) or white WING ANTI-ICE COLD L–R CAS message (XLS+) illuminates to indicate the condition (Figure 30-8).

L/R WING O'HEAT
Annunciator flashes to indicate a bleed-air leak into the wing purge air passage. The affected side wing anti-ice automatically shuts off. If wing anti-ice is in use, it reactivates when the leading edge cools (cycle ON and OFF). Wing overheat sensors are active with or without the anti-ice switches ON.

XL/XLS ANNUNCIATOR

WING ANTI-ICE OVERTEMP L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard

There are three over temperature switches in each wing for a total of six switches. The switches are behind the heat shield on the forward wing spar.

When the temperature is over 160°F at either switch, the switch sends a ground signal to the EICAS, which posts the message for the respective side.

There is also a temperature switch inside the fuselage at the wing root on both sides which trips at 220°F. All three overtemp switches per side are wired in parallel for a total of two inputs to EICAS. When the temperature is normal at all three switches, the respective EICAS input is open and the message is removed.

XLS+ CAS MESSAGE

Figure 30-7. Wing Anti-Ice Overtemp Indications

COMPONENTS

The major components of the wing anti-ice system are as follows (Figure 30-9):

- Pylon precooler overtemperature switches
- Precooler controller/actuator temperature sensors
- Pressure regulating shutoff valves
- Crossfeed valve
- Undertemperature switches
- Forward wing spar overheat switches

WING ANTI-ICE COLD L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
White			

In air operation - the white message displayed when wing anit-ice or crossflow is selected on, and the surface is not warmed up yet. If, after 285 seconds of cold, the white message becomes amber. The amber message also can come up if the surface has warmed up and then cooled off again. Once the amber message is shown, it remains for 5 seconds after the condition is removed.

One ground operation - th white message is displayed when wing anti-ice or crossflow is selected on, until the surface becomes warm, then it goes out. There is no 285 second timer on the ground. The amber message also can come up if the surface has warmed up and then cooled of again.

White message logic is:
- ANTI-ICE on **AND**
- NOT amber message **AND**
- in air **AND**
- Surface cold

OR
- ANTI-ICE on **AND**
- NOT amber message **AND**
- On ground **AND**
- The surface was cold when selected on **AND**
- The surface had remained cold since selected on

Figure 30-8. Wing Anti-Ice Cold Indication

- Wing leading edge anti-ice panel assemblies
- Instrument control panel

NOTES

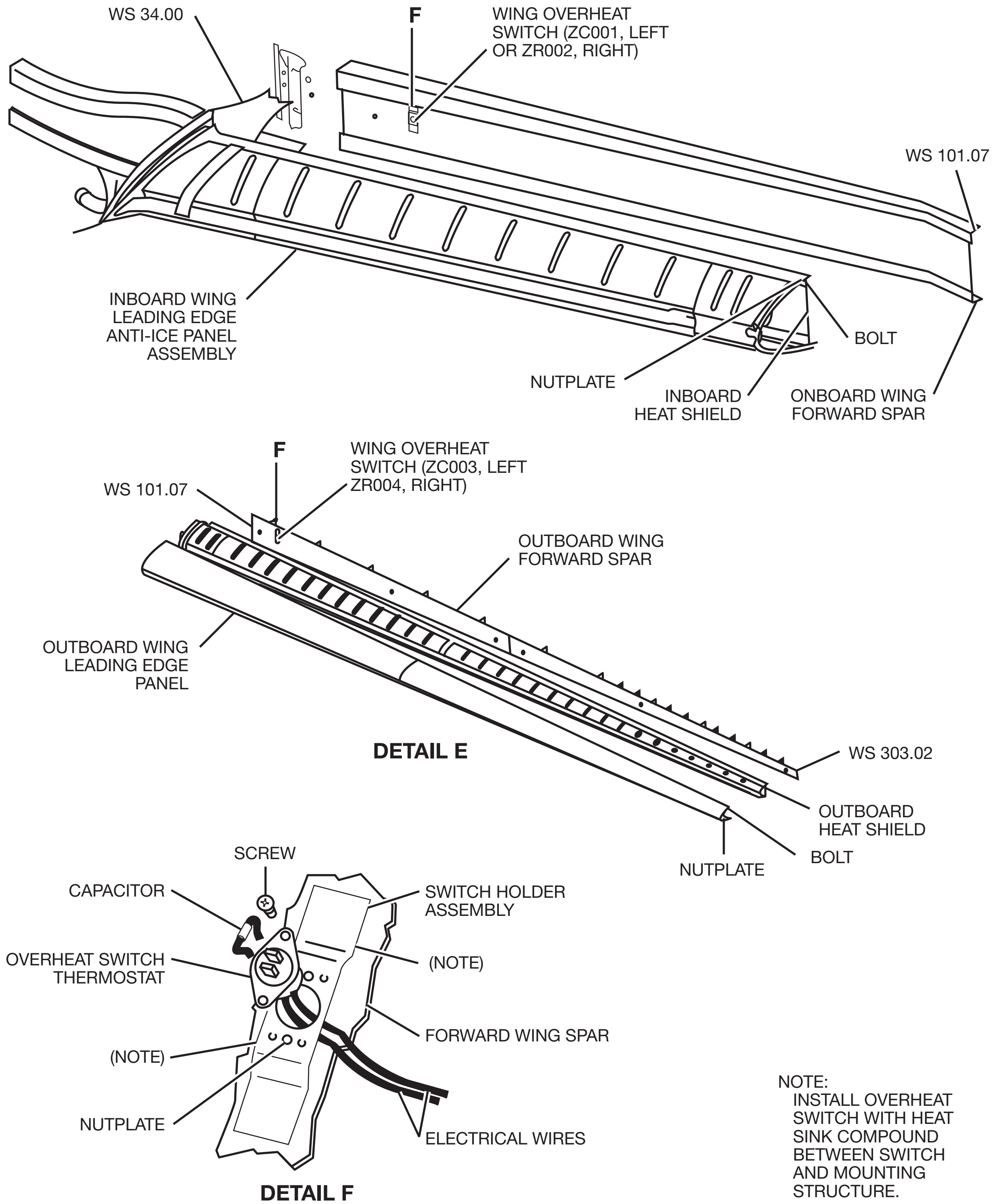


Figure 30-9. Wing Anti-ice Overheat Switches

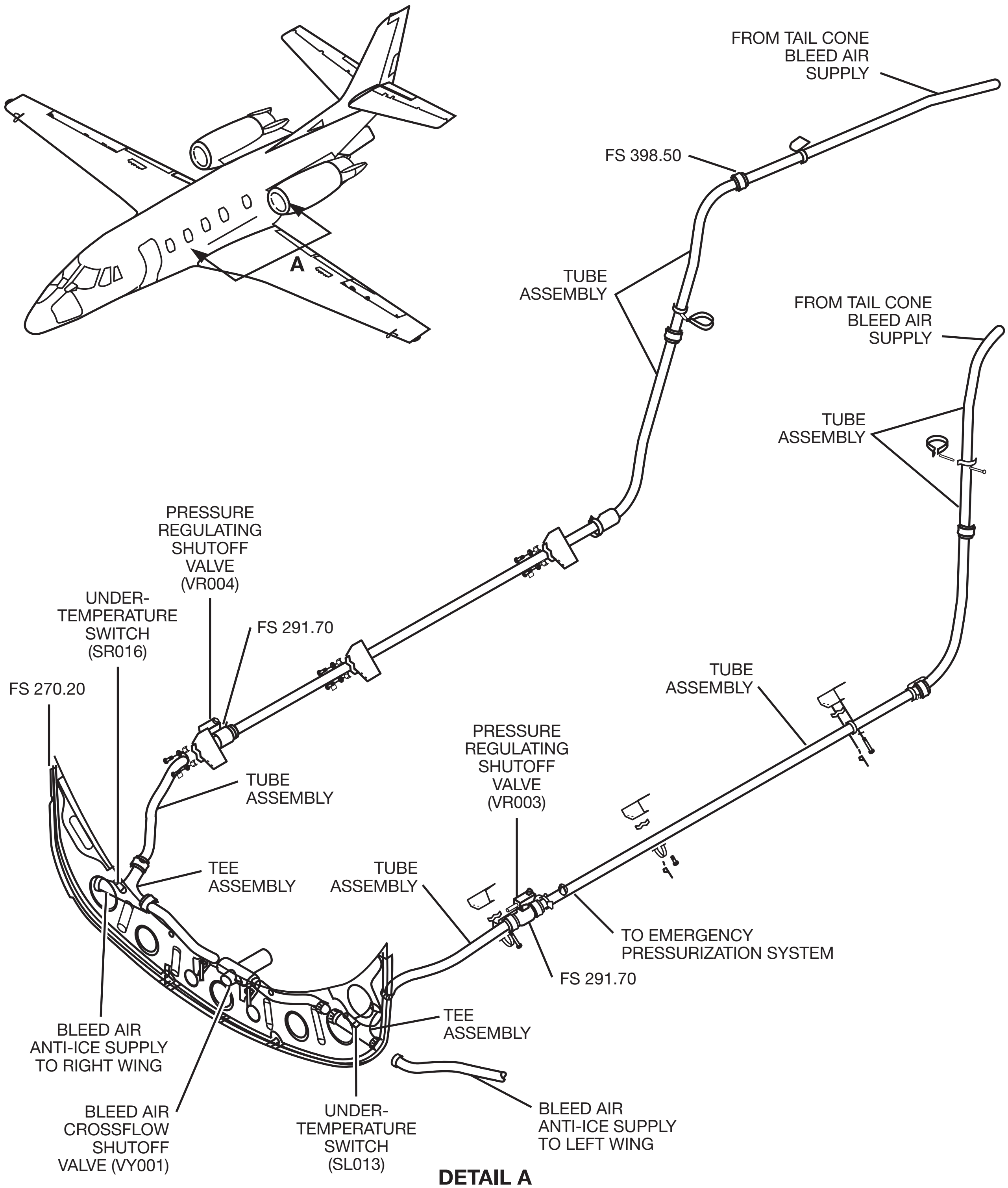


Figure 30-10. Wing Anti-ice Plumbing and Valves

Wing Supply System

The anti-ice bleed air supply involves the routing and control of engine bleed air from the tail cone, forward along the fuselage above each wing to the wing anti-ice manifold assemblies at the wing root leading edge (Figure 30-10).

The wing anti-ice valves (pressure regulating shutoff valves) are normally powered closed. Setting the L–R WING/ENGINE ANTI-ICE switch to the WING/ENGINE ON position and having a ground applied to the relay, removes power from the valve. In the absence of electrical power the valve is driven open by the upstream pressure. When the downstream pressure becomes greater than the upstream pressure, the valve closes, regardless of electrical power.

The wing leading-edge, pressure-regulating shutoff valves control the bleed air pressure downstream of the valves to 16 psig ±3 psig.

The wing anti-ice supply bleed-air crossflow valve is controlled by a two-position WING XFLOW switch, on the switch panel. During single-engine operation, the bleed-air crossflow shutoff valve activates, allowing the operating engine to supply anti-icing to both wing leading-edge anti-ice panel assemblies.

The WING XFLOW (ON) position, in conjunction with the WING/ENGINE ANTI-ICE L or R switch, applies power and opens the bleed-air crossflow shutoff valve, allowing anti-icing capability to both wings.

The WING XFLOW (OFF) position, removes power and closes the bleed air crossflow shutoff valve.

The wing anti-ice supply bleed-air under temperature switches are at FS 270.20 in the wing root area. The undertemperature switches monitor the temperature of bleed air entering the wing leading-edge anti-ice panel assemblies. When the bleed air passing over the undertemperature switch is less than 230°F (110°C), the switch closes and the appropriate L–R WING ANTI-ICE annunciator (XL/XLS) or white WING ANTI-ICE COLD L–R CAS message (XLS+) illuminates to indicate the condition.

An undertemperature condition can be caused by the following:

- Insufficient bleed air flow
- Leakage in the bleed air lines
- Malfunctioning controlling components (valves, sensors, switches)
- Failure of the ram air modulating temperature control valve to fully close
- Improper air circulation at the leading edge anti-ice panels
- Electrical malfunctions

NOTES

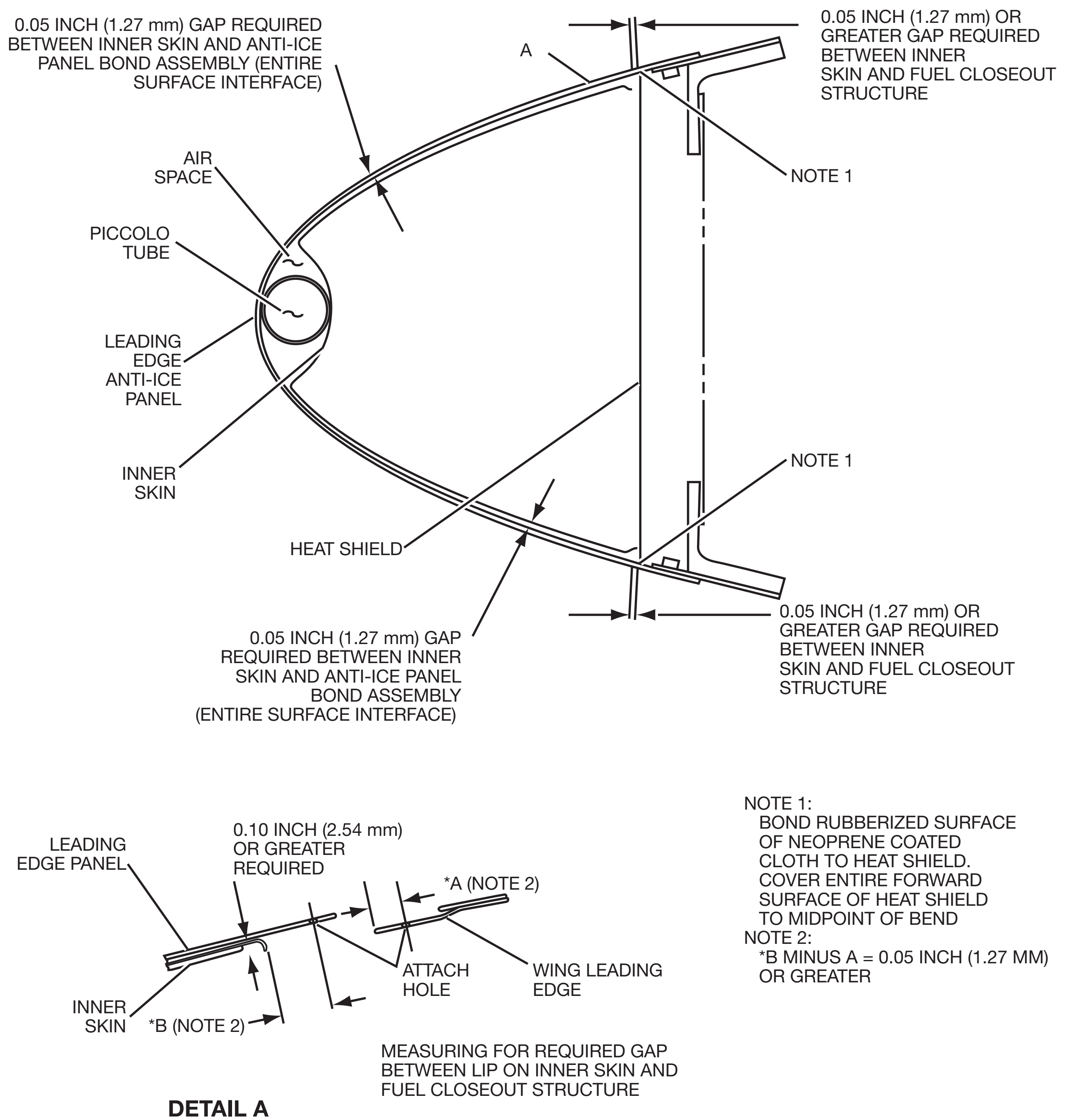


Figure 30-11. Wing Heated Leading Edge Cross-Section

Wing Anti-ice Panel Assembly

The wing leading edge anti-ice panel assembly consists of:

- Inboard and outboard stainless steel leading edge assemblies with inner diffusers
- Heatshields
- Piccolo tubing

A two piece inboard and outboard piccolo tube runs the entire length of the wing leading edge anti-ice panel. The piccolo tube has holes drilled at various spacing and angles to provide proper bleed-air heat distribution to the wing leading-edge. The wing leading-edge anti-ice panel assembly is divided into two distinct chambers (Figure 30-11).

In the first chamber, bleed air from the wing anti-ice valves is supplied to the inboard and outboard piccolo tubing which runs the entire length of the wing leading edge. After the bleed air exits the piccolo tube, the bleed air impinges on the leading edge of the wing. An inner lining directs the air flow near the leading edge to extract the maximum amount of heat possible. Spent bleed air is then discharged from the wing leading edge cavities through overboard vents in the lower surface of the outboard aft wing.

There is a second chamber between the fuel bays and the first chamber. It is vented by a ram air scoop on the bottom side of the wing root area. This chamber prevents hot bleed air or fuel vapors from accumulating. Ram air is kept separate from the bleed air chamber at all times. Ram air travels outboard in the wing and exits into the last bay of the wing, which is a dry bay. Air exits the wing assembly at the trailing edge of the wing tip.

Wing overtemperature switches, inboard wing/fuselage overheat and inboard wing overheat switch, are set to close when the temperature in the wing leading edge cavity exceeds 160°F (71°C). A closed wing overheat switch causes the L–R WING O'HEAT annunciator (XL/XLS) or amber WING ANTI-ICE OVERTEMP L–R CAS message (XLS+) to illuminate; and the respective wing pressure regulating shutoff valve closes, shutting off bleed air to the overheated wing panel. Overtemperature indication may indicate bleed air leaks at monitored locations. Such leaks are a hazardous condition which must be immediately investigated and corrected.

XLS aircraft incorporate an additional 220°F switch in each wing, mounted on the lower surface of the inboard leading edge panel. If the panel reaches 220°F, the respective wing pressure regulating shutoff valve will close. The 220° temperature sensor was relocated to the wing root adjacent to the bleed air tee assembly on the XLS+.

CAUTION

Operating the system on the ground on an extremely hot day, with engines at a setting of 70% N_2 or greater, may cause an overtemperature indication even though there is no system failure.

The engine anti-ice switches operate both the left and right wing leading edges and engine air inlet systems. Operate the system only long enough to see the annunciator lights extinguish, then shut the system down. Continued operation of the system may cause damage to the heated panels.

Ram airflow is not available to precool the engine bleed air during ground operation. Engine operation above approximately 70% N_2 can illuminate the L–R AIR O'HEAT annunciator. The engines must not be run above 70% N_2 for greater than 1 minute unless the bleed systems—environmental and anti-ice systems—are selected off.

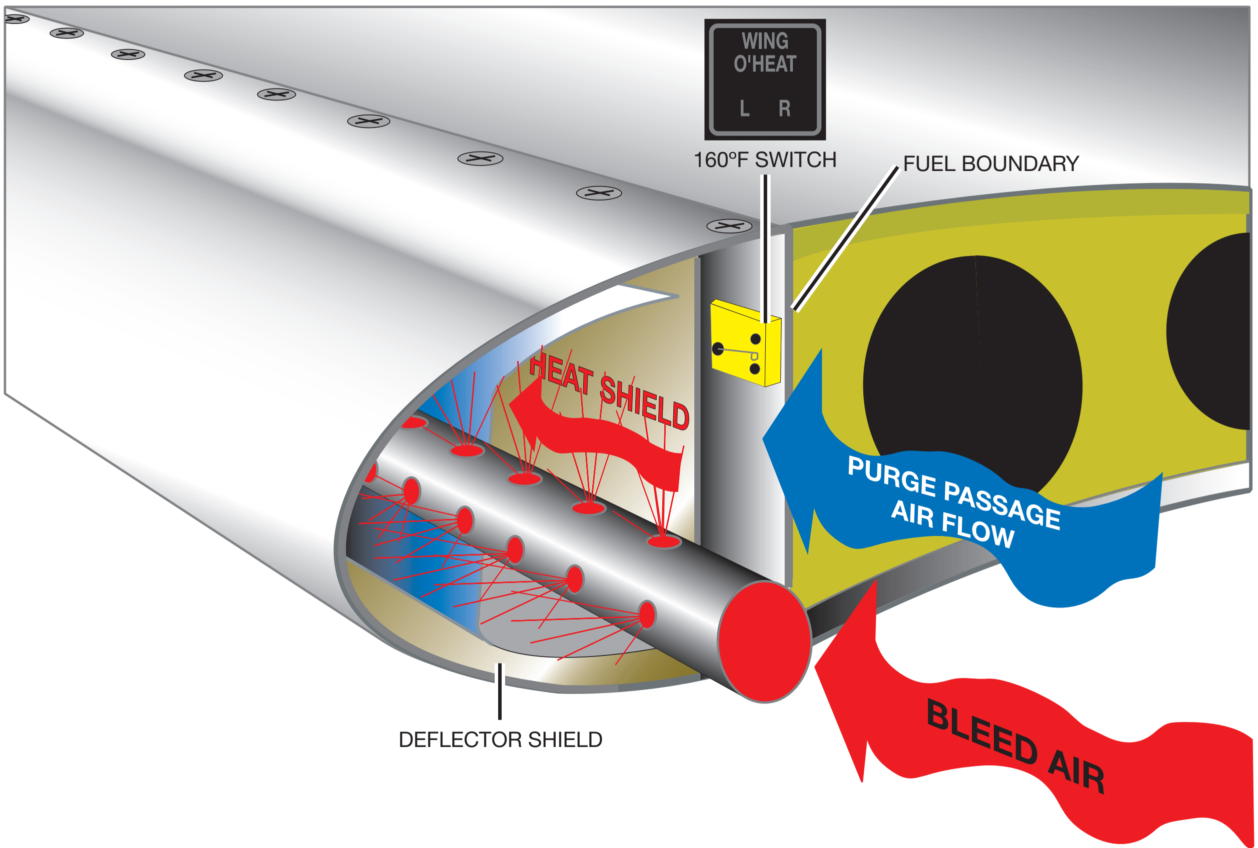


Figure 30-12. Wing Leading Edge Cross Section

OPERATION

The bleed-air heated-wing leading-edge is operated by placing the WING/ENGINE ANTI-ICE switches in the ON position (Figure 30-12). When either switch is in the WING/ENGINE ON position, electrical power is removed from the engine inlet anti-ice pressure regulating valve and the wing anti-ice pressure regulating shutoff valve. This allows bleed air to flow to both the engine inlet assemblies and the wing leading edge anti-ice assemblies.

The wing anti-ice systems are connected to bleed ports by a tee arrangement with the engine inlet anti-ice systems. Bleed air is routed from the tee fittings, through the pylon-mounted precoolers. Downstream of the precoolers, the bleed air passes overtemperature sensors for the temperature control systems. The temperature sensors feed a signal to the precooler controller/actuator. The controller/ actuator contains a linear actuator that moves the door on the scoop, and performs the temperature control function. The bleed air then passes over the overheat switches—which are set at 560°F ± 10°F (293°C ± 5°C). Next, the bleed air is distributed to the cabin pressurization system and the wing leading-edge anti-ice systems. From the cabin pressurization system, the bleed air is routed forward to the pressure regulating shutoff valves. Downstream from the pressure regulating shutoff valves, the two wing supply lines are connected by a crossfeed line, with a crossfeed shutoff valve connecting both leading-edge systems. At the junction with the leading edges of the wings, the bleed air passes over the undertemperature switches, which annunciates a cold wing condition. The bleed air then flows into the wing leading edge assemblies, piccolo tubes that distribute the hot bleed air along the leading edges.

DIAGNOSTICS

Maintenance

Maintenance practice for the wing leading-edge bleed-air anti-ice supply system components consist of:

- Removal/installation of under temperature switches
- Wing structure overheat switches
- Bleed-air crossflow (isolation) valve
- Pressure regulating shutoff valves
- Bleed-air interconnect tubing, tee fittings
- Ferrule couplings
- Clamps and bracket assemblies.

NOTE

Any time anti-ice supply bleed air connections are made, a leak check must be performed.

NOTES

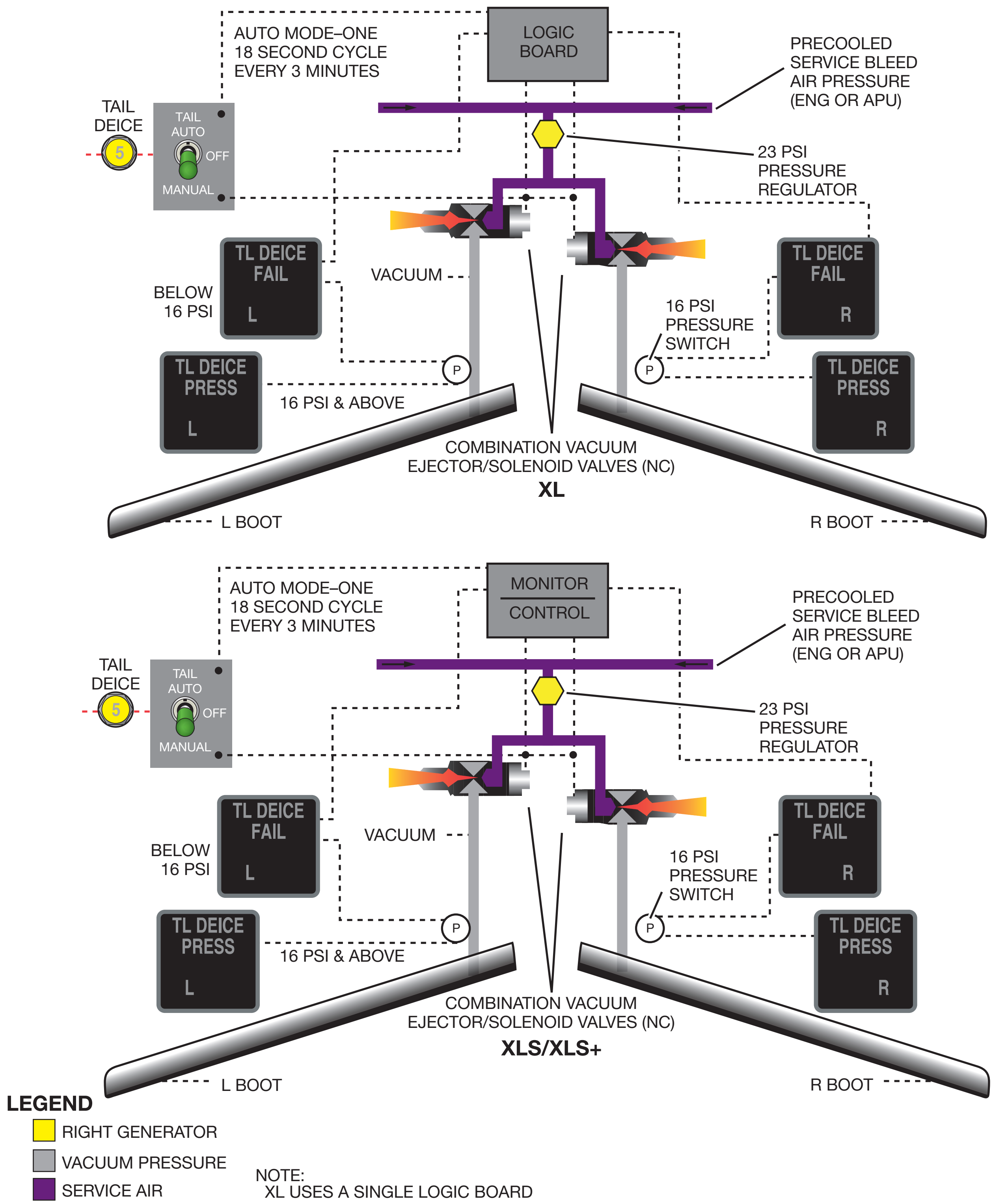


Figure 30-13. Pneumatic Deice System—Boots Inflated

PNEUMATIC (TAIL) DEICE

DESCRIPTION

The tail deice system for the horizontal stabilizer is a pneumatic boot system (Figure 30-13). The pneumatic boots are bonded to the horizontal stabilizer leading edge. Their function is to break the ice buildup on the airfoil leading edges.

The major components of the pneumatic deice system are:

- TAIL deice control switch
- Deice timer/logic board
- Two tail deice control valves
- Two deice pressure switches
- Deice annunciators (L–R TL DEICE PRESS. L–R TL DEICE FAIL)
- Left and right deice boots that are bonded to the horizontal stabilizer leading edge

Pressure regulated bleed air, controlled by the control valves and timer, alternately inflates and deflates the pneumatic boots.

CONTROLS

The AUTO–OFF–MANUAL TAIL deice control switch is on the environmental-pressurization (tilt) panel with the other anti-ice/deice switches. It is a three-position switch (Figure 30-13).

Two identical pressure regulating tail deice control valves are used in the horizontal stabilizer deice system. The left and right tail deice control valves are in the tail cone compartment on the left side. Both valves are accessible through the forward tail cone access door. When a new tail deice control valve is to be installed, check the position of the vent port in relation to the old valve.

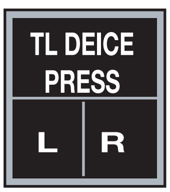
There are two pressure switches in the deice lines that actuate an annunciator to provide a visual indication to the pilot of proper operation of the deice boots. The pressure switches operate at 16 ±1 psi. The left and right pressure switches are inside the vertical stabilizer on the left and right sides, below the horizontal stabilizer. They are accessible through the vertical stabilizer access panel.

The inflation cycle of the pneumatic deice boots is controlled by the surface deice timer in the pilot side console printed circuit board (PCB) or left nose on XLS/XLS+.

NOTES

OPERATION

The pneumatic boots are operated by momentarily placing the TAIL deice switch to the MANUAL or AUTO position. Bleed air is extracted from the engines and routed by tubing to the deice control valves, then to the rubber deice boots. The bleed air is controlled by the pressure regulator valves and a timer. The TAIL deice switch MANUAL position is a momentary position that actuates both deice control valves, which in turn inflates both tail deice boots simultaneously as long as the switch is depressed. After inflating, the deice boots are deflated and held down by a vacuum ejector built into the deice control valves. A visual indication of boot inflation is provided by illumination of the TL DEICE PRESS L or R annunciator (XL/XLS) or white TAIL DE-ICE PRESS ON L–R CAS message (XLS+) (Figure 30-14).



L/R TL DEICE PRESS
Illuminates steady to indicate the respective horizontal stabilizer boot has inflated properly. With the deice switch in AUTO, normal operation is indicated by an 18-second cycle period:
Left light illuminates for 6 seconds, light extinguishes for 6 seconds, right light illuminates for 6 seconds.
The cycle will repeat approximately three minutes later. Deice switch in manual illuminates the L and R lights simultaneously.

XL/XLS ANNUNCIATOR

TAIL DE-ICE PRESS ON L-R			
Color	**Inhibited By**		**Debounce**
White	LOPI	TOPI	Standard

The 560XLS+ uses a rubber boot to deice the tail vertical and horizontal surfaces. The pilots select a switch which sends service air to inflate the boots, causing the ice to pop off. This message is displayed when there is air pressure in the boot. In the service air supply system, there is a pressure switch which sends a ground signal to the EICAS system when the pressure is over 16 PSI. When the EICAS receives the ground, it posts the message for the respective side. After popping the ice off, the boot deflates, and the pressure switch sends an open signal to the EICAS, which removes the message.

XLS+ CAS MESSAGE

Figure 30-14. Tail Deice Indications

NOTES

When the AUTO TAIL position is selected on the deice control switch, the timer is activated to start the inflation cycle. During the first six-second interval, the left horizontal stabilizer boot inflates and the L TL DEICE PRESS annunciator (XL/XLS) or white TAIL DE-ICE PRESS ON L CAS message (XLS+) illuminates. During the next six-second interval (7 to 12 seconds) the left horizontal stabilizer boot deflates and the L TL DEICE PRESS annunciator (XL/XLS) or white TAIL DE-ICE PRESS ON L CAS message (XLS+) extinguishes. Then during the final six-second interval (12 to 18 seconds), the right horizontal stabilizer boot inflates and the R TL DEICE PRESS annunciator (XL/XLS) or white TAIL DE-ICE PRESS ON R CAS message (XLS+) illuminates. For the remainder of the cycle, the boots are held down with vacuum from the ejectors and the TL DEICE PRESS annunciators (XL/XLS) or white TAIL DE-ICE PRESS ON CAS messages (XLS+) are extinguished. Whenever the boots are not in the inflation portion of the cycle, vacuum is applied to deflate and hold down the boots. After three minutes, the boots will repeat this cycle until selected OFF.

In the event that the pressure to the boot does not reach 16 psig (110.3 kPa), an amber TAIL DEICE FAIL L/R annunciator (XL/XLS) or amber TAIL DE-ICE FAIL L–R CAS message (XLS+) illuminates. Additionally, if at any time within the AUTO cycle the pneumatic boots do not cycle appropriately, the TAIL DEICE FAIL L/R annunciator (XL/XLS) or amber TAIL DE-ICE FAIL L–R CAS message (XLS+) illuminates (Figure 30-15).

NOTE

The total cycle may vary from 16 to 20 seconds.

L/R TL DEICE FAIL
Annunciator flashes after the system is selected on, and one of the following failures occurs:
- Tail deice valve has a loss of voltage.
- Tail deice system has a loss of pressure during a six second cycle ON time.
- Activates MASTER CAUTION lights.

XL/XLS ANNUNCIATOR

TAIL DE-ICE FAIL L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
When a failure of the tail de-icing system is detected by the Tail De-Ice PC Card, the card sends an open signal to the EICAS system, which posts the message for the respective side. When the tail de-ice system has normal operation, it sends a ground signal and the EICAS removes the message.			

XLS+ CAS MESSAGE

Figure 30-15. Tail Deice Fail Indications

NOTES

30 ICE AND RAIN PROTECTION

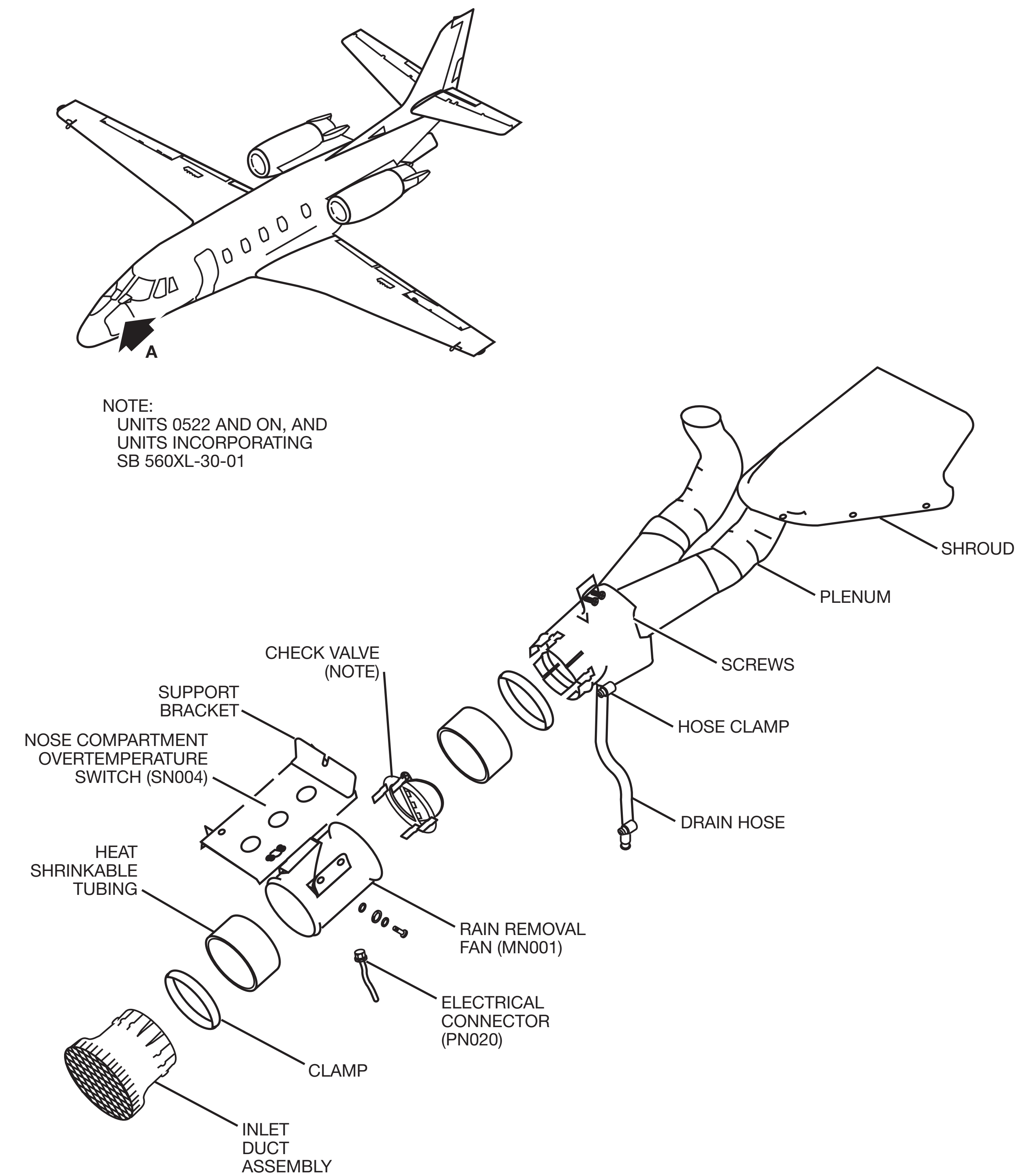


Figure 30-16. Windshield Rain Removal System

WINDSHIELD RAIN REMOVAL

DESCRIPTION

This section describes the windshield rain removal system. A nose compartment blower motor forces air across the windshield to remove moisture.

The windshield rain removal system consists of:

- Blower motor
- Shroud assembly that directs airflow
- Ducting
- Wiring
- WINDSHIELD AIR ON/OFF rain removal switch

OPERATION

The windshield rain removal system supplies forced air from a two-speed blower motor, through ducting to a shroud assembly which serves as a nozzle, directing air across the exterior surface of the windshield (Figure 30-16).

Electrical power for the blower motor in the nose compartment is supplied from a 15 amp W/S AIR circuit breaker on the left cockpit CB panel. The blower motor is controlled by an ON/OFF WINDSHIELD AIR switch. The switch is on the anti-ice switch panel.

If the WINDSHIELD AIR switch is in the ON position the blower motor runs at high speed. This is the rain removal mode. If the WINDSHIELD AIR switch is in the OFF position, the blower is off. However, if the nose compartment overtemperature switch detects nose compartment temperature greater than 95°F (35°C), the blower motor operates at low speed. This is the avionics equipment cooling mode.

NOTES

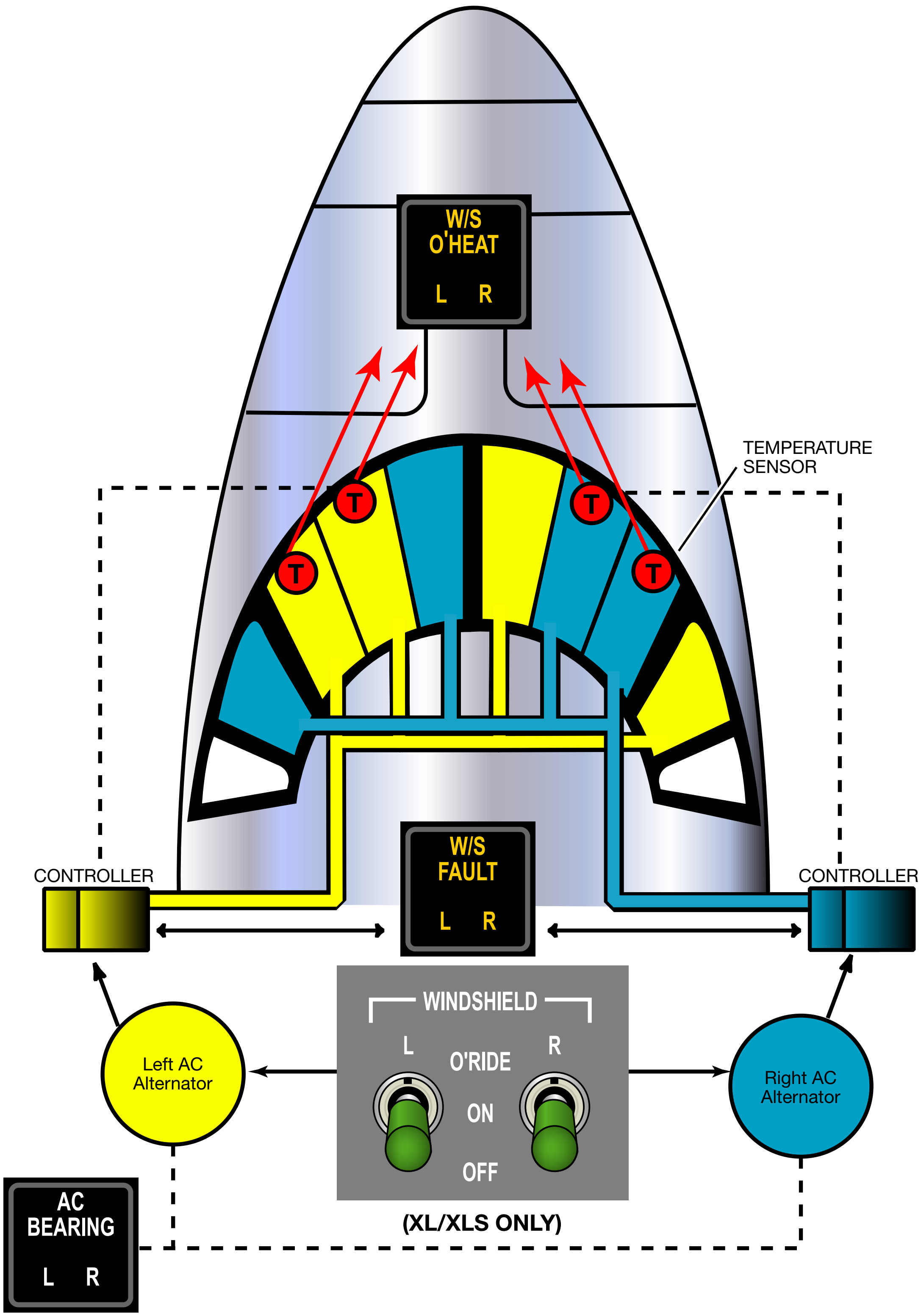


Figure 30-17. Electrically Heated Windshield System

ELECTRIC HEATED GLASS WINDSHIELDS/ SIDE WINDOWS ANTI-ICE

DESCRIPTION

This section describes the heated glass windshield anti-ice system utilized on the aircraft (Figure 30-17).

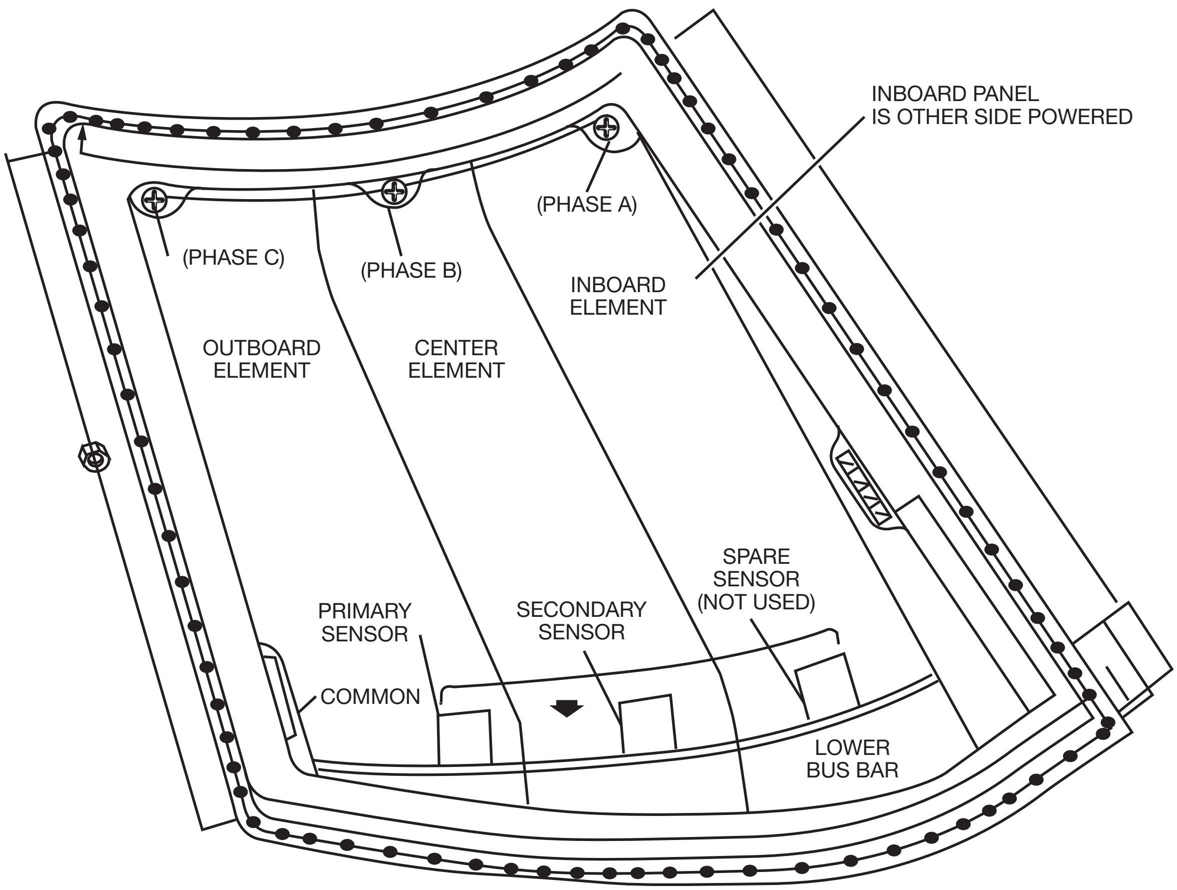
The system consists of:

- Left and right electrically heated windshields
- Left and right electrically heated forward side windows
- Electrical control units in the tail cone
- Associated switches
- Annunciators
- Relays and electrical wiring
- Anti-ice and defog capabilities for these flight compartment windows

CAUTION

Do not apply unauthorized rain repellent coatings or compounds to the electric heated glass windshield or associated heated glass side windows. Surface Seal™ is the only authorized rain repellent coating. Apply only with windshield manufacturer authorization and instructions.

NOTES



LEFT WINDSHIELD SHOWN, RIGHT WINDSHIELD THE SAME

Figure 30-18. Electrically Heated Windshield Assembly

OPERATION

The windshield assembly is laminated, all glass construction with bonded fiberglass edge attachments. Heating is accomplished through electrically conductive film, applied to the inner surface of the outer glass ply (Figure 30-18). Power is provided by two 3.0 KVA AC alternators (one on each engine) supplying 115/200 volt/3-phase power at a frequency of 200 to 400 Hz. The left and right electric heated main windshields are divided into three heated zones, each utilizing one phase of the AC power (provided by the alternators). The left and right electric heated forward side windows are heated as one section and are electrically connected to main windshields in parallel.

There are three integral temperature sensors in each windshield assembly. One sensor is utilized as a primary, one as a secondary (backup) sensor, and the third is unusable, because it is located in an area that is heated by the opposite controller. The primary and secondary temperature sensors are connected electrically to a control unit (one for each windshield) in the baggage compartment. Left and right control units monitor windshield temperature via the primary sensor. If a fault occurs in either primary sensor, the control unit automatically switches to the secondary (backup) sensor to provide constant temperature monitoring. Left and right main windshields are regulated at a temperature of 110°F (43°C). Each control unit also incorporates a ramp heating feature which is initiated each time the system is switched on. This function heats the windshields slowly to the regulated temperature.

NOTES

CONTROLS AND INDICATIONS

XL/XLS

There are two WINDSHIELD L and R switches that control the system, on the switch panel, forward of center pedestal.

The toggle-type switches have three positions:

- O'RIDE
- ON
- OFF

Placing either left or right windshield switch to the ON or heat (center position) will initiate a ramp-heating function which heats the windshield at a slower rate to regulated temperature. To heat the windshield more rapidly, the O'RIDE (upper position) may be selected. The ON position is used for normal system operation.

XLS+

The switch for the windshield anti-ice system was removed on the anti-ice system as well as override of the ramp up of temperature. On engine start the windshield ramps up to 110°F.

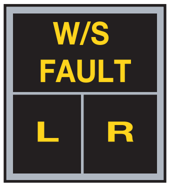
Cockpit L–R,W/S circuit breakers for the electric heated windshield anti-ice system are on the left CB panel. Other associated circuit breakers and relays are in the electrical power junction box, and may be identified by a placard on the junction box cover.

The L–R W/S O'HEAT and L–R W/S FAULT annunciators (XL/XLS) are in the annunciator panel below the cockpit glareshield. The amber WINDSHIELD OVERTEMP L–R and WINDSHIELD HEAT INOP L–R CAS messages (XLS+) are displayed on the EICAS. The L and R W/S O'HEAT annunciators (XL/XLS) or amber WINDSHIELD OVERTEMP L–R CAS message (XLS+) are illuminated when either or both windshields reach an overheat condition. An overheat condition occurs when the primary or secondary temperature sensors in windshield detect a temperature above 140°F (60°C). The L–R W/S FAULT annunciators (XL/XLS) or WINDSHIELD HEAT INOP L–R CAS message (XLS+) are illuminated anytime a fault is detected in the system. These fault conditions include:

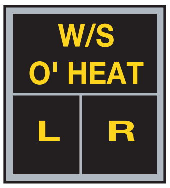
- Shorted or open circuitry/wiring
- Overheat condition
- Phase imbalance
- And/or faulty temperature sensors

Illumination of the L–R W/S FAULT annunciators (XL/XLS) or WINDSHIELD HEAT INOP L–R CAS message (XLS+) removes electrical power and shuts down the system (Figure 30-19).

NOTES



L/R W/S FAULT
Annunciator illuminates steady for eight seconds then will begin flashing if the windshield temperature controller has detected a fault prior to both engine starts while the aircraft is on the ground. Activates MASTER CAUTION lights. Annunciator flashes if the windshield temp controller detects a fault after engine starts with aircraft on the ground or in flight. Activates MASTER CAUTION lights.



L/R W/S O'HEAT
Flashes to indicate the respective windshield has over-heated. The W/S FAULT also illuminates and windshield heat shuts down. The system may automatically reactivate after cooling followed by another system shutdown at the overheat point (cycle on and off).

XL/XLS ANNUNCIATORS

WINDSHIELD HEAT INOP L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	*8 Second
	*ESDI	SIPI	
The windshield is electrically heated. The heating is controlled by a windshield heat controller. The windshield heat controller is powered when the aircraft battery switch is turned on. **This message is displayed when the controller has detected a failure.** When a failure is detected, the controller sends a ground signal to the EICAS system, which displays the message. When the input is open, the message is not displayed.			

WINDSHIELD OVERTEMP L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	*4 Second
This message is displayed when the windshield controller has detected an overheat situation. The overheat could result in structural damage. When the controller detects the overheat, it sends a ground to the EICAS system, which displays the message. An open signal removes the message.			

XLS+ CAS MESSAGES

Figure 30-19. Windshield Heat Indications

An integral self-test circuit allows the electric-heated windshield anti-ice system to be checked for integrity, by positioning the aircraft rotary test switch to the W/S TEMP position. Initiation of this test illuminates the W/S O'HEAT L and R annunciator (XL/XLS) or WINDSHIELD HEAT INOP L–R CAS message (XLS+) and the W/S FAULT L and R annunciators (XL/XLS) or WINDSHIELD HEAT INOP L–R CAS message (XLS+) for approximately 3–4 seconds. A momentary application of power to the windshields also verifies the integrity of the temperature sensors, the electrical circuit and the tail cone units.

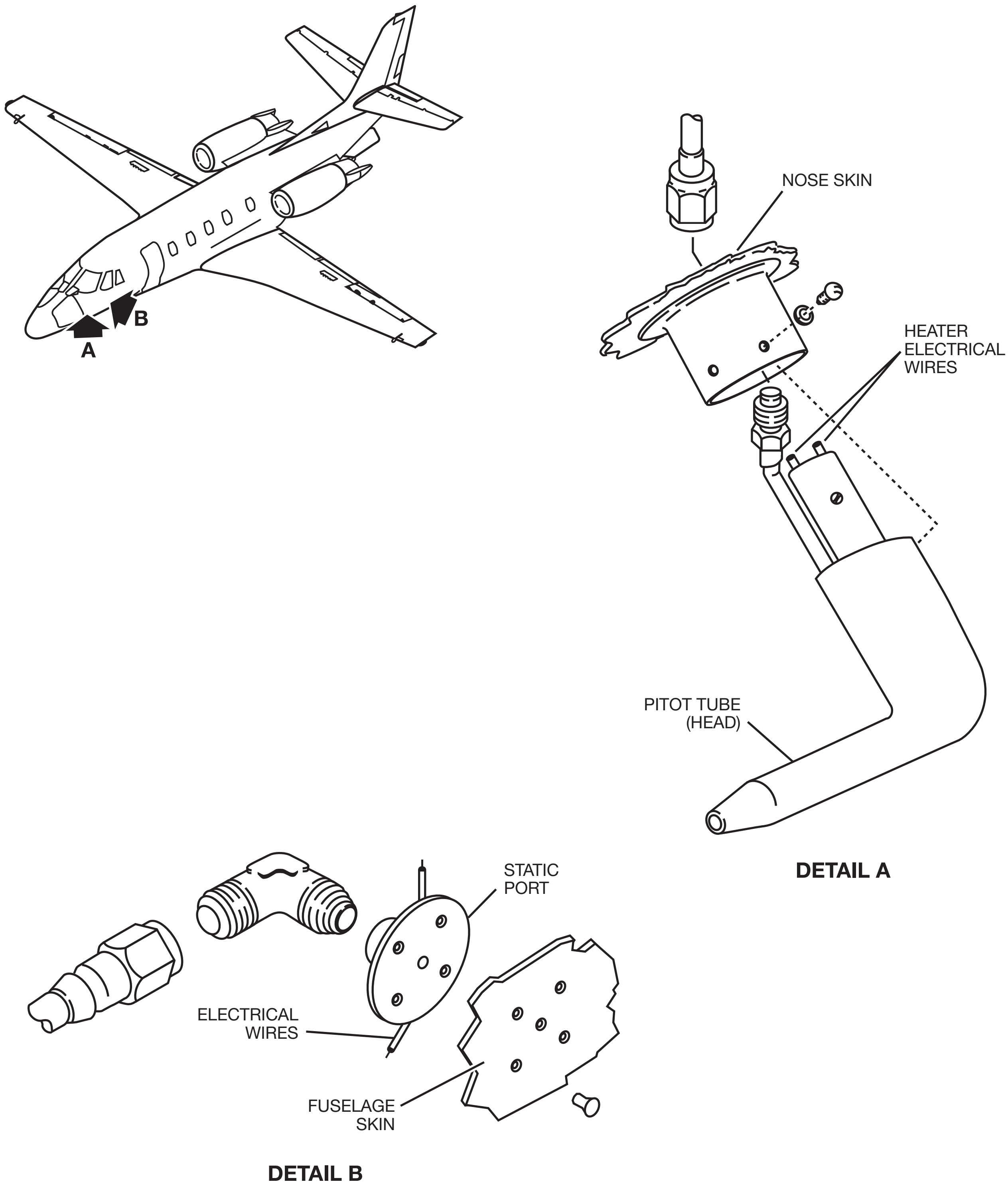


Figure 30-20. Pitot/Static Anti-ice System

PITOT/STATIC ANTI-ICE SYSTEM

DESCRIPTION

This section provides troubleshooting procedures and maintenance practices for maintaining the pitot and static anti-ice systems. Anti-icing for the pitot tubes and static ports is accomplished electrically to prevent ice formation. Should the pitot tube(s) and/or static port(s) become restricted or blocked from ice formation, unreliable or complete failure of flight instruments and other pitot-static supported systems occurs.

The pitot static anti-ice systems are comprised of electrically-heated pitot tubes and electrically-heated static ports. A warning system consists of current sensors which illuminate annunciator panel warning lights in case of pitot static anti-ice system heating element(s) failure. The PITOT & STATIC anti-ice switch is on the ANTI-ICE panel and controls the pilot, copilot, and standby pitot static anti-ice systems as well as the AOA vane heater.

The pitot static anti-ice system consists of three independent systems:

- Pilot pitot static system
- Copilot pitot static system
- Back up or standby system

Each system consists of a pitot probe and a left and right static port (Figure 30-11). The three systems are required to provide redundancy in the event of system failures. The pitot static systems provide altitude and airspeed indications to the crew, as well as provide a reference pressure source for the cabin pressure gauge. All of the pitot probes and the static ports are electrically anti-iced.

The pitot/static anti-ice is generally a trouble-free system. Normally a malfunction (such as an inoperative heating element) is indicated by illumination of the LR P/S HTR and STBY P/S HTR annunciators (XL/XLS) or amber PITOT/STATIC COLD L–R–STBY CAS message (XLS+) (Figure 30-21). An illuminated annunciator, with the PITOT & STATIC anti-ice switch in the ON position, indicates that one or more of the four heaters (on the left or right side) is inoperative. The element may be burned out or may have defective electrical connection wiring. To determine which heater is defective and to isolate the fault, perform an operational check of the system. The pitot/static anti-ice heating element is an integral part of the pitot tube assembly and the static port assembly. A defective heating element requires replacement of the pitot tube or static port.

P / S HTR L | R	**L/R PITOT/STATIC HEATER** ON GROUND—Annunciator illuminates steady to indicate the pitot-static heater switch is OFF. IN FLIGHT—Annunciator flashes to indicate the switch is OFF or an inoperative heating element, activates MASTER CAUTION lights.
STBY P/S HTR	**STBY P/S HTR** ON GROUND—Annunciator illuminates steady to indicate the standby pitot-static heater switch is OFF. IN FLIGHT, annunciator flashes to indicate the stand-by pitot-static heater is off or inoperative, activates MASTER CAUTION lights.

XL/XLS ANNUNCIATORS

PITOT/STATIC COLD L-R-STBY			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
White			

The amber message(s) are displayed when the pitot/static heat is selected on, but current is not flowing in one of the heaters. It is also displayed if the heat is selected off, and the airplane is in the air. The advisory message is displayed on ground when the pitot/static switch is selected off.

XLS+ CAS MESSAGE

Figure 30-21. Pitot/Static Indications

HEATED DRAINS

DESCRIPTION

Electrically heated drains are provided to prevent ice formation that would impair normal drainage from the drains.

The left forward refreshment center and cockpit relief tube are equipped with heated drains which operate on direct current (DC) voltage. The heated drains may be placed forward, midship, aft or a combination thereof, depending on the interior configuration. Electrical power is taken from the interior junction box electrical circuit.

NOTE

Power is supplied to the heated drains anytime power is applied to the aircraft, the DRAIN HEATERS circuit breaker in the interior junction box is engaged and the pitot-static switch is selected ON (XL). Heated drains on the XLS/XLS+ are controlled with the PITOT/STATIC ANTI-ICE switches.

NOTES

QUESTIONS

1. Bleed air flowing into the wing leading edge panels that is too cold is annunciated by:
   A. WING TOO CLD L/R CAS message
   B. BLEED AIR OVERTEMP L–R CAS message
   C. No CAS message appears
   D. WING ANTI-ICE COLD L–R CAS message

2. If the TAIL DE-ICE FAIL L–R CAS message appears:
   A. In MANUAL mode, considered normal
   B. IN MANUAL mode, the timer is inoperative
   C. In AUTO mode, the inflation pressure may be too low
   D. In MANUAL mode, boot did not deflate

3. The purpose of the WING CROSS FLOW switch is to:
   A. Allow hot bleed air to transfer between the wings
   B. Equalize bleed-air pressure between the engines
   C. Keep fuel levels equal in each wing
   D. Fail open during DC power failure

4. Select the correct statement concerning windshield rain removal:
   A. Windshield wipers are effective only during heavy rain
   B. Windshield is coated with a rain repellent
   C. WINDSHIELD AIR switch blows air across the windshield
   D. Both B and C

5. If the WING ANTIICE L/R annunciator illuminates:
   A. Wing is too cold; add power for more bleed-air heat
   B. Indicates failure of the wing overheat sensor
   C. Wing is too hot, reduce power to help cooling
   D. It will stay on for three minutes and restart the anti-ice cycle

6. ENG ANTI-ICE L or R annunciator remains illuminated with the ENG ANTI-ICE switch on and:
   A. 160°F overtemperature switch beneath the wing root fairing closes
   B. Temperature at the engine inlet lip is less than 60°F.
   C. XFD is selected
   D. Engine stator valve open

7. What indication would be seen if pressure to the horizontal boots was too low?
   A. No surf deice light
   B. TL DE-ICE PRESS annunciator
   C. TL DE-ICE FAIL annunciator
   D. No indication

8. During a preflight inspection, turn the pitot/static heater switch on but the P/S HTR light remained on. What component has possibly failed?
   A. True airspeed probe heater
   B. Static port heater
   C. Angle of attack probe heater
   D. All of the above

9. What indications are observed when turning on the WING/ENGINE ANTI-ICE switches when the engines are at idle?
   A. Slight rise in ITT and a decrease in N_2 speed
   B. Slight rise in N_2 speed and a decrease in ITT
   C. Green igniter light illuminates
   D. Both A and C

10. Which of the following components are powered open?
    A. Wing crossover valve
    B. Left or right wing anti-ice valves
    C. Engine inlet valve
    D. None of the above, all these valves are deenergized open

11. What would the indication be if the wing leading edge temperature was 230°F or greater?
    A. WING ANTI-ICE L or R annunciator extinguishes
    B. WING O'HEAT light would illuminate
    C. Wing ice valve would be powered closed
    D. Anti-ice valve would close and the WING O'HEAT light would come on

NOTES

CHAPTER 31
INDICATING AND RECORDING SYSTEMS

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 31
INDICATING AND RECORDING SYSTEMS

INTRODUCTION

This chapter describes and pictorially presents instruments, control panels, and components not related to a specific system. Information is also provided on components that record, store, compute data, and give visual or aural warnings from unrelated systems.

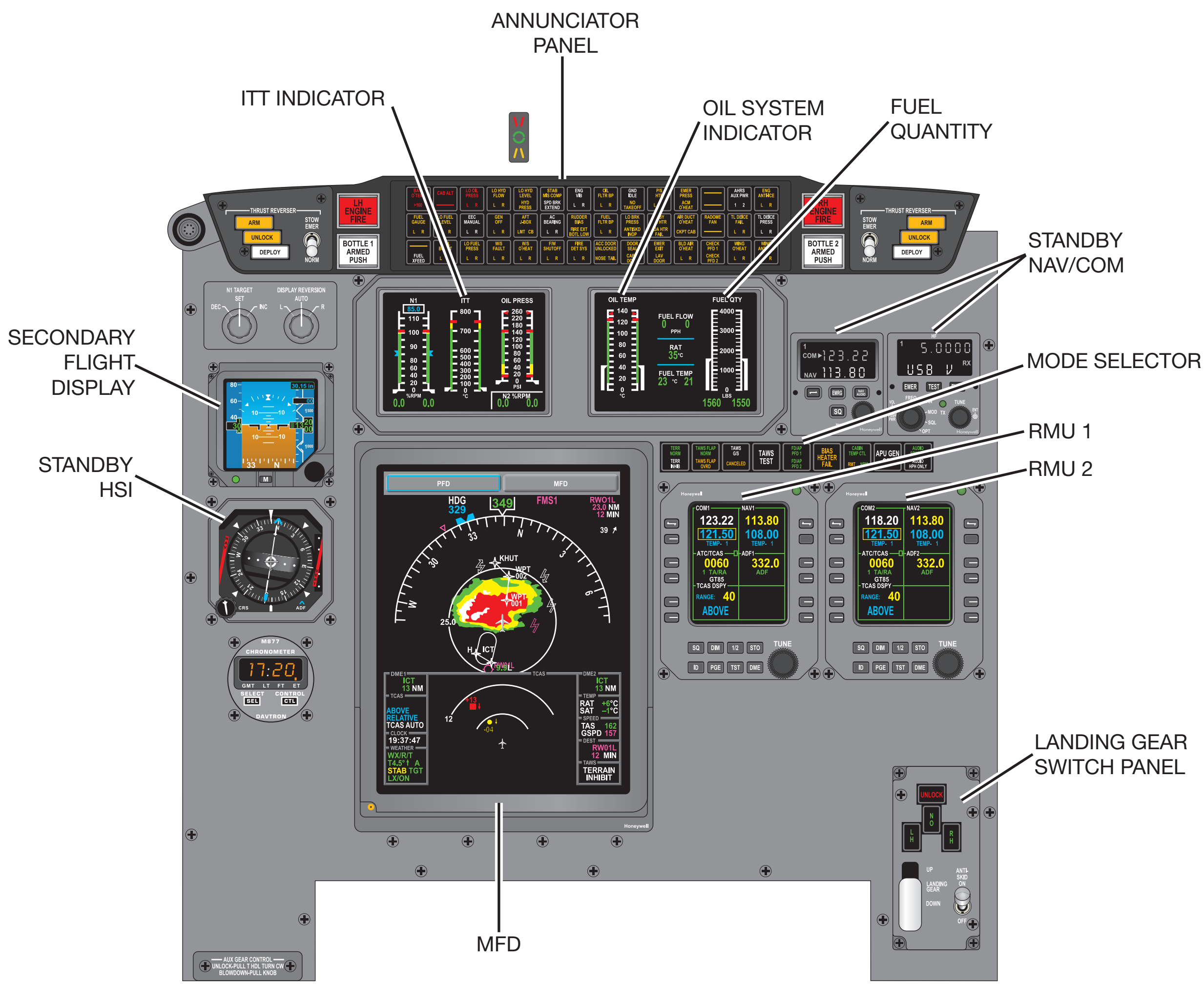


Figure 31-1. Pilot and Copilot Instrument Panel Installation (XLS)

GENERAL

This section covers the standard configurations of the instrument panel. The four individual panels described include:

- Left panel
- Center panel
- Right panel
- Tilt panel

Instruments on the panels include:

- Rotary TEST knob
- Digital clock
- Flight hour meter
- Flight data recorders
- Optics switches
- MASTER CAUTION and MASTER WARNING lights

DESCRIPTION

The instrument panel mounts all flight, engine, and miscellaneous instruments. There are also control panels and switches on the instrument panel (Figure 31-1).

Most of the instruments are on the front side of the instrument panel and do not require removal of the individual panel for instrument maintenance.

The control panels are illuminated by electroluminescent panels. For maintenance practices of these panels, refer to Chapter 33—"Flight Compartment Primary Lights—Maintenance Practices" in the *Aircraft Maintenance Manual* (*AMM*).

For CB panel maintenance, refer to Chapter 24—"Circuit Breaker Panel—Maintenance Practices" in the *AMM.*

NOTES

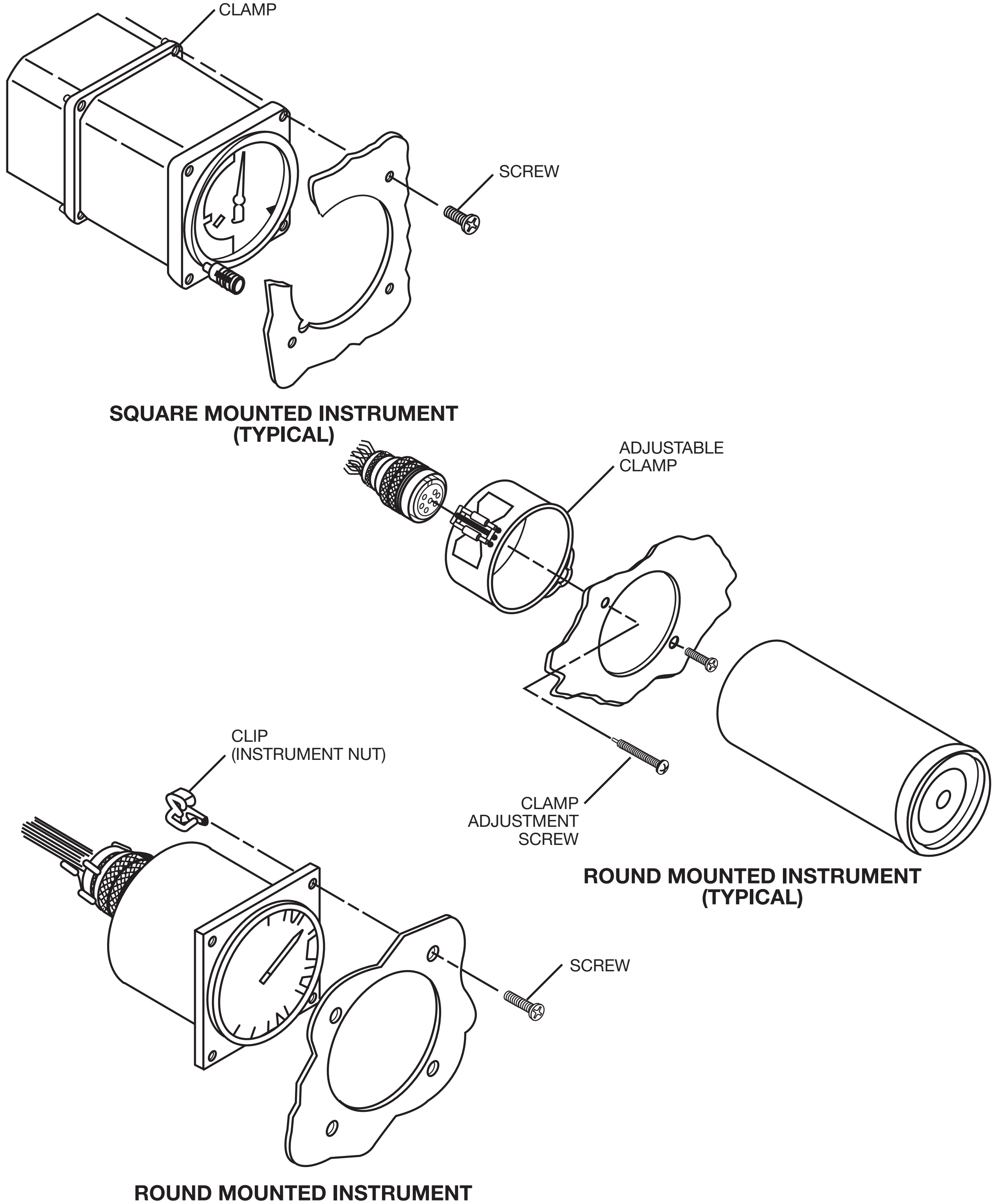


Figure 31-2. Panel Component Connections

DIAGNOSTICS

Instrument Glass Lenses

The following instructions describe safety precautions and removal of clamp-type and bezel mounted instruments (Figure 31-2). After installing glass lens instruments, clean the lens.

Before performing instrument and control panel maintenance, applicable safety precautions must be selected in accordance with the work to be accomplished. When replacing an instrument, switch, or similar components, disengage circuit breakers.

The following instructions are typical for instruments of this shape and mounting.

Safety Precautions

1. Remove power from system where maintenance is to be performed.
2. Disengage the applicable circuit breaker.
3. Tag the circuit breaker with a caution sign.

CAUTION

Do not connect battery when maintenance is in progress.

4. Deenergize the battery bus (alternate method).

   a. Disconnect the battery connects.

   b. Tag the battery with a warning sign.

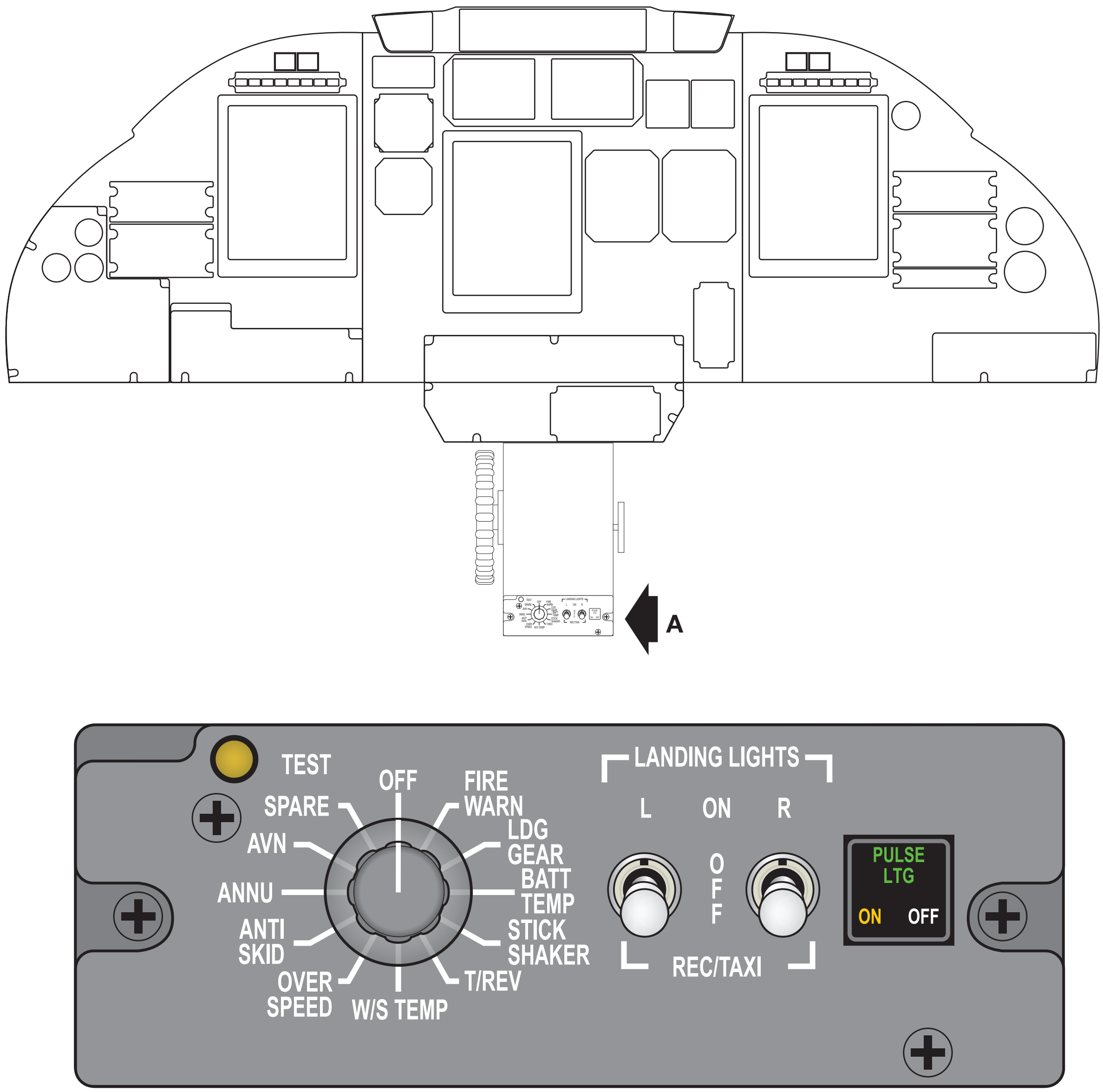
   c. Disconnect external power (if connected).

WARNING

Do not apply external power when maintenance is in progress.

   d. Tag the external power receptacle with a warning sign.

NOTES



DETAIL A

Figure 31-3. Rotary TEST Knob

Rotary Test Knob (XL/XLS)

This section describes the indications in each detent position of the rotary TEST knob. Certain indications must be present to verify a satisfactory self-test before proceeding to the next position. Following is a brief description of each of these indications (Figure 31-3).

Off Position—With the rotary knob in the OFF position, the red light above the rotary knob extinguishes and the test system is inoperative.

NOTE

The red light above the rotary TEST knob illuminates for all the other test positions, including the SPARE position.

FIRE WARN Position—With the rotary TEST knob in the FIRE WARN position, the LH–RH ENGINE FIRE lights on the fire tray illuminate.

LDG GEAR Position—With the rotary TEST knob in the LDG GEAR position, the green LH, RH, and NO lights illuminate, The red GEAR UNLOCK light illuminates, the gear warning horn sounds.

BATT TEMP Position—With the rotary TEST knob in the BATT TEMP position, the red BATT O'TEMP and >160° annunciator lights flash, The battery temperature gauge indicates 160°F, The MASTER WARNING light flashes. Press the MASTER WARNING light and verify the light extinguishes.

STICK SHAKER Position—The STICK SHAKER fires immediately on pilot and copilot columns. The angle of attack indicator needle moves to the top of the RED band.

T/REV—The left and right, ARM, UNLOCK, and DEPLOY lights illuminate steady. The MASTER WARNING RESET switchlights flash (approximately two flashes per second). Press MASTER WARNING RESET and verify light extinguishes.

W/S TEMP—With windshield heat selected on, the L–R W/S O'HEAT annunciator illuminates steady for 3 to 4 seconds then extinguish.

NOTE

If windshield heat is selected on with the engines shut down, W/S FAULT illuminates because the AC alternator is not supplying power.

OVERSPEED—The pulsating OVERSPEED audible warning horn sounds. The MADC output reverts to Functional Test Mode and PFD1/2 indicates 265 KIAS, Mach 0.4, 5,000 feet altitude and a vertical speed of 2,000 ft/min.

ANTI-SKID—With the antiskid switch on, the ANTI-SKID INOP annunciators flash for 3 to 4 seconds then extinguish. The MASTER CAUTION RESET switchlights illuminate steady during the self-test.

ANNU—Turn AVIONIC PWR switch to ON. All annunciator panel legends illuminate, and altitude alert warning audio horn sounds.

The MASTER WARNING lights and MASTER CAUTION light illuminate steady and are non-cancelable.

Both red turbine overspeed lights flash.

The engine instrument LCDs show steady 8s.

The AP OFF and YD OFF annunciators illuminate steady.

The Flight Director Mode Selector (FDMS) buttons illuminate left to right and then remain steady.

The annunciators to the right of the FDMS panel illuminate steady. They are as follows, (but may vary depending on which options are installed):

1. FD/AP PFD 1, FD/AP PFD 2
2. TERR NORM, TERR INHIB
3. GPWS FLAP NORM, GPWS FLAP OVRD
4. GPWS G/S, CANCELLED
5. GPWS TEST
6. PHONE CALL

All A/P control panel lights illuminate steady.

The green A/C ON light above the A/C switch illuminates steady.

A pulsating aural horn, which is a combination of the following 3 inputs sounds:

1. Autopilot disconnect tone (steady).
2. Altitude alert tone (steady).
3. Phone call tone (pulsating and becomes steady when the PHONE CALL button is depressed).

NOTES

AVN

Turn AVIONIC PWR switch to ON.

The MASTER CAUTION RESET switchlight illuminates steady and is cancelable.

The Flight Director Mode Selector (FDMS) buttons illuminates left to right and then remain steady.

All A/P control panel lights illuminate steady.

After a short delay the following annunciators flash indicating a successful self-test:

1. AP PITCH MISTRIM
2. AP ROLL MISTRIM
3. RADOME FAN
4. CHECK PFD1
5. CHECK PFD2

The annunciators to the right of the FDMS panel illuminate steady. They are as follows (but may vary depending on which options are installed):

1. FD/AP PFD 1, FD/AP PFD 2
2. TERR NORM, TERR INHIB
3. GPWS FLAP NORM, GPWS FLAP OVRD
4. GPWS G/S, CANCELLED
5. GPWS TEST
6. PHONE CALL

The AP OFF and YD OFF annunciators illuminate steady.

A pulsating aural horn, which is a combination of the following 3 inputs sound:

1. Autopilot disconnect tone (steady).
2. Altitude alert tone (steady).
3. Phone call tone (pulsating and becomes steady when the PHONE CALL button is depressed).

SPARE

This position is a spare, and does not activate any system.

After the test is complete, rotate the test knob to OFF.

NOTES

Figure 31-4. Davtron Digital Clock

INDEPENDENT INSTRUMENTS

DESCRIPTION

Independent instruments described in this section include:

- Left digital clock
- Right digital clock
- Flight hour meter

Independent instrument consists of components or systems that are not related to any major system of the aircraft.

DIGITAL CLOCK (CHRONOMETER)

The digital clock (12-hour) is a standard installation on the left switch panel and an optional installation on the right meter panel (Figure 31-4).

RAM AIR TEMPERATURE GAUGE

Ram air temperature (RAT) comes off the right engine TTO probe, which is also used for the right electronic engine control (EEC).

NOTES

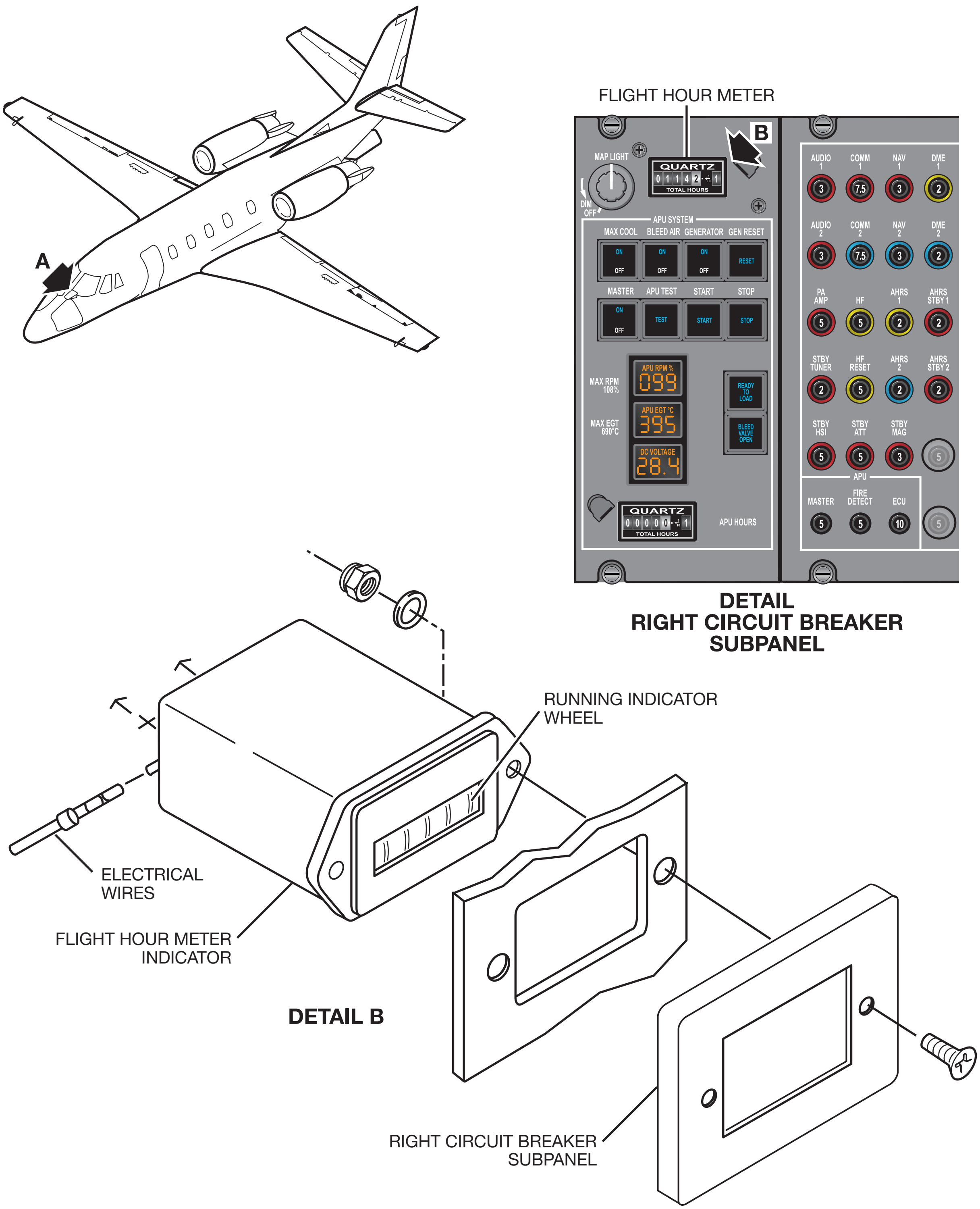


Figure 31-5. Flight Hour Meter

FLIGHT HOUR METER

Maintenance practices for the flight hour meter consist of removal/installation and preventative maintenance (Figure 31-5). Damaged or malfunctioning meter shall be replaced.

If electrical power is applied to the aircraft distribution bus with the landing gear extended to the flight position or if the landing gear left squat switch is bypassed (jumpered), disengage the FLT HR METER circuit breaker to prevent logging flight hours by the flight hour meter.

NOTES

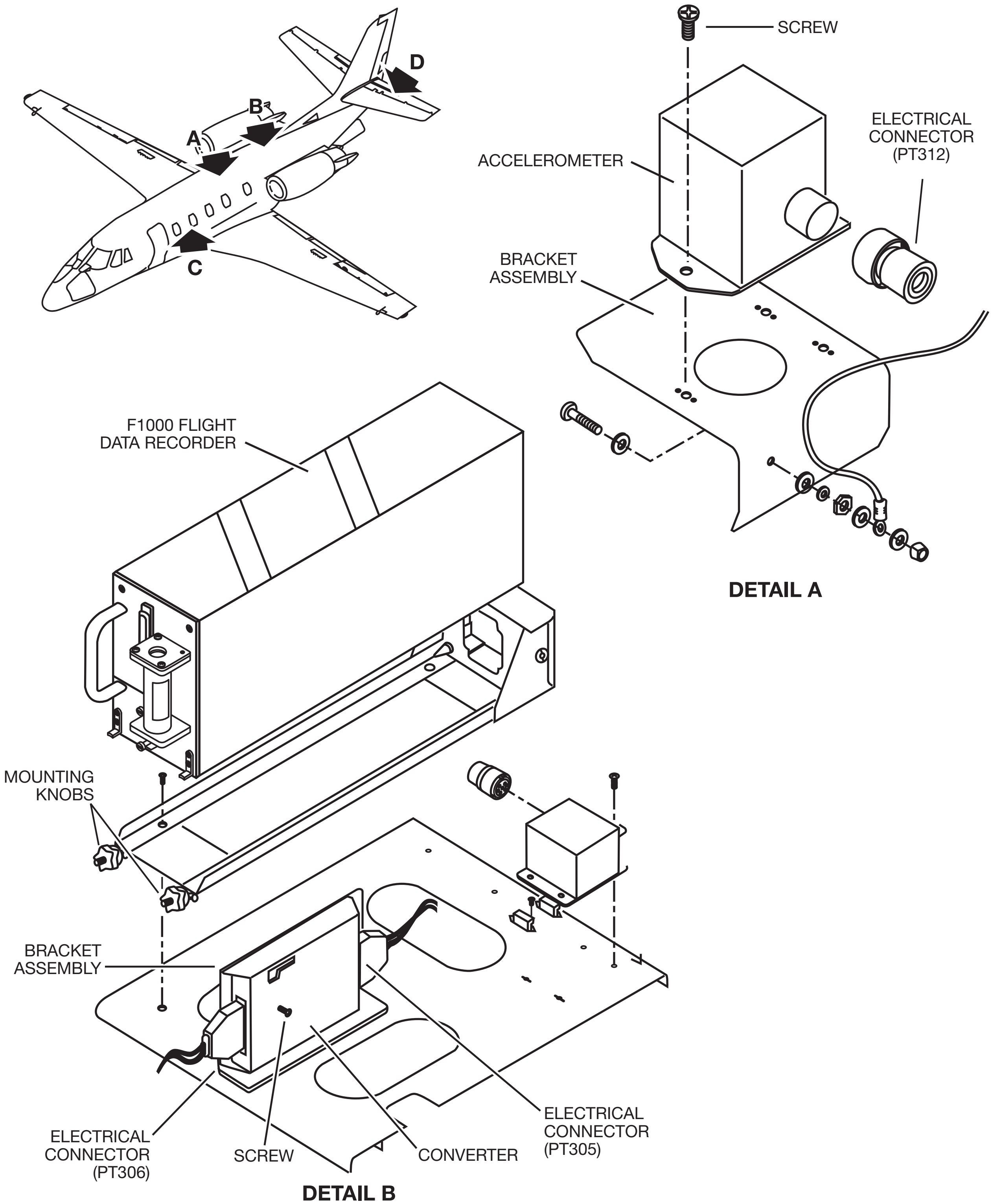


Figure 31-6. Flight Data Recorder

F1000 FLIGHT DATA RECORDER

DESCRIPTION

The Fairchild F1000 solid state flight data recorder (FDR) system consists of a solid state flight data recorder, converter, impact switch, and remote accelerometer

The FDR with solid state memory is a crash-protected airborne data recording system with complete ARINC 542/542A electric compatibility (Figure 31-6). It accepts 6 to 24 parameters. The FDR utilizes a modular crash survivable store unit (CSSU) for protection of the solid state FDR memory. The FDR is powered by 28.5 ± 0.5 VDC provided through the impact switch. The impact switch cuts power to the recorder when the switch is tripped with 5Gs for deceleration.

The F1000 system monitors the aircraft functional parameters and processes and stores the data in crash-protected solid state memory. The system also generates system performance signals that are monitored in the aircraft cockpit signifying the mission readiness of the FDR. The FDR stores the most recent 25 hours of flight history.

The F1000 system incorporates a Dukane underwater (acoustical) locator beacon. This beacon is on the recorder front panel for quick removal and/or replacement of the underwater locator beacon battery. The battery must be replaced every 6 years. A decal indicating the battery expiration date is on the front panel of the recorder.

The multi-axial accelerometer is a hermetically sealed instrument for simultaneous measurement of acceleration along two axis: vertical and longitudinal. It consists of two separate, rugged sensors responding to force along the two axis.

The flight data recorder records all data in solid state memory (no moving parts). It receives and records all flight and aircraft system information as it is received from aircraft sensors. A front-mounted automatic test equipment (ATE) connector on the FDR downloads data using a portable acquisition unit whether the recorder is on or off the aircraft. Data can be displayed and printed without removing the FDR from the aircraft. For data retrieval and analysis, refer to the *Aircraft Flight Manual (AFM)*. The ATE connector is also used with automatic or bench test equipment for final recorder checkout and for checkout of the recorder on the aircraft during calibration check.

The impact switch is a power interrupt switch that removes power from the FDR to prevent recording over data in an aircraft mishap.

The F1000 uses minicomputers for data readout, testing, and calibration. The diagnostic software are all menu-driven and functionally arranged.

OPERATION

Operation of the flight data recorder is automatic and requires no action on the part of flight crew members. During operation, recording is accomplished by means of solid state memory. Continuous internal checking of the transcribed data ensures that correct data is being recorded.

The F1000 incorporates hardware and software built-in tests (BIT). The BIT routines are performed at power-up and continuously during the operation of the recorder. Upon detection of an error or fault, the FDR (depending on the severity of the fault) illuminates the FDR FAIL annunciator and/or tags the flight data with a discrete fault bit. Additionally, a fault dependent hexadecimal code is logged into the nonvolatile memory of the FDR upon the event of a fault. The hexadecimal code is translated into a status message and can be polled by ground support equipment for analysis.

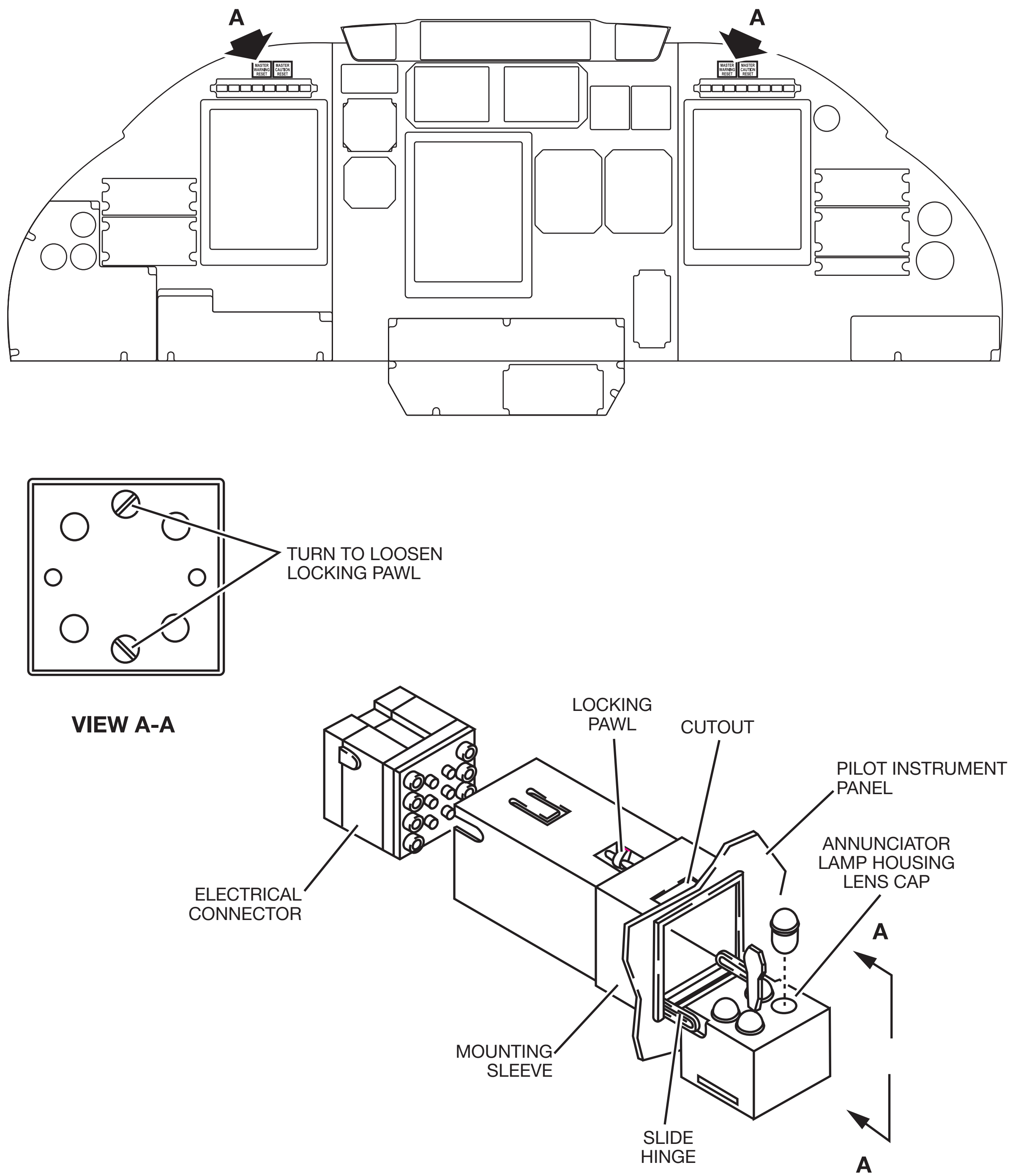


Figure 31-7. Aerospace Optics Switches

Extreme G-forces, as encountered in an aircraft mishap, cause the impact switch to open, This removes power from the FDR system. The impact switch, by removing power from the system, prevents the recorder from continuing to run and recording nothing, which would eventually erase all information previously recorded.

AEROSPACE OPTICS SWITCHES (SWITCHLIGHTS)

There are Aerospace Optics switches in the left instrument panel, center instrument panel, and right instrument panel (Figure 31-7). The number of Aerospace Optics switches is determined by the navigation equipment in the aircraft.

Some of the Aerospace Optics switches in the instrument panel function as indicators and not switches.

DIAGNOSTICS

Removal/installation of the lamp(s) is accomplished while the Aerospace Optics switches are installed. The removal/installation procedures are same regardless of location.

Removal

1. Disengage the applicable circuit breaker for Aerospace Optics switches being removed.

2. Using the extraction tool disconnect the connector from Aerospace Optics switch.

3. Open the annunciator lamp housing/ lens cap and push the slide hinge away from the annunciator, removing the annunciator to gain access to locking pawl.

4. Unscrew the locking pawl and remove the mounting sleeve.

5. Remove the Aerospace Optics switch from the instrument panel.

Installation

1. Position the Aerospace Optics switch instrument panel.

2. Slide the mounting sleeve on the Aerospace Optics switch. Ensure that cutout on mounting sleeve faces aft.

3. Tighten the locking pawl on the mounting sleeve.

4. Install the annunciator lamp housing/ lens cap in the slide hinge.

5. Close the annunciator lamp housing/ lens cap and install the electrical connector.

6. Engage the applicable circuit breaker and perform an operational test of the affected system.

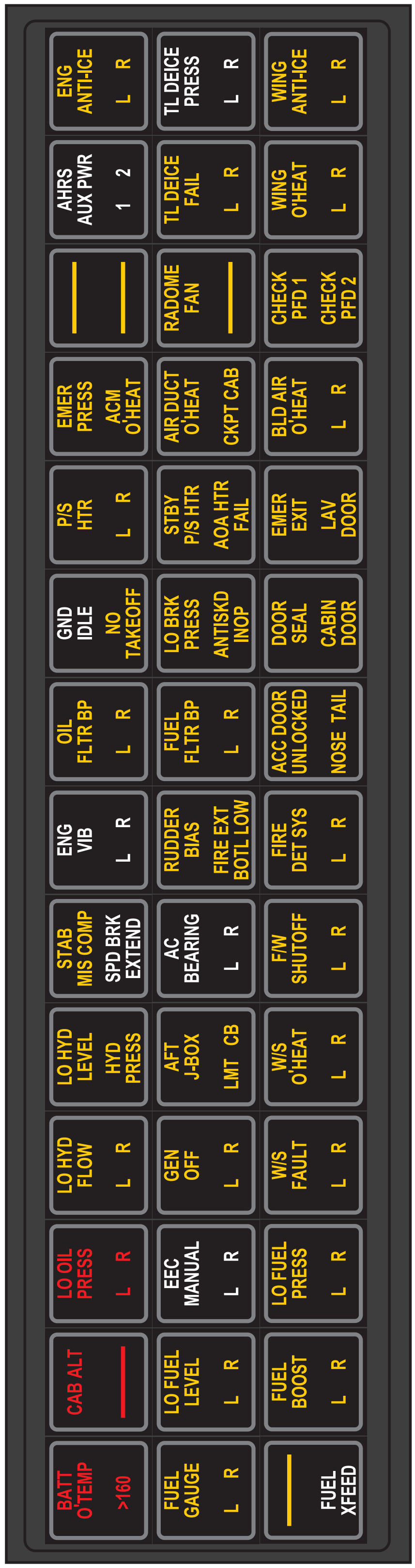


Figure 31-8. Annunciator Panels

MASTER WARNING LIGHTS AND ANNUNCIATORS

The MASTER WARNING lights and annunciators provide a visual indication to the pilot of certain conditions and/or functions of selected systems (Figure 31-8 and Table 31-1).

The annunciator panel is in the fire tray and contains a cluster of caution/warning lights with selected color lenses and legend plates arranged according to aircraft systems. The annunciators operate in conjunction with the MASTER WARNING lights on the pilot instrument panel and on the copilot instrument panel.

A rotary TEST knob is on the pedestal to verify the integrity of the MASTER WARNING and annunciator lamp filaments.

OPERATION

Each annunciator segment has a legend that illuminates to indicate an individual system fault. Red lights indicate a warning malfunction that requires immediate corrective action. Amber lights indicate a caution malfunction requires immediate attention, but not necessarily action. White lights indicate a system function has been accomplished. The MASTER WARNING lights illuminate simultaneously with red annunciators alone. Both amber generator annunciators illuminate to alert the operator of the system fault on the annunciator panel. The MASTER WARNING light incorporates a reset switch, which is actuated by pushing in on the warning light lens. The annunciator, when actuated, turns off (resets) the MASTER WARNING light, making the system available to alert the operator if any other system fault occurs. The MASTER WARNING light stays illuminated until reset, even if the malfunction that caused the light to illuminate has been corrected. The annunciator remains on until the system fault has been corrected.

NOTES

Table 31-1. ANNUNCIATOR PANELS

LEGEND	DESCRIPTION
BATT O'TEMP >160	The BATT O'TEMP (battery overtemperature) light illuminates if the temperature of the battery exceeds 145°F. If the temperature increases above 160°F the >160°F portion of the light will also illuminate. This system operates from a different sensor than the battery temp gauge.
CAB ALT	The CAB ALT (cabin altitude) light illuminates to warn of cabin altitude in excess of 10,000 ft. If the pressure controller detects operation out of a high altitude airport, the light flashes at14,500 ft. cabin altitude.
LO OIL PRESS L R	The LO OIL PRESS light illuminates to indicate oil pressure when pressure in the specified engine is below 20 psid.
LO HYD FLOW L R	The LO HYD FLOW light illuminates to indicate that hydraulic fluid flow rate is below normal. Light will flash after 5 seconds illumination. Could indicate pump failure.
LO HYD LEVEL HYD PRESS	The LO HYD LEVEL light illuminates to indicate low fluid quantity in the hydraulic reservoir. (<75ci.) The HYD PRESS light illuminates to indicate that the hydraulic loading valve has closed, pressurizing the hydraulic system. (If on for 40 seconds in flight, it will flash).
STAB MIS COMP SPD BRK EXTEND	The STAB MISCOMP light illuminates to indicate the horizontal stab position does not agree with the flap handle position after 30 seconds of travel. (Unless the Landing Gear is also selected, then the delay is 40 seconds). The SPDBRK EXTEND light illuminates to indicate that both speedbrakes are fully extended.
ENG VIB L R	The ENG VIB light illuminates to indicate vibration has been detected in the designated engine.
OIL FLTR BP L R	The OIL FLTR BP light illuminates to indicate an impending bypass of the engine oil filter. Illuminates flashing, changes to steady after pressing the MASTER CAUTION.
GND IDLE NO TAKEOFF	The GND IDLE light illuminates when the aircraft is on the ground and the ECC's are in auto. The NO TAKEOFF light illuminates steady when the horizontal stab, wing flaps, speed-brakes or elevator trim are not in proper position for takeoff. Light flashes and MASTER CAUTION illuminates if throttles are advanced beyond 54° TLA.
P/S HTR L R	The P/S HTR light illuminates to indicate loss of DC power to the left or right pitot static heat system. Illuminates flashing, changes to steady after pressing the MASTER CAUTION (in flight).
EMER PRESS ACM O'HEAT	The EMER PRESS light illuminates to indicate that the Emergency Pressurization has been activated, manually or automatically. The ACM O'HEAT light illuminates to indicate that the ACM has overheated through a 420°F switchand shutdown.

Table 31-1. ANNUNCIATOR PANELS (Cont)

LEGEND	DESCRIPTION
AP PITCH MISTRIM	Annunciator flashes to indicate the autopilot elevator servo is not trimmed properly, (excessive sustained pressure). UP or DN light will illuminate on the A/P controller. Activates MASTER CAUTION lights. (XLS AP PITCH MISTRIM annunciations appear in the PFDs.)
AP ROLL MISTRIM	Annunciator flashes to indicate the autopilot aileron servo is not trimmed properly (excessive sustained pressure). Activates MASTER CAUTION lights. (XLS AP ROLL MISTRIM annunciations appear in the PFDs.)
AHRS AUX PWR 1 2	The AHRS AUX PWR light illuminates to indicate the AHS is currently powered by the auxiliary battery due to loss of normal DC power.
ENG ANTI-ICE L R	The ENG ANTI-ICE light illuminates to indicate the engine nacelle lip temperature is too low (60°F) or a fault with the stator valve has been detected.
FUEL GAUGE L R	The FUEL GAUGE light illuminates to indicate a fault detected in the fuel gauging system. Check BITE lights prior to turning off the battery switch during shutdown.
LO FUEL LEVEL L R	The LO FUEL LEVEL light illuminates to indicate the fuel quantity remaining in the respective wing has droped to less than 360 +/– 20 lbs for 30 seconds.
EEC MANUAL L R	The EEC MANUAL light illuminates to indicate that an engine electronic computer is offline and the engine is operating in manual mode.
GEN OFF L R	The GEN OFF light illuminates to indicate the respective generator power relay is open and the generator is offline. Both lights on will trigger the MASTER WARNING flasher.
AFT J-BOX LMT CB	The AFT J-BOX LMT light illuminates to indicate an open 225 amp current limiter in the aft J-Box. The AFT J-BOX CB light illuminates to indicate a popped start control CB in the aft J-Box.
AC BEARING L R	The AC BEARING light illuminates to indicate a primary bearing failure in the AC alternator.
RUDDER BIAS FIRE EXT BOTL LOW	The RUDDER BIAS light illuminates to indicate the rudder bias control valve and the commanded position do not agree after a one second delay. The FIRE EXT BTL LOW light illuminates to indicate low pressure of 500 psi or less in one or both fire extinguisher bottles.

Table 31-1. ANNUNCIATOR PANELS (Cont)

LEGEND	DESCRIPTION
FUEL FLTR BP L R	The FUEL FLTR BP light illuminates to indicate an impending bypass of the respective fuel filter.
LO BRK PRESS ANTISKD INOP	The LO BRK PRESS light illuminates to indicate loss of power brake pressure below 750 psi. The ANTI SKD INOP light illuminates to indicate that the anti-skid system is inoperative in test mode, or the switch is off. It will also illuminate along with the LO BRK PRESS annunciator.
STBY P/S HTR AOA HTR FAIL	The STBY P/S HTR light illuminates to indicate loss of DC power to the standby pitot static heat system. The AOA HTR FAIL light illuminates to indicate loss of DC power to the angle of attack probe heater.
AIR DUCT O'HEAT CKPT CAB	The AIR DUCT O'HEAT light illuminates to indicate the supply air duct to the cabin or cockpit has exceeded 300°F.
RADOME FAN FDR FAIL	The RADOME FAN light illuminates to warn of failure of the radome cooling fan. FDR FAIL Advisory—Indicates the optional flight data recorder is inoperative (not used on XLS).
TL DEICE FAIL L R	The TL DEICE FAIL light illuminates to indicate the respective horizontal stab deice boot is not inflating properly less than 16 psi.
TL DEICE PRESS L R	The TL DEICE PRESS light illuminates to indicate the respective horizontal stab deice boot is inflating and pressure is greater than 16 psi.
FUEL XFEED	The FUEL XFEED light illuminates to indicate the fuel crossfeed valve has opened. The light will flash if crossfeed switch is in off and the crossfeed valve is still open for greater than 10 seconds. Does not illuminate if crossfeed valve does not fully open when crossfeed is selected.
FUEL BOOST L R	The FUEL BOOST light illuminates to indicate the activation of the electric boost pump. (automatic or manual activation)
LO FUEL PRESS L R	The LO FUEL PRESS light illuminates to indicate the fuel pressure in the engine supply line is low below 5.3 psi.
W/S FAULT L R	The W/S FAULT light illuminates to indicate the detection of a fault in the windshield anti-ice system.

Table 31-1 ANNUNCIATOR PANELS (Cont)

LEGEND	DESCRIPTION
W/S O'HEAT L R	The W/S O'HEAT light illuminates to indicate an overheat condition (>140°F) of the windshield.
F/W SHUTOFF L R	The F/W SHUTOFF light illuminates to indicate the respective fuel & hydraulic firewall shutoff valves are both closed.
FIRE DET SYS L R	The FIRE DET SYS light illuminates to indicate a failure in the respective fire detection system.
ACC DOOR UNLOCKED NOSE TAIL	The ACC DOOR UNLOCKED NOSE light illuminates to indicate that at least one of four nose avionics door latches is not secure. The ACC DOOR UNLOCKED TAIL light illuminates to indicate either the forward tail cone access door, the baggage compartment door, or the batery door is not secure.
DOOR SEAL CABIN DOOR	The DOOR SEAL light illuminates to warn of pressure less than 5.5 psid in the cabin door seal. The CABIN DOOR light illuminates to indicate that the cabin door is not locked properly and/or the vent door did not close.
EMER EXIT LAV DOOR	The EMER EXIT light illuminates to warn the emergency exit door is open. The LAV DOOR light illuminates to indicate that the interior lavatory door is not latched open with flaps down.
BLD AIR O'HEAT L R	The BLD AIR O'HEAT light illuminates to indicate the respective bleed air system has exceeded 560°F.
CHECK PFD 1 CHECK PFD 2	The CHECK PFD 1 light illuminates to indicate the pilot flight display system is not operating properly. The CHECK PFD 2 light illuminates to indicate the copilot flight display system is not operating properly.
WING O'HEAT L R	The WING O'HEAT light illuminates to indicate the air temperature between the wing leading edge heatshield and the wing forward spar has exceeded 160°F.
WING ANTI-ICE L R	The WING ANTI-ICE light illuminates to indicate the wing anti-ice bleed air temperature is too low below 220°F.

Table 31-1 ANNUNCIATOR PANELS (Cont)

LEGEND	DESCRIPTION
AP OFF YD OFF	(XL only) Illumination occurs when the autopilot or yaw damper is manually disconnected by the crew or automatically disconnected due to malfunction. This annunciator is next to the L and R MASTER WARNING/MASTER CAUTION switchlights. XLS—AP and YD OFF annunciations appear in the L and R PFDs.
APU GEN OFF	(XLS only) Steady illumination indicates the APU is operating and its generator is off line.
BIAS HEATER FAIL	(XL and XLS) Steady illumination indicates the rudder bias heating blanket is heating. Flashing light indicates blanket sensor failure. Pressing the light causes steady illumination. This annunciator does not activate the MASTER CAUTION lights.
FD/AP PFD 1 FD/AP PFD 2	(XL and XLS) Switchlight indicates the No.1 or 2 flight director is controlling the autopilot. Press the switchlight to change flight directors. Switching flight directors with the autopilot engaged causes the autopilot to revert to basic pitch and heading hold modes. The flight director modes must be reselected.
TERR NORM	(XL and XLS) Switchlight indicates the enhanced GPWS or TAWS warnings occur normally and the terrain map is displayed on the MFD.
TERR INHIB	(XL and XLS) When selected, inhibits the enhanced TAWS (EGPWS) warnings and the terrain map. Modes 1–7 remain active.
TAWS FLAP NORM	(XLS) Switchlight indicates that the TOO LOW FLAPS audio warning activates when the aircraft is below approximately 245 feet AGL, less than 160 KIAS, and landing flaps are not selected.
TAWS FLAP OVRD	(XL and XLS) When pressed, the switch disarms or cancels the audio warning for landing with flaps less than 35°. The XL switchlight is labeled GPWS FLAP NORM and GPWS O'RIDE. The functions are the same.
TAWS G/S	(XLS) Switchlight indicates normal GLIDESLOPE audio warnings are active for deviations below the glideslope. The GLIDESLOPE warning sounds if the aircraft is below 1000 feet AGL, descending greater than 500 fpm, and below 1.3 dots.
CANCELED	(XLS) When pressed, disables the GLIDESLOPE audio warnings. The XL switchlight is labeled GPWS G/S and O'RIDE. The functions are the same.
TAWS TEST	(XLS) Pressing the switchlight initiates the TAWS system test. This test function is inhibited inflight. The XL switchlight is labeled GPWS TEST. The functions are the same.

Table 31-1 ANNUNCIATOR PANELS (Cont)

LEGEND	DESCRIPTION
AUDIO SPK/HPH	(XL and XLS) Indicates normal operating mode (default position). Audio communications are active through the cockpit speakers and crew headsets.
AUDIO HPH ONLY	Pressing the switchlight mutes all avionics audio through the cockpit speakers including TCAS and TAWS (EGPWS). The gear horn and NO TAKEOFF warnings are not inhibited.
PHONE CALL	(XL and XLS) (Optional) Steady illumination for an incoming HF radio call.
CABIN TEMP CTL RMT NRM	(XL and XLS) Indicates that cabin temerature is controlled from the cockpit temerature controller. When pressed, transfers the cabin temperature control to the cabin.
ARM	Illumination indicates pressure is available to the thrust reverser (pressure is sensed passed the isolation valve). Illumination is normal on ground during TR operation, but abnormal inflight. Illumination inflight causes the red MASTER WARNING lights to flash.
UNLOCK	Illumination indicates the thrust reverser is unlocked. Illumination is normal on ground during TR operation, but abnormal inflight. Illumination inflight causes the red MASTER WARNING lights to flash.
DEPLOY	Illumination of the white light indicates the thrust reverser is deployed. Illumination is normal on ground during TR operation, but abnormal inflight.
LH ENGINE FIRE / RH ENGINE FIRE	Illumination indicates high temperature is detected in the engine nacelle. 1. Closes the fuel F/W shutoff valve. 2. Closes the hydraulic F/W shutoff valve. 3. Deactives the engine generator (opens the field relay). 4. Disarms the thrust reverser. 5. Arms the engine fire bottles.
BOTTLE 1 ARMED PUSH	Illumination of the white light indicates the respective engine fire bottle is armed. When pressed, the bottle discharges. The red ENGINE FIRE switchlight must be pressed to illuminate the BOTTLE ARMED lights.
APU FIRE	Illumination indicates high temperature in the APU compartment. The APU automatically shuts down and the APU FAIL light illuminates. Pressing the red switchlight discharges the APU fire bottle. If the switchlight is not pressed, the fire bottle automatically discharges in 8 seconds.
APU RELAY ENGAGED	Illumination indicates the APU relay is engaged during the APU start. Illumination also occurs when the APU generator participates in an engine start.
APU FAIL	Illumination indicates the APU will not start due to a system malfunction (i.e., the APU fire bottle is low or the fire detection system is inoperative). If the APU is operating, the light indicates the APU is shutting down. Reasons for automatic shutdown include fire detected in the APU compartment or the fire bottle is low. Limitation: Starting the APU is prohibited whenever the APU FAIL light is illuminated.
READY TO LOAD	Illumination indicates the APU start is complete and at operating speed (95% rpm + 4 seconds). The APU generator and bleed air can be selected after illumination. The light remains illuminated during APU operation.
BLEED VAL OPEN	Illumination indicates APU bleed air valve (BAV) is other than closed.

31 INDICATING AND RECORDING SYSTEMS

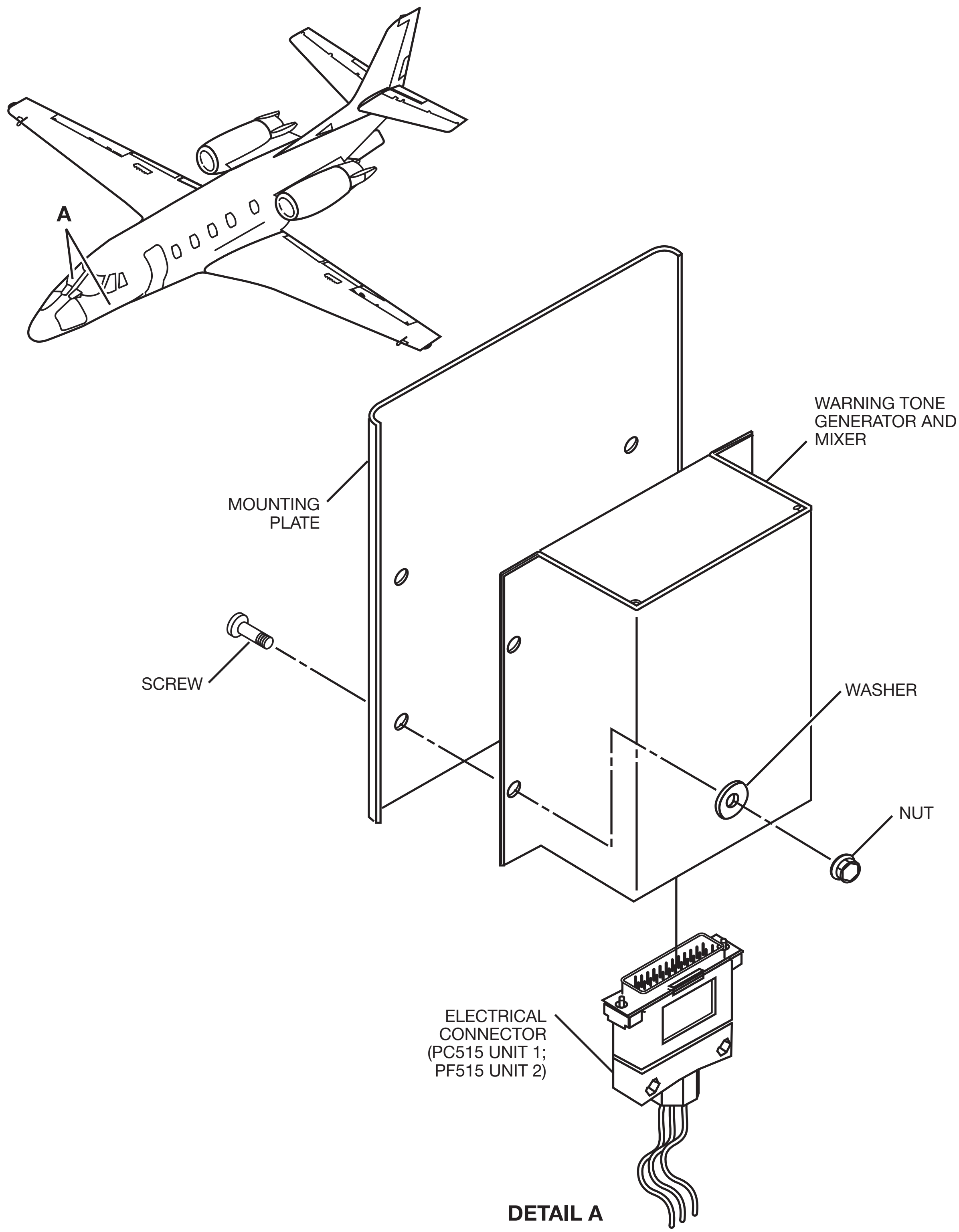


Figure 31-9. Aural Warning System

AURAL WARNING SYSTEM

DIAGNOSTICS

This section provides procedures for removal and installation of the aural warning system warning tone generator and mixer boxes in the pilot (warning tone generator and mixer 1) and copilot (warning tone generator and mixer 2) side consoles (Figure 31-9).

Removal

1. Disengage the WARN AUDIO 1 or WARN AUDIO 2 circuit breaker on the right CB panel.
2. Remove the pilot or copilot seat as required for access. Refer to Chapter 25—"Flight Crew Seats–Maintenance Practices" in the *AMM*.
3. Remove the side console access panel 245CL or 246CR to access the generator and mixer. Refer to Chapter 6—"Access Plates and Panel Identification—Description and Operation" of the *AMM*.
4. Disconnect electrical connector (PC515 or PF515).
5. Remove screws and washers that secure the generator and mixer to the mounting plate. Remove the generator and mixer from the aircraft.

Installation

1. Position the generator and mixer on mounting plate and secure with screws and washers.
2. Connect electrical connector (PC515 or PF515).
3. Install side console access panel.
4. Engage the WARN AUDIO 1 or WARN AUDIO 2 circuit breaker on the right CB panel.
5. Install the pilot or copilot seat as required.

NOTES

Figure 31-10. MFD Locations

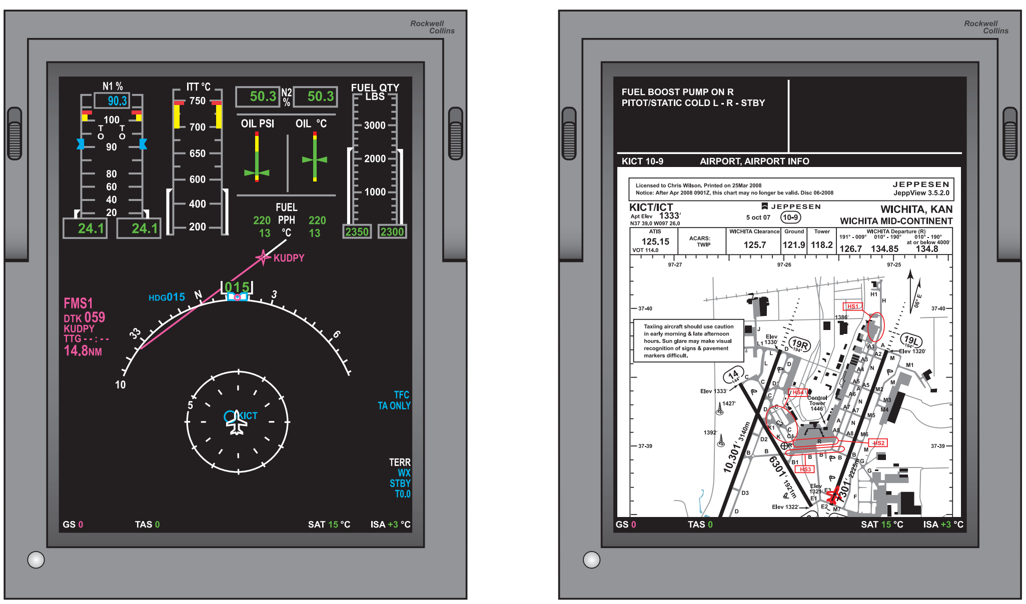


Figure 31-11. EICAS Display with Avionics Turned ON

31 INDICATING AND RECORDING SYSTEMS

ENGINE INDICATION AND CREW ALERTING SYSTEM—AIRCRAFT 6001 AND SUBSEQUENT

The engine indication and crew alerting system (EICAS) gives the flight crew primary engine operating parameters, and monitoring of the aircraft systems. The EICAS system is divided into two primary functions, the engine indicating system (EIS) and the crew alerting system (CAS). During usual operation the EIS and CAS are displayed at the top of the MFDs, with the EIS on the left MFD and the CAS on the right MFD (Figure 31-10). The CAS can also be selected on the PFDs instead of the HSI for dual PFD reversion.

An engine start page, with EIS and CAS windows, is displayed on the pilot MFD for engine starts. The page goes back to usual operation when the ELECTRICAL BATT ON/OFF switchlight is in the ON position and the ELECTRICAL AVIONICS ON/ OFF switchlight is in the ON position (Figures 31-11 and 31-12).

The source of the EICAS messages comes from the data concentrator unit (DCU). The DCU receives discrete signal inputs, serial inputs, and analog inputs from the different systems on the aircraft.

CAS MESSAGES

The CAS is used to show advisory messages, conditions, warning messages, caution messages, system failures, and procedure status messages. The CAS messages are triggered by signals or groups of signals sent by the DCU or the full-authority digital engine-control (FADEC).

The CAS messages are shown at the top of the right MFD, if the right MFD fails the CAS will be moved to the left MFD, and are stacked by color. CAS messages are classified as WARNING, CAUTION, or ADVISORY and are displayed in priority order and operate in

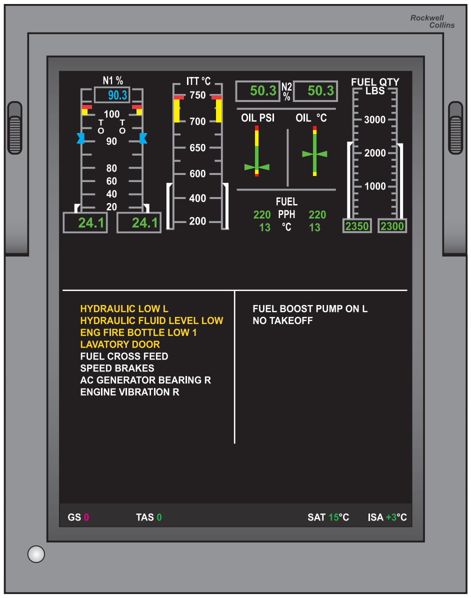


Figure 31-12. EICAS Display with Avionics Turned OFF

conjunction with the MASTER WARNING RESET and MASTER CAUTION RESET switchlights.

The master warning system provides visual indications to the flight crew of the following:

- Unsafe operating conditions requiring immediate attention
- Crew advisory warnings that require attention, but not necessarily immediate action
- Advisory indications that some specific system(s) are in operation

New CAS messages are always added to the top of their color area. Red (warning) messages show on the top and are accompanied by a voice message, amber (caution) messages show in the middle, and white (advisory) messages show on the bottom.

The CAS messages that have the characters "L–R" are related to the systems that are divided into left and right subsystems. Different signals will trigger the messages for the different sides and the "L–R" section of the CAS message will show the related side.

The CAS summary format is used in the case of dual MFD failure and is selected on the pilot or copilot PFD using the PFD menu if there are no warning messages present. If a new caution or warning message appears the CAS summary format will pop-up on the lower screen of the right PFD. If the highest priority CAS message is a caution, the CAS stack may be removed and a CAS MSG annunciation will appear on both PFDs. The CAS MSG color will be the same as the highest priority CAS message on the stack.

The CAS messages are put in order by importance (warnings, cautions, and advisories) and then by the order of when it shows (the most current messages on the top). For a list of the CAS messages and their applicable colors, refer to Tables 31-4 through 31-6. The warning messages are shown in red and show that it is necessary for the flight crew to identify the problem immediately. These messages will also trigger the master warning switch lights to come on and are accompanied by a voice message or tone.

The caution messages are shown in amber and show that steps to correct a problem may be necessary. These messages will also trigger the master caution switch lights to come on.

The advisory messages are shown in white as show in Figure 31-13.

Some of the CAS messages can be more than one color. The same message cannot show as two colors at the same time during usual conditions. If signals are received for the two color conditions, the more important color is used. When the message changes to a more important color (white to amber or amber to red), the message will flash and a response is necessary. When the message changes to a less important color (red to amber or amber to white), the message is added to the top of the applicable list and a response is not necessary. The message will not flash and will not trigger the master caution/warning switch lights to come on.

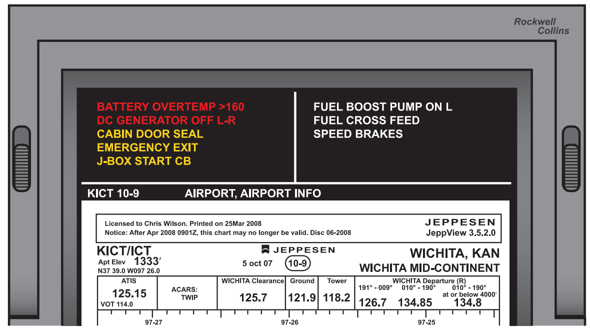


Figure 31-13. CAS Message Displayed on MFD 2

The CAS button on the cursor control panel (CCP) can be used to view all of the pages of CAS messages if there is more than one. Amber and white messages can go out of view on the display. If this occurs, use the CAS button on the CCP to access the next page of messages.

Inhibits are used to prevent some CAS messages from showing during different conditions. Takeoff operation phase inhibit (TOPI) and landing operation phase inhibit (LOPI) are used to decrease the quantity of work for the flight crew during takeoff and landing. At different times, a bus or a processor failure will trigger an inhibit to prevent incorrect data from causing a message to show (Figure 31-14).

The CAS messages that the inhibits effect cannot come into effect or change to a more important color while the inhibit is on. If the messages were on before, they will continue to show correctly. The functions to accept the messages still operates correctly.

With the systems that have left side and right side messages, one of the two sides can be inhibited while the other side continues to operate correctly.

OIL PRESSURE LOW L-R			
Color	**Inhibited By**		**Debounce**
Red	LOPI	TOPI	Standard
	ESDI	EFI	
	SIPI		

This message is displayed when the engine oil pressure is low. Each engine has a pressure switch. When the oil pressure drops below 20 PSI, the switch sends a ground signal to the EICAS, which posts the message. Due to the hysteresis, when the oil pressure increases above 35 PSI, the switch sends an open signal to the EICAS, which removes the message.

Figure 31-14. Sample CAS Message Inhibits

There are six CAS inhibit modes. They are:

- LOPI
- TOPI
- SIPI
- Engine shutdown inhibit (ESDI)
- Engine fail inhibit (EFI)
- Emergency power inhibit (EMER)
- Ground/air inhibit (ON GROUND/ IN AIR)

EMER—CAS messages are not displayed with emergency power on. The following voice and tone alerts are still heard:

- TAWS Warning and Caution Aurals
- TCAS Warning and Caution Aurals
- Autopilot Disconnect tone
- Overspeed tone
- Stall Warn tone
- SELCAL tone

TOPI—This decreases the quantity of work for the flight crew during takeoff. TOPI comes on when one of the following three conditions occurs:

- The aircraft changes from and on-ground to in-flight condition
- One of the two indicated airspeeds changes from less than 80 knots to more than 80 knots
- Then N_1 indication is greater than 70%

TOPI will go off when one of the following three conditions occurs:

- The aircraft has been airborne for more than 30 seconds
- The radio altitude is more than 400 feet (121.9m) above ground level
- One of the two airspeed indications is less than 80 knots

LOPI—This decreases the quantity of work for the flight crew during landing. LOPI comes on when one of the following two conditions occurs:

- The aircraft changes from an in-flight to an on-ground condition
- The radio altitude is less than 400 feet (121.9m) AGL

LOPI will go off when one of the following three conditions occurs:

- The aircraft has been on the ground for more than 30 seconds
- One of the two indicated airspeeds is less than 40 knots
- The radio altitude is more than 400 feet (121.9m) AGL

SIPI—Prevents CAS message from showing during the engine start cycle.

ESDI—This prevents CAS messages from showing during an engine shutdown. ESDI is triggered by the FADEC.

EFI—This prevents CAS messages from showing during an engine failure. EFI is triggered by the FADEC.

GROUND/AIR—CAS messages with the ON GROUND inhibit will not show while the aircraft is on the ground. Messages with the IN AIR inhibit will only show while the aircraft is on the ground.

CAS messages are shown a short period of time after they are triggered. The function of this debounce (interval) time is for sensors when they are not stable. The standard minimum debounce time is 200 milliseconds and the maximum debounce time is 399 milliseconds.

DCU

The DCU operates in real time. It receives, puts together and transmits analog, discrete and serial bus data.

The DCU has the functions that follow:

- Shows aircraft system data to the flight crew on the MDSs
- Supplies functions through interfaces with the flight data recorder (FDR), the maintenance diagnostic computer (MDC), and with the systems that have an interface with EICAS
- Gives the radio interface unit (RIU) instructions to give aural warnings to the flight crew. These aural warnings can be tones or voices

The DCU is installed, in a mounting tray with two hold down clamps, in the right nose avionics compartment.

AUDIO WARNING SYSTEM

Various audio warnings are incorporated into the aircraft systems that warn of specific conditions and malfunctions.

Nearly all red CAS messages are also accompanied by aural voice alerts that announce the text of the CAS message displayed. There is no aural voice alert associated with the red EMERGENCY DESCENT CAS message.

Two or more voice aurals cannot play simultaneously. Table 31-2 illustrates the order of priority of the various voice alerts. If two or more voice aurals are trying to play, the one with the highest priority sounds until:

- Acknowledged via the MASTER WARNING RESET switchlight
- Voice aural with a higher priority becomes active
- Associated condition that caused the voice aural to announce is resolved

If a voice aural is currently announcing and a higher priority voice aural becomes active, the lower priority voice aural finishes announcing before the higher priority voice aural begins announcing. The terrain awareness and warning system (TAWS) and traffic alert and collision avoidance system (TCAS) aurals are generated by the respective system unit. When the TAWS or TCAS voice aurals become active while a lesser priority is playing, the aural warning system immediately stops announcing the lower priority voice aural and immediately begins announcing the TCAS or TAWS aural.

When any amber CAS message displays, the master caution attention chime sounds.

Use the rotary TEST knob to test the audio system and various other warning systems.

Table 31-3 describes the associated system audio and CAS message functions for the different TEST knob positions.

Table 31-2. AURAL WARNINGS

PRIORITY	RED CAS MESSAGES VOICE AURALS
1	TAWS WARNING AND CAUTION AURALS
2	TCAS WARNING AND CAUTION AURALS
3	LEFT AND RIGHT ENGINE FIRE
4	LEFT ENGINE FIRE
5	RIGHT ENGINE FIRE
6	LEFT AND RIGHT ENGINE FAIL
7	LEFT ENGINE FAIL
8	RIGHT ENGINE FAIL
9	APU FIRE
10	BAGGAGE SMOKE DETECT
11	LAVATORY SMOKE DETECT
12	CABIN ALTITUDE
13	DC GENERATORS OFF
14	BATTERY OVERTEMP
15	LEFT AND RIGHT OIL PRESSURE LOW
16	LEFT OIL PRESSURE LOW
17	RIGHT OIL PRESSURE LOW
18	NO TAKEOFF (NOTE 2)
19	MASTER CAUTION
PRIORITY (NOTE)	**TONE AURALS**
1	AUTOPILOT DISCONNECT (NOTE 1)
2	ALTITUDE ALERTS (NOTE 2)
3	LANDING GEAR (NOTE 3)
4	OVERSPEED (NOTE 2)
5	STALL WARN
6	FMS VTA
7	SELCAL
8	PHONE CALL

NOTE 1:
The AP disconnect horn is canceled by any of the following means:
- AP disconnect yoke switch
- Manual trim yoke switch
- Go-around switch

NOTE 2:
Canceled when condition is corrected

NOTE 3:
Per landing gear horn logic contained in the gear monitor PCB

Table 31-3. TEST INDICATIONS

ROTARY TEST POSITION	AURAL	CAS MESSAGE(S)	NOTES
FIRE WARN	Left and right engine fire, *(Baggage Smoke Detect), *(Lavatory Smoke Detect) voice aurals	ENGINE FIRE L–R *(BAGGAGE SMOKE DETECT) *(LAVATORY SMOKE DETECT)	This position illuminates either ENG FIRE switchlight, BOTTLE ARMED PUSH switchlights, and MASTER WARNING RESET switchlights *(Voice, aural and CAS messages only activate if system is installed. If baggage and lavatory smoke detect systems are both installed, DCU plays voice aural by priority).
LANDING GEAR	Gear warn tone	None	This position provides a test signal to the landing gear PCB to illuminate all three green down/lock lights and red unlock light on the gear handle. The gear warning tone signal is also triggered from the PCB to the DCU.
BATT TEMP	Battery overtemp voice aural	BATTERY OVERTEMP >160	This position swings the battery temperature indicator needle to 160°F. The MASTER WARNING RESET switchlights also illuminate along with the CAS message.
STICK SHAKER	Stall warning tone	None	This position tests the AOA computer, and the computer activates the pilot and copilot stick shaker motors, flash the AOA red indexer light, and moves the AOA pointer to the top of the AOA scale on the PFD. The stall warning tone signal to the DCU also comes from the computer.
THRUST REV	None	None	This position illuminates all T/R lights, ARM, UNLOCK, and DEPLOY, in the firetray and MASTER WARNING RESET switchlights.
W/S TEMP	Caution tone	WINDSHIELD HEAT INOP L–R WINDSHIELD OVERTEMP L–R	This position tests the W/S controller. With engine running, CAS messages and M/C switchlights illuminate for 3 to 4 seconds, and then extinguish, unless there is a W/S controller failure or a sensor failure. Then the CAS messages stay on. With engines shutdown, WINDSHIELD HEAT INOP illuminates and remains on due to alternator not supplying power to the W/S controller.
OVERSPEED	Overspeed tone	None	This position provides a test signal to the DCP to trigger the tone. AVIONICS switchlight must also be ON to play the tone.
ANTISKID	Caution tone	ANTISKID FAIL	This position tests the skid control unit. The unit provides a fail signal to the DCU and triggers the CAS message and MASTER CAUTION RESET switchlights for 3 to 4 seconds. If there is a unit failure, then the CAS message stays illuminated.
ANNUNCIATOR	None	None	This position tells the DCU to illuminate the MASTER WARNING RESET and MASTER CAUTION RESET switchlights. All AOA indexer lights illuminate.
AVIONICS	None	None	This position tests all TAWS, except TAWS TEST, lighted switches. All green audio panel transmit select lights illuminate.
SPARE	None	None	Nothing should come on at this position.
OFF	None	None	Nothing should come on at this position.

TEST SYSTEM

The TEST knob is on the forward portion of the center pedestal directly below the copilot flight management system (FMS) keypad and above and to the right of the throttle quadrant (Figure 31-15). The knob offers several positions of test. Complete functionality is attained only when the BATT and AVIONICS switchlights are both ON. A red light above the TEST knob illuminates whenever the TEST knob is in any position but OFF.

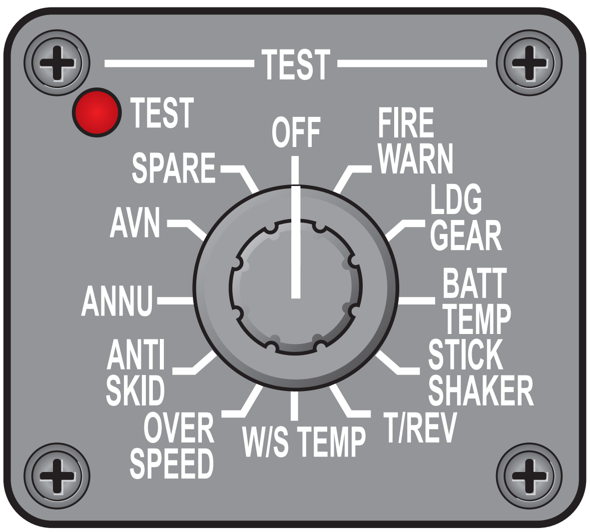


Figure 31-15. Rotary TEST Knob

Refer to Tables 31-4 through 31-6 for a list of the CAS messages and their applicable colors.

NOTES

Table 31-4. RED EICAS MESSAGES

APU FIRE

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard

This message is displayed when a fire is detected in the APU by a fire loop. 28 Volts on the input to EICAS means a fire has been detected, which causes the message to be displayed. Open circuit means a fire has not been detected, which causes the message to be removed. A voice aural is also triggered with this message.

BAGGAGE SMOKE DETECT

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard

This message is displayed when smoke is detected in the baggage compartment. When smoke is present, a smoke detector sends a ground signal to the EICAS system, which posts the message. When there is no smoke, the signal is an open, and the message is removed.

BATTERY OVERTEMP > xxx

Color	Inhibited By		Debounce
Red	LOPI	TOPI	8 Second

"xxx" = 145 or 160
This message is displayed when the battery temperature sensor measures above 145°F or 160°F. This is implemented as 2 messages in the Collins CAS system, one with 145, and the other with 160. However, both messages will not display at the same time. There is an 8 second time delay off for each message. For input characteristics, see Battery Temp Sensor Chart. This CAS message is also accompanied by a "BATTERY OVERTEMP" aural voice alert. The message may also be cross-checked against the Battery Temp gauge on the LH instrument panel.

CABIN ALTITUDE

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard

This message is displayed when the cabin altitude is too high. The CABIN ALTITUDE CAS message comes on at 14,500 ft during high altitude mode and at 10,000 ft for normal operation. When the input is 28V, the message is displayed. When the input is open, the message is not displayed. An associated voice aural is played with this message.

DC GENERATOR OFF L-R

Color	Inhibited By		Debounce
Red	LOPI	TOPI	*Standard
Amber	*ESDI	SIPI	

This message is displayed when the respective generator contactor is open. The EICAS input is connected to the moving bar which connects the contactor input to the output when the contactor is closed. The connection is made through a circuit breaker to limit the current in case of a fault. 28 Volts on the EICAS input means that the contactor is closed. The message is red if both left and right are open. The message is amber if only one is open. This message is also inhibited during engine start.

* The engine shutdown inhibit (ESDI) is not active in the air.

EMERGENCY DESCENT

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard

This message is displayed when FGC sets 429 LABEL 271, Bit 25 = 1. The FGC sets the 429 LABEL when the FGC is configured for emergency descent mode and the EDM Active DCU input senses 28VDC. The EDM Active input sees 28VDC when the cabin altitude exceeds 14,500 ft. This message is the only red CAS warning message without an associated voice aural.

ENGINE FAILED L-R

Color	Inhibited By		Debounce
Red			Standard

This message is posted when the engine has failed. It is posted when FADEC 429 Label 271, bit 18 = 1. This bit shall be active if the engine speed drops below the minimum idling speed and the throttle is not in the cutoff position. The bit shall not be active during engine start procedures. When this message is present, it also inhibits all the messages with the Engine Fail Inhibit. Unless otherwise specified in the message description, the engine inhibits only the respective side message.

(*) = with exceptions

Table 31-4. RED EICAS MESSAGES (Cont)

ENGINE FIRE L-R

Color	Inhibited By		Debounce
Red			Standard
			1 Second

This message is displayed when the engine fire detection system has detected a fire. The fire detect system is a continuous link, temperature sensitive pneumatic detector system as defined by Cessna SCD 9912036. The overall average detect setting is 445°F with a discrete setting of 626°F. The detect element (P/N 9912036-11) is a single loop routed throughout the nacelle to sense the AGB, fuel, and bleed line areas as defined by Cessna drawing 6654300: Fire Detect Instl. An integrity monitor is built into the fire detection responder assembly. The integrity monitor is in the form of a current carrying conductor. If the fire detect loop is shorted the ENGINE FIRE CAS message is generated. If the fire detect loop is severed, the design is such that both ends of the loop continue to function.

LAVATORY SMOKE DETECT

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard

This message is displayed when smoke is detected in the lavatory. When smoke is present, a smoke detector sends a ground signal to the EICAS system, which posts the message. When there is no smoke, the signal is an open, and the message is removed.

NO TAKEOFF

Color	Inhibited By		Debounce
Red	LOPI	In Air	Standard
White			

On the ground, the white NO TAKEOFF message will illuminate if one or more of the following conditions exist:

- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position
- **Speed Brakes** are out of takeoff position

As the throttles are advanced beyond 43° TLA, airspeed less than 67 knots, and thrust reversers not deployed, the red NO TAKEOFF message will illuminate if one or more the following conditions exist:

- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position

The red message also produces a voice aural “No Takeoff”.

The EICAS system receives 2 Ground/Open inputs. If the No Takeoff input is ground, the message is displayed with the color being determined by the No Takeoff w/MW input. If the No Takeoff input is open, the message is not displayed and the No Takeoff w/MW input has no effect. If No Takeoff w/MW input is ground, the color is red, otherwise it is white.

OIL PRESSURE LOW L-R

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard
	ESDI	EFI	
	SIPI		

This message is displayed when the engine oil pressure is low. Each engine has a pressure switch. When the oil pressure drops below 20 PSI, the switch sends a ground signal to the EICAS, which posts the message. Due to the hysteresis, when the oil pressure increases above 35 PSI, the switch sends an open signal to the EICAS, which removes the message.

Table 31-5. AMBER EICAS MESSAGES

ACM OVERTEMP

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the ACM has overheated. When the ACM is too hot, a 28V signal is sent to the EICAS, which posts the message. When the ACM is normal temperature, an open signal is sent to the EICAS, which removes the message.

ADC SSEC MISCOMPARE

Color	Inhibited By		Debounce
Amber		TOPI	10 Second

The message is displayed when the pilot and copilot ADCs are on different SSECs. The ADCs use different SSEC for gear down and gear up. The ADCs automatically switch to the gear up SSEC at 28.5K ft. The SSEC is selected by the nose gear downlock switch on the LH ADC and the LH main downlock on the RH ADC. If altitude differs enough between ADCs, this could also trip an altitude comparator monitor.

AFT BAGGAGE DOOR

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the baggage door is open. In the baggage door, there are 2 mechanical switches. When either switch detects the door is open, it presents a ground to the EICAS system. The message is displayed when either switch indicates the door is open. When a switch detects the door is closed, it presents an open. When both inputs are open, the message is removed.

ANTISKID FAIL

Color	Inhibited By		Debounce
Amber	POD	TOPI	*20 seconds

This message is displayed when the antiskid system has failed or the LOW BRAKE PRESSURE message is displayed. For I/O definition of low brake pressure, see LOW BRAKE PRESSURE message. When the antiskid controller determines a failure has occurred, it sends a ground signal to the EICAS, which posts the message after 20 seconds in the air and immediately on the ground. When the antiskid computer has normal operation, it sends an open, which removes the message if the LOW BRAKE PRESSURE message is also removed.

This message is inhibited for 20 seconds during initial DCU power up. This is to prevent a nuisance indication due to the antiskid controller performing a power up test and activating the fail output for 6 seconds.

The Antiskid On 28V/open input is used for troubleshooting the ANTISKID FAIL CAS message. The state of this input is captured by the MDC anytime the CAS message is active.

(*) = with exceptions

AOA HEAT FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the pitot/static heat switch is on and the AOA probe is not being heated. The AOA heater power is controlled by the pitot/static heat switch. The AOA computer detects current to the AOA heater and presents an open circuit to the EICAS system. When the AOA computer does not detect current to the AOA heater, it presents a ground to the EICAS system, which posts the message if the pitot/static switch is ON. The advisory PITOT/STATIC COLD L-R-STBY message is used to alert the crew if the pitot/static switch is OFF.

APU FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the APU has failed. An APU failure indicates either the APU ECU has reported a failure or the APU fire bottle is low. The APU fail light on the RH panel will also come on simultaneously with the CAS message. A 28 Volt input to EICAS means the APU has failed, which causes the message to be displayed. Open circuit means the APU has not failed, which causes the message to be removed.

APU GENERATOR OFF

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

This message is displayed when the APU is on and the APU generator relay is not closed. The message is amber if the APU generator switch is selected on, and it is white if the APU generator is not selected on. 28 Volts on the input means that the APU is on, the APU generator relay is closed, and the APU generator switch is selected on, respectively. Open means the APU is not on, the APU generator relay is open, and the APU generator switch is selected off, respectively. If the APU generator is reset while the APU generator is on-line, the APU generator reset switch will turn off the generator relay, and the white message shall appear.

APU ON

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message indicates the APU is on above 30,000 feet. APU operation is not approved above 30,000 feet.

BATTERY DOOR

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the battery door is open. In the battery door, there is a prox switch and a relay to invert the logic. When the door is away from the prox switch, the prox switch and relay combination presents a ground to the EICAS system, which displays the message. When the door is closed, an open is presented to the EICAS system, which removes the message.

Table 31-5. AMBER EICAS MESSAGES (Cont)

BLEED AIR OVERTEMP L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	20 Second

This message is displayed when the supply bleed air from the engine is too hot. A temperature switch in the supply duct provides a 28V signal to the EICAS, which posts the message after 20 seconds. When the supply temperature is normal, the switch provides an open to the EICAS, which removes the message.

CABIN AIR DUCT OVERTEMP

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the supply air in the cabin air duct is too hot. A temperature switch in the supply duct provides a ground signal to the EICAS, which posts the message. When the supply temperature is normal, the switch provides an open to the EICAS, which removes the message.

CABIN DOOR

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the cabin door is open. The cabin door is monitored by a logic PC card. The PC card monitors several inputs for correct sequencing. The PC card will also trigger the message as needed to prevent a switch failure from being latent. The PC card also controls a valve for the purpose of inflating the door seal and a solenoid for the purpose of opening the vent door. When the door is open, the PC card sends a ground to the EICAS system, which displays the message. When the door is closed, the PC card removes the ground, which causes the message to be removed.

CABIN DOOR SEAL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the pressure in the cabin door seal is less than 5 PSI. There is a pressure switch connected to the cabin door seal. Normally, the switch is closed, causing a ground to be presented to the EICAS system, which displays the message. When the pressure goes above 5 PSI, the switch opens, removing the ground input, which removes the message.

COCKPIT AIR DUCT OVERTEMP

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the supply air in the cockpit air duct is too hot. A temperature switch in the supply duct provides a ground signal to the EICAS, which posts the message. When the supply temperature is normal, the switch provides an open to the EICAS, which removes the message.

DC GENERATOR OFF L-R

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard
Amber	*ESDI	SIPI	*1.0 Seconds

This message is displayed when the respective generator contactor is open. Refer to red EICAS message for details.

DCU CHANNEL A FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1 Second
	POD		

This message indicates a loss of redundancy for processing of CAS or EIS parameters. The DCU disregards any information from faulty daughtercards, so all information presented should be correct.

DCU CHANNEL B FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1 Second
	POD		

This message indicates a loss of redundancy for processing of CAS or EIS parameters. The DCU disregards any information from faulty daughtercards, so all information presented should be correct. Only DCU Channel B is powered in EMER. If a DCU Channel B fail is present prior to switching to EMER power, the overspeed aural alert, landing gear warning aural, and 2 position tail lockout at 215 kts will not be functiona

DCU FAN FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

This message is displayed when the DCU cooling fan has failed. The fan should be functional prior to dispatch. If the failure occurs on ground, the message will be amber and will remain amber should the aircraft dispatch with the fan failed. If the failure occurs in air, the message will be white and will remain white until the aircraft has landed and LOPI inhibit is completed. If the fan fails in air, the aircraft may continue to the destination, but the fan should be repaired prior to dispatching again.

EFIS COMPARE INOP

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	5 Second

This message is displayed when the EFIS MISCOMPARE monitor is not being performed because one of the display has lost the cross side data used for performing the comparison.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

EFIS MISCOMPARE

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	5 Second

This message is displayed when a monitored miscompare has occurred. The associated yellow comparator flag will be displayed to indicate which monitored parameter has tripped the miscompare. Monitored parameters are: baro altitude, airspeed, attitude, heading, radio altitude, localizer and glideslope.

EMERGENCY EXIT

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the emergency exit is open. In the emergency exit, there is a proximity switch which detects the door pin. When the door pin is away from the prox switch, the switch presents a ground to the EICAS system, which displays the message. When the door pin is near the switch, it presents an open circuit, which removes the message.

EMERGENCY PRESSURIZATION

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when emergency pressurization is active. When emergency pressurization is active, 28V is provided to the emergency pressurization valve to provide additional inflow into the cabin. This 28V signal is also sent to the EICAS system. When the input is 28V, the message is displayed. When the input is open, the message is not displayed. The EICAS system also provides a ground/open output which is used by the audio attentuation PC board. When the emergency pressurization input is 28V, the output is ground. When the input is open, the output is open.

ENG FIRE BOTTLE LOW 1-2

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when an engine fire bottle is low, as measured by a pressure switch on the bottle. When the bottle is low, it sends a ground signal to the EICAS system, which posts the message. When the bottle is filled, it sends an open signal which removes the message.

ENG FIRE DETECT FAIL L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is posted when one of the engine fire detectors has failed. When a failure is detected, the fire detection controller sends a ground to the EICAS system, which displays the message. When the system is operating normally, the controller sends an open, which causes the EICAS to remove the message.

(*) = with exceptions

ENGINE ANTI-ICE COLD L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	5 Seconds
White	ESDI	SIPI	1 Second

In air operation - the white message is displayed when anti-ice is selected on, and the surface is not warmed up yet. If, after 285 seconds of cold, the white message becomes amber. The amber message also can come up if the surface has warmed up and then cooled off again. Once the amber message is shown, it remains for 5 seconds after the condition is removed.

On ground operation - the white message is displayed when anti-ice is selected on, until the surface becomes warm, then it goes out. There is no 285 second timer on the ground. The amber message also can come up if the surface has warmed up and then cooled off again.

The amber message can also be displayed, on ground or in air, if the fan/stator anti-ice valve is not in the correct position for more than 5 seconds.

ANTI-ICE on is: respective engine side anti-ice selected on or engine/wing anti-ice turned on. For I/O definition of engine/wing anti-ice, see WING ANTI-ICE COLD L-R.

Amber message logic is the following with a 5 second debounce on and off:
- ANTI-ICE on **AND**
- NOT engine shutdown **AND**
- In air **AND**
- Surface cold more than 285 seconds

OR
- ANTI-ICE on **AND**
- NOT engine shutdown **AND**
- Surface cold **AND**
- The surface was warm at least once since being selected on

OR
- NOT engine shutdown **AND**
- Engine fan/stator anti-ice valve is not in correct position

White message logic is the following for more than 1 second:
- ANTI-ICE on **AND**
- NOT engine shutdown **AND**
- NOT amber message **AND**
- In air **AND**
- Surface cold

OR
- ANTI-ICE on **AND**
- NOT engine shutdown **AND**
- NOT amber message **AND**
- On ground **AND**
- The surface was cold when selected on **AND**
- The surface has remained cold since selecting on

Engine cold is ground for cold, open for warm. Eng A/I On is ground for engine anti-ice selected on, open for off. F/S Valve Clsd is ground for valve closed, open for valve open. The valve is open to provide anti-icing to the fan and stator.

31 INDICATING AND RECORDING SYSTEMS

Table 31-5. AMBER EICAS MESSAGES (Cont)

ENGINE CONTROL FAULT L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
	POD	EFI	
		*SIPI	

This message is posted when a FADEC channel has failed. It is posted when FADEC 429 Label 271, bit 12 = 1 (Channel Fail Indication).

This message is inhibited for 20 seconds during initial DCU power up. This is to prevent nuisance indication due to the FADEC performing a power up test and activating this bit for 10 seconds.

* The message is also inhibited by an engine and/or APU start on the ground.

FIREWALL SHUTOFF L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	2 Second
White			Standard

The advisory white message indicates normal operation while the amber message indicates abnormal operation. Normal operation for firewall shutoff is both fuel and hydraulic shutoff valves closed when the ENGINE FIRE switches are selected. The aircraft has a red ENGINE FIRE and white FIRE BOTTLE ARM annunciator switches in the firetray for each engine. The ENGINE FIRE annunciator indicates the fire detection system has detected an engine fire for the respective engine. It closes the hydraulic and fuel firewall shutoffs and illuminates the FIRE BOTTLE ARM annunciator switch when pressed. Pressing it again will open the valves. The FIRE BOTTLE ARM switch deploys the fire bottles to extinguish the fire. Abnormal operation indicated by an amber FIREWALL SHUTOFF CAS message means the fuel and hydraulic valves of the same side are not in the commanded position.

When both fuel and hydraulic shutoff's on one side become closed, the white message for the respective side will be displayed. If one valve should open the message will turn amber after 2 seconds. The 2 second delay allows for both valves to open when commanded without triggering an amber message.

When the firewall shutoffs are closed, a switch in the valve sends a 28 Volt signal to the EICAS system. When the valve is not closed, the switch sends an open signal to the EICAS system.

FUEL BOOST PUMP ON L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White	SIPI		

The amber message is displayed when the fuel boost pump is on, fuel pressure is low, and the throttle is not in cutoff. Once the amber message is displayed, it will remain latched until the fuel pressure becomes normal and the fuel boost pump is off. This message is inhibited during start and when the engine is not running.

The white message is displayed when the fuel boost pump is selected on, APU running, or not turned on by low fuel pressure.

When the boost pump is on, the EICAS receives the same 28V signal which drives the pump, and it posts the message. When the pump is off, the EICAS reads a ground through the resistance of the pump. For I/Os for throttle in cutoff and low fuel pressure, see the FUEL PRESSURE LOW message.

FUEL CROSS FEED

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*10 Second
White			

Fuel Cross Feed operation - When the fuel selector is selected to the left tank or right tank, the normal operation is to increase the fuel pressure in the tank you are cross feeding from, then open the fuel cross feed valve, and reduce the fuel pressure in the tank you are not cross feeding from.

The white message is displayed when the fuel cross feed valve is commanded open from the cockpit crossfeed switch. **The amber message is displayed when the fuel cross feed valve is not in agreement with the selected crossfeed switch position.** The white message has the standard debounce, and the amber message has a 10 second debounce.

When fuel cross feed is not selected, a ground is sent to the EICAS system from the switch in the cockpit. When cross feec is selected, an open is sent to the EICAS system. When the cross feed valve is either open or closed, one of two switches in the valve sends a 28 Volt signal to the EICAS. When the valve is neither open or closed, neither switch is made and both inputs are open.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

FUEL FILTER BYPASS L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
	*ESDI	SIPI	

This message is displayed when the fuel filter impending bypass is true. This message has two different sets of inputs that can trigger the message. A configuration strap is used to tell the DCU which set of inputs to use. The two sets of inputs are either the impeding/actual fuel bypass switches or the differential pressure transducers.

With the fuel bypass configuration strap pin grounded, the impeding and actual fuel bypass switches are used to trigger the message, They measure pressure across the fuel filter. The impending fuel bypass is set to trip at 14 +/- 2 PSID (14 PSI = 44.34 mV) and is the trigger for the CAS message, while the actual bypass is set to trip at 26 +/- 2 PSID (26PSI = 78.06 mV) and is provided for fault monitoring only (no CAS message). The typical pressure drop across the fuel filter is approximately 1.2 PSID. The fuel filter pressure relief valve will open at 32 +/- 2 PSID as measured across the fuel filter.

Without the fuel bypass configuration strap pin grounded, the differential pressure transducer is used to trigger the message. The DCU transmits differential fuel pressure, corrected for sensor excitation voltage error and filtered per PWC requirements, to the FADEC via GPBUS-5 label 346 at a 10 Hz update rate.

FUEL GAUGE L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when there is a fault in the fuel quantity indicating system, as determined by the fuel quantity signal conditioner. When the signal conditioner detects a failure, it sends a ground signal to the EICAS system, which posts the message. When the signal conditioner is in normal operation, it sends an open to the EICAS, which removes the message.

FUEL LEVEL LOW L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*34 Second

This message is displayed when the fuel level in the fuel tank is low as determined by a float switch. When the fuel level is less than approximately 360 lbs, the float switch sends a ground signal to the EICAS system, which displays the message. When the fuel level is greater than 360 lbs, the switch sends an open to the EICAS system, which removes the message. The message has a 34 second debounce on, and a 32 second debounce off.

There are dual paths for presentation of a low fuel condition on the XLS+. In addition to the CAS message, the fuel quantity display on the MFD will turn amber and flash for ten seconds for indication of a low fuel condition. This is a Level A independent path that does not go thru the DCU.

FUEL PRESSURE LOW L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
	ESDI	SIPI	

The message is displayed when the fuel pressure is low, and the respective engine is running. For the purposes of this message, engine running begins when the start contactor disengages and ends when the throttle is put into cutoff.

For I/O definition of engine start, see: Start Contactor in the power distribution system section. When the fuel pressure is low, a pressure switch provides a ground signal to the EICAS system, which posts the message. When the pressure is normal, the switch sends an open signal to the EICAS, which removes the message. Fuel cutoff is a switch in the throttle quadrant which detects if the throttle is in cutoff. When it is in cutoff, a ground is provided to the EICAS system. When it is not in cutoff, an open signal is provided.

GROUND IDLE L-R

Color	Inhibited By		Debounce
Amber		TOPI	1 Second

This message is displayed if a FADEC failure should result in ground idle mode in air. When FADEC 429 Label 271, bit 16 = 1 (Ground Idle Indication) and the aircraft is in air, the EICAS posts the message. This message has TOPI and 1 second debounce.

HYDRAULIC FLOW LOW L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*5 Second
	*ESDI	SIPI	

This message is displayed when the hydraulic flow is low after engine start. The message has a 5 second debounce on, and a 3 second debounce off. On the output of each engine driven pump, there is a flow sensitive switch, which sends a ground to the EICAS system when the flow is low, which displays the message after 5 seconds. When the flow is normal, the switch provides an open signal, which removes the message after 3 seconds.

* The engine shutdown inhibit (ESDI) is not active in the air.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

HYDRAULIC PRESSURE

Color	Inhibited By		Debounce
Amber	*LOPI	*TOPI	*40 Second
White			Standard

This message is displayed when hydraulic pressure is in the hydraulic system. The message changes to amber if there is pressure for more than 40 seconds in the air. There is a hydraulic pressure switch which provides a ground to the EICAS system when the pressure is above 185 PSI, which displays the message. When the pressure drops below 155 PSI, the switch opens and the message is removed.

* The white message does not have TOPI or LOPI, the amber message has TOPI and LOPI.

HYDRAULIC FLUID LEVEL LOW

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the hydraulic fluid level in the reservoir is low. There is a mechanical switch on the reservoir which provides a ground signal to the EICAS when the fluid level is low. When the EICAS receives the ground, it posts the message. When the fluid level is normal, an open is sent to EICAS, which removes the message.

IAPS FAULT

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1 Second
White			

This message is displayed when the IEC monitor has detected a fault in the environmental control of the IAPS. Faults that will trigger this message include:

1. Fan too slow
2. Fan too fast during heating
3. Command to be on, but it's not
4. Command to be off, but it's on
5. Left or Right transducer fail
6. HTR CMD or HTR ARM switch failure (monitored for open or short)

The IAPS cooling fan is part of the IEC-3001 environmental control module. The fan should be functional prior to dispatch. If the failure occurs on ground, the message will be amber and will remain amber should the aircraft dispatch with the fan failed. If the failure occurs in air, the message will be white and will remain white until the aircraft has landed and LOPI inhibit is completed. If the fan fails in air, the aircraft may continue to the destination, but the fan should be repaired prior to dispatching again. The amber message is also inhibited during APU start on the ground.

IAPS OVERTEMP

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	15 Second

Each IAPS channel monitors the opposite channel's power supply for overheat conditions. This message is displayed when the power supply has overheated and is entering the overtemp shutdown cycle. After this message appears, the IAPS will shut down in 3 minutes. An IAPS shutdown will result in loss of the FMS, FD/AP, and YD.

J-BOX CURRENT LIMITER

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when one of the two 225 Amp limiters in the power J-Box have opened. There are two 5 amp sense breakers in parallel with the limiters. When the limiter opens, current starts flowing through the breaker, which then trips. The breaker has a set of auxiliary contacts which sends a ground to the EICAS system, which posts the message. The auxiliary contact are wired in parallel so that only one input is needed for the EICAS system. When both breakers are engaged, an open is sent to the EICAS system, which removes the message. See: Power Distribution System Schematic.

J-BOX START CB

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when one of four breakers for the start cards has tripped. The breaker has a set of auxiliary contacts which sends a ground to the EICAS system when it has tripped. The EICAS posts the message when it gets the ground, and removes the message when the input is open.

LAVATORY DOOR

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the lavatory door is closed and the aircraft is on the ground or flaps out of 0° position. On the door, there is a mechanical switch. When the door is closed, the switch presents a ground to the EICAS system, which displays the message. When the door is open, the switch presents an open to the EICAS system, which removes the message.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

LOW BRAKE PRESSURE			
Color	Inhibited By		Debounce
Amber	POD		*20 Second

This message is displayed when the brake pressure is low and the right main gear is down and locked. The gear down condition is implemented outside of EICAS. This message is inhibited for a 20 second debounce period during initial DCU power up to allow the brake pressure to build up. When the pressure is under 900 PSI, and the right gear is down and locked, the pressure switch sends a ground to the EICAS system, which posts the message after 20 seconds in the air and during initial DCU power up. After 20 seconds of initial power up, if low brake pressure comes back, the message will immediately come on. When the pressure is over 1100 PSI, the pressure switch sends an open to the EICAS system, which removes the message.

The “LOW BRAKE PRESSURE” cautionary CAS message functions differently from the other cautionary CAS messages. Once the logic equation goes true on the ground, the message will continue to flash and the master caution light will continue to illuminate steady, regardless if the master caution reset switch is pressed. The flashing message and the steady master caution light output shall continue to function this way, until the logic equation goes false. In the air, the message can be acknowledge with the master caution reset switch.

A single Master Caution tone alert associated with this message shall sound only once for the duration of the condition.

The Brake CB Engaged 28V/open input is used for troubleshooting the LOW BRAKE PRESSURE CAS message. The state of this input is captured by the MDC anytime the CAS message is triggered.

NOSE DOOR			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when either nose door is open. There is one switch for each door latch, 2 latches per door, and 2 doors per airplane, for a total of 4 inputs to the EICAS system. When the latch is unlatched, the switch will present a ground to the EICAS system, which will display the message. Any of the 4 inputs can trigger the message. When a latch is latched, the switch will present an open circuit to the EICAS system. When all 4 inputs are open, the message will be removed.

OIL FILTER BYPASS L-R			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
	ESDI	SIPI	

This message is displayed when the oil filter is impending bypass. When oil pressure remains below the trip point, the oil filter differential pressure switch sends a ground to the EICAS system, which removes the message. When the pressure exceeds the max allowable pressure differential across the oil filter, the switch sends an open signal to the EICAS, which posts the message.

PITCH TRIM FAIL			
Color	Inhibited By		Debounce
Amber		TOPI	1 Second

This message is displayed when the autopilot control of elevator trim is inoperative.

PITOT/STATIC COLD L-R-STBY			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

The amber message(s) are displayed when the pitot/static heat is selected on, but current is not flowing in one of the heaters. It is also displayed if the heat is selected off, and the airplane is in the air. The advisory message is displayed on ground when the pitot/static switch is selected off.

A current sensor is wired in series with each heater. When current is flowing through the heater, it also flows through the current sensor. The current sensor has a coil and a set of contacts, very similar to a relay. There are 3 current sensors for each set of ports, and 3 sets of ports per airplane, for a total of 9 current sensors. The current sensors for each set of ports are wired in parallel. When current is flowing, an open is provided to the EICAS system. When the current is not flowing, a ground is provided to the EICAS system, which posts the message according to the logic in the Pitot/Static Logic Chart.

PRESS SOURCE NOT NORM			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the pressurization selector in the cockpit is not in the NORM position, and emergency pressurization is not active. The EICAS system receives a 28V logic signal when the pressurization selector is in the NORM position. When the input has 28V, the message is not displayed. When the input is open, the EICAS displays the PRESS SOURCE NOT NORM CAS message if the EMERGENCY PRESSURIZATION CAS message is not active.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

RADOME FAN FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1.5 Second

This message is displayed when the fan in the nose radome has failed. When the fan has failed, a ground signal is sent to the EICAS system, which posts the message.

RAT HEAT FAIL L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
	POD		

This message is displayed when the FADEC detects a failure of the TTO heater. When a failure is detected, the FADEC sets 429 Label 275, bit 14 = 1 “TTO Heater Fail LSS Indication”, which causes the message to be displayed.

This message is inhibited for 20 seconds during initial DCU power up. This is to prevent nuisance indication due to the FADEC performing a power up test and activating this bit for 10 seconds.

RETRIM L-R WING DOWN

Color	Inhibited By		Debounce
Amber		TOPI	5 Second

This message indicates that the autopilot is detecting a lateral mistrim. In other words, the aileron servo is holding a load. L and R are mutually exclusive.

RETRIM NOSE UP-DOWN

Color	Inhibited By		Debounce
Amber		TOPI	5 Second

This message indicates that the autopilot is detecting a longitudinal mistrim. In other words, the elevator servo is holding a load.

Normally, the autopilot would command stabilizer trim to relieve the load before tripping this message. If the trim is not running, the PITCH TRIM FAIL message would then be displayed. This message means that there is a load, the AP is commanding and getting stabilizer trim, and the load is not going away.

The distinction is that a large force should be expected to control the aircraft when the AP disconnects, whereas PITCH TRIM FAIL indicates a small force should be expected. UP and DOWN are mutually exclusive.

RUDDER BIAS FAULT

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1 Second

This message monitors the rudder bias control valve for proper operation. The EICAS system gets 3 inputs: one input is the command going to the valve, and the other 2 inputs are from two mechanical switches within the valve that indicate the position the valve is in. For the command input, 28 Volts means the valve is being commanded to open, and open means the valve is being commanded to close. For the sense inputs, ground means that the valve is in the respective position, and open means the valve is not in the respective position. The message is posted according to the logic in the Rudder Bias Fault Truth Table.

RUDDER BIAS HEAT FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
		*SIPI	

This message is displayed when the rudder bias heater blanket is failed as determined by the Rudder Bias Heater PC card. When the heater blanket has failed, the PC card sends an open signal to the EICAS system, which posts the message. When the heater blanket is operating normally, the PC card sends a ground signal, which causes the EICAS to remove the message.

* The message is also inhibited by an engine and/or APU start on the ground.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

STAB MISCOMPARE

Color	Inhibited By		Debounce
Amber	LOPI		Standard

The horizontal stabilizer changes positions through the operation of a hydro-mechanical actuator. An electrical control and monitoring system controls the flow of hydraulic fluid to the horizontal stabilizer actuator (HSA). The electrical control system receives command input from the flap selector handle on the pedestal in the cockpit. When the flight crew selects FLAPS 0°, the stabilizer moves to the "Cruise" position (approximately +1° incidence). When the flight crew selects a flap position other than FLAPS 0°, the horizontal stabilizer moves to the "Takeoff - Landing" position (approximately -2° incidence).

The logic for the STAB MISCOMPARE caution CAS message resides in the two position tail PCB. The DCU receives two discrete inputs from the two position tail PCB. **The Stab Position Master Caution discrete indicates the two position tail is not in the correct position for the aircraft configuration. The Stab Position Fail indicates the inputs to the two position tail PCB are contradictory or invalid and the correct stab position cannot be determined. Either of these discrete will generate the STAB MISCOMPARE caution CAS message.**

The two position tail PCB receives inputs from the flap handle switches, the two position tail position switches, and the airspeed >215 discrete output from the DCU.

The two position tail PCB will set the Stab Position Master Caution discrete for the following conditions:

1. If the stab position does not reach the up position within 32 ± 3 seconds after flaps retracted, or within 42 ± 3 seconds of landing gear operation.
2. If the stab is moving at airspeeds greater than 215 Kts.

The two position tail PCB will set the Stab Position Fail discrete for the following conditions:

1. If the flap handle switches indicate flaps up and flaps down simultaneously.
2. If the stab position does not reach the up position within 32 ± 3 seconds after flaps retracted, or within 42 ± 3 seconds of landing gear operation.
3. If the stab is moving at airspeeds greater than 215 Kts.
4. If the stab position does not reach the down position within 32 ± 3 seconds after flaps are moved out of the 0° position or within 42 ± 3 seconds of landing gear operation.

Either Stab Position Fail w/MC or Stab Position Fail will result in the STAB MISCOMPARE CAS caution message and the accompanying MASTER CAUTION RESET annunciator light activation.

(*) = with exceptions

TAIL DE-ICE FAIL L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

When a failure of the tail de-icing system is detected by the Tail De-Ice PC Card, the card sends an open signal to the EICAS system, which posts the message for the respective side. When the tail de-ice system has normal operation, it sends a ground signal and the EICAS removes the message.

TAILCONE ACC DOOR

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the tailcone access door is open. On the door, there is a mechanical switch. When the door is open, the switch presents a ground to the EICAS system, which displays the message. When the door is closed, the switch presents an open to the EICAS system, which removes the message.

TAWS BASIC FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1.0 Second

This message is displayed when the radio altimeter based ground prox modes of the TAWS function have failed, and the TAWS SYSTEM FAIL message is not active.

TAWS SYSTEM FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1.0 Second

This message is displayed when all the TAWS functions (ground prox, windshear and terrain) have failed. When this message is displayed, it inhibits the TAWS BASIC FAIL, TAWS WINDSHEAR FAIL, and TAWS TERRAIN FAIL messages.

TAWS TERR FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1.0 Second

This message is displayed when the enhanced modes of the TAWS function have failed, and the TAWS SYSTEM FAIL message is not active.

TAWS TERR NOT AVAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1.0 Second

This message is displayed when the GPS data received by the TAWS unit is not within required accuracy, or GPS data is not available.

Table 31-5. AMBER EICAS MESSAGES (Cont)

TAWS WINDSHEAR FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1.0 Second

This message is displayed when the windshear modes of the TAWS function have failed, and the TAWS SYSTEM FAIL message is not active.

WINDSHIELD HEAT INOP L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*8 Second
	*ESDI	SIPI	

The windshield is electrically heated. The heating is controlled by a windshield heat controller. The windshield heat controller is powered when the aircraft battery switch is turned on. **This message is displayed when the controller has detected a failure.** When a failure is detected, the controller sends a ground signal to the EICAS system, which displays the message. When the input is open, the message is not displayed.

The message is inhibited during engine start, engine shutdown and not running. For I/O of engine start see: GCU Engine Start.

* The 8 second debounce and engine shutdown are removed by windshield rotary test. The engine shutdown inhibit (ESDI) is also removed in the air.

WINDSHIELD OVERTEMP L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*4 Second

This message is displayed when the windshield controller has detected an overheat situation. The overheat could result in structural damage. When the controller detects the overheat, it sends a ground to the EICAS system, which displays the message. An open signal removes the message.

The 4 seconds debounce is removed by windshield test.

WING ANTI-ICE COLD L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

In air operation - the white message is displayed when wing anti-ice or crossflow is selected on, and the surface is not warmed up yet. If, after 285 seconds of cold, the white message becomes amber. The amber message also can come up if the surface has warmed up and then cooled off again. Once the amber message is shown, it remains for 5 seconds after the condition is removed.

On ground operation - the white message is displayed when wing anti-ice or crossflow is selected on, until the surface becomes warm, then it goes out. There is no 285 second timer on the ground. The amber message also can come up if the surface has warmed up and then cooled off again.

Neither message can come on if anti-ice is selected off. X-flow counts as on for both sides, even if the side that you are cross flowing from is off.

ANTI-ICE on is: X-flow selected on OR respective side wing anti-ice selected on.

Amber message logic is:
- ANTI-ICE on **AND**
- In air **AND**
- Surface cold more than 285 seconds

OR
- ANTI-ICE on **AND**
- Surface cold **AND**
- The surface was warm at least once since being selected on

OR either of the above was true in the last 5 seconds.

White message logic is:
- ANTI-ICE on **AND**
- NOT amber message **AND**
- In air **AND**
- Surface cold

OR
- ANTI-ICE on **AND**
- NOT amber message **AND**
- On ground **AND**
- The surface was cold when selected on **AND**
- The surface has remained cold since selecting on

X-flow is ground for x-flow selected off, open for x-flow selected on. Wing cold is ground for cold, open for warm. Anti-Ice Off is ground for engine and wing anti-ice selected off, open for engine and wing anti-ice selected on.

(*) = with exceptions

Table 31-5. AMBER EICAS MESSAGES (Cont)

WING ANTI-ICE OVERTEMP L-R			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

There are three over temperature switches in each wing for a total of six switches. The switches are behind the heat shield on the forward wing spar.

When the temperature is over 160°F at either switch, the switch sends a ground signal to the EICAS, which posts the message for the respective side.

There is also a temperature switch inside the fuselage at the wing root on both sides which trips at 220°F. All three overtemp switches per side are wired in parallel for a total of two inputs to EICAS. When the temperature is normal at all three switches, the respective EICAS input is open and the message is removed.

(*) = with exceptions

Table 31-6. WHITE EICAS MESSAGES

AC GENERATOR BEARING L-R

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard
	ESDI	EFI	
	SIPI		

The AC Generator is used only for windshield heating. This message is displayed when the bearing has worn out. There is a sensor built into the bearings at each end of the generator. The sensors are wired in parallel. When the bearing is worn out, a ground is provided to the EICAS system, which displays the message. When the input is open, the message is not displayed. This message is also inhibited during engine start.

APU FIRE BOTTLE LOW

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard

This message is displayed when the APU fire bottle is low, as measured by a pressure switch on the bottle. When the bottle is low, it sends a ground signal to the EICAS system, which posts the message. When the bottle is filled, it sends an open signal which removes the message. The APU FAIL message will be display with this message.

APU GENERATOR OFF

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

This message is displayed when the APU is on and the APU generator relay is not closed. The message is white if the APU generator is not selected on. Refer to amber EICAS message for details.

CAS MISCOMPARE

Color	Inhibited By		Debounce
White			20 Second

This message is displayed when there is a miscompare of EICAS messages or aurals between the DCU channels.

DCU FAN FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

This message is displayed when the DCU cooling fan has failed. Refer to amber EICAS message for details.

DCU RIGGING INVALID

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard

This message indicates that the Flap position sensor potentiometer and/or the FDR flight surface position sensor RVDT rigging data stored in the DCU NVRAM (Non-volatile RAM) is invalid. Re-rigging needs to be performed. It is displayed when the NVRAM is failed, has been cleared, or there is a miscompare of the data. Possible causes of this message include swapping DCUs from aircraft to aircraft or installing a new or repaired DCU.

ENGINE ANTI-ICE COLD L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	*5 Seconds
White	ESDI	SIPI	1 Second

This message indicates that anti-ice is selected on and the surface has not yet warmed up. Refer to amber EICAS message for details.

ENGINE VIBRATION L-R

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard
	ESDI	EFI	
	SIPI		

This message is displayed when the engine vibration exceeds the allowed limit. Each engine has an accelerometer, which is wired to a monitoring unit. The vibration monitor sends a ground signal to the EICAS, which posts the message. When the vibration is within limits, the monitor sends an open to the EICAS, which removes the message. This message is also inhibited during engine start.

FDR FAIL

Color	Inhibited By		Debounce
White	LOPI	TOPI	7 Second

This message is displayed when the Avionics master switch is selected on and the FDR has sent a fail discrete output. If the FDR is not powered or there is a failure detected within the unit, the FDR will send a ground signal to the EICAS system, which will post this message after 7 seconds. If the input is open, the message will not be posted.

FIREWALL SHUTOFF L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	2 Second
White			Standard

The advisory white message indicates normal operation. Refer to amber EICAS message for details.

(*) = with exceptions

Table 31-6. WHITE EICAS MESSAGES (Cont)

FUEL BOOST PUMP ON L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White	SIPI		

The white message is displayed when the fuel boost pump is selected on, APU running, or not turned on by low fuel pressure. Refer to amber EICAS message for details.

FUEL CROSS FEED

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	
White	SIPI		*Standard

The white message is displayed when the fuel cross feed valve is commanded open from the cockpit crossfeed switch. The white message has the standard debounce, and the amber message has a 10 second debounce. Refer to amber EICAS message for details.

HYDRAULIC PRESSURE

Color	Inhibited By		Debounce
Amber	*LOPI	*TOPI	
White			*Standard

This message is displayed when hydraulic pressure is in the hydraulic system. Refer to amber EICAS message for details.

IAPS FAULT

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	1 Second
White			

This message is displayed when the IEC monitor has detected a fault in the environmental control of the IAPS. If the failure occurs in air, the message will be white and will remain white until the aircraft has landed and LOPI inhibit is completed. Refer to amber EICAS message for details.

NEW DATALINK MESSAGE

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard

This message is displayed when a Universal Graphical Weather image or ACARS text message is available for viewing.

NO TAKEOFF

Color	Inhibited By		Debounce
Red	LOPI	In Air	Standard
White			

On the ground, the white NO TAKEOFF message will illuminate if one or more of the following conditions exist:

- **Flaps** not within takeoff range (<7° or >15°)
- **Elevator** out of trim for takeoff
- **Horizontal Stabilizer** is out of takeoff position
- **Speed Brakes** are out of takeoff position

Refer to red EICAS message for details.

PITOT/STATIC COLD L-R-STBY

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

The white message is displayed on ground when the pitot/static switch is selected off. Refer to amber EICAS message for details.

RUDDER BIAS COLD

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard
		*SIPI	

This message is displayed while the rudder bias heater system is cold and it is not failed. The rudder bias actuator is wrapped with an electrical heater blanket . The heating is controlled by a Rudder Bias Heater PC card. When PC card senses the heater blanket is cold, the card sends an open signal to the EICAS system, which posts the message if it is not failed. When the heater blanket has warmed up, the card sends a ground, which causes the message to be removed.

* The message is also inhibited by an engine and/or APU start on the ground.

SELCAL DATALINK

Color	Inhibited By		Debounce
White	LOPI	TOPI	1 Second

This message is displayed when the SELCAL code is received on the datalink. It produces the SELCAL aural defined in SELCAL HF 1-2 VHF 1-2-3.

(*) = with exceptions

Table 31-6. WHITE EICAS MESSAGES (Cont)

SELCAL HF 1-2 VHF 1-2-3

Color	Inhibited By		Debounce
White	LOPI	TOPI	1 Second

SELCAL is a system that monitors the HF and VHF COMM radio for an aircraft specific code sequence. When the code for that particular aircraft is received, this message is displayed. This message produces a unique tone. The SELCAL aural is a 1000 Hz, 0.2 second chime, followed by a 850 Hz, 0.2 second chime, followed by a second set of 1000 Hz / 850 Hz chimes.

Discrete 429 bits are used to provide SELCAL signaling.

VHF Selcal decoding is accomplished in the VHF-200. HFSELCAL decoding is acommplished in the RIU.

VHF-3 transmits L005 VHF SELCAL (decoded tone combo) to RIU. Then RIU-6 transmits to IAPS. Then both GPBUS-1 & GPBUS-5 transmit L175 Aural Annunciation Word that provides SELCAL discrete indication for VHF#1, VHF#2, VHF#3, HF#1, HF#2, & Datalink.

In order to simplify the logic implementation, the SELCAL messages are displayed one item per line, for example, if HF 1 and VHF 1 are active at the same time then, SELCAL HF 1 and SELCAL VHF 1 would be on separate lines in the CAS stack.

On XLS+, VHF 2 is currently not equipped for Selcal.

SPEED BRAKES

Color	Inhibited By		Debounce
White		TOPI	Standard

This message is displayed when either speed brake panel is extended. On each speed brake, there is a mechanical switch which sends a 28 Volt signal to the EICAS to display the message. When the speed brake is not extended, an open signal is sent to the EICAS system.

TAIL DE-ICE PRESS ON L-R

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard

The 560XLS+ uses a rubber boot to deice the tail vertical and horizontal surfaces. The pilots select a switch which sends service air to inflate the boots, causing the ice to pop off. This message is displayed when there is air pressure in the boot. In the service air supply system, there is a pressure switch which sends a ground signal to the EICAS system when the pressure is over 16 PSI. When the EICAS receives the ground, it posts the message for the respective side. After popping the ice off, the boot deflates, and the pressure switch sends an open signal to the EICAS, which removes the message.

WING ANTI-ICE COLD L-R

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

The white message is displayed when wing anti-ice or crossflow is selected on, and the surface is not warmed up yet. Refer to amber EICAS message for details.

(*) = with exceptions

31 INDICATING AND RECORDING SYSTEMS

QUESTIONS

1. During BATT EMER operations, the MASTER WARNING system:
   A. Continues to function normally
   B. Is inoperative
   C. Displays only CAS messages without the MASTER WARNING RESET/MASTER CAUTION RESET switchlights
   D. Continues to display red CAS messages, but, do not display amber or white CAS messages

2. The MASTER WARNING RESET and MASTER CAUTION RESET switchlights display as follows:
   A. Both switchlights illuminate flashing.
   B. The MASTER CAUTION RESET switchlight illuminates flashing, and, the MASTER WARNING RESET switchlight illuminates steady.
   C. The MASTER WARNING RESET switchlight illuminates flashing, and, the MASTER CAUTION RESET switchlight illuminates steady.
   D. Both switchlights illuminate steady.

3. Which CAS messages will be accompanied by an aural warning?
   A. All CAS messages are accompanied by aural warnings.
   B. All red CAS messages except EMERGENCY DESCENT are accompanied by an aural warning.
   C. All red and some critical amber CAS messages are accompanied by aural warnings.
   D. Only red ENGINE FIRE L-R or DC GENERATORS OFF CAS messages are accompanied by aural warnings.

4. Selecting ANNU on the rotary TEST knob results in the following indications for a successful test:
   A. All MASTER WARNING RESET and MASTER CAUTION RESET switchlights and all AOA Indexer lights illuminate.
   B. All MASTER WARNING RESET and MASTER CAUTION RESET switchlights and stand alone annunciator lights illuminates.
   C. All MASTER WARNING RESET and MASTER CAUTION RESET switchlights illuminates and all aural alerts play.
   D. Only the MASTER WARNING and MASTER CAUTION switchlights illuminates.

CITATION XL/XLS/XLS+ MAINTENANCE TRAINING MANUAL

VOLUME 2

REVISION 0.3

FlightSafety International, Inc.
Marine Air Terminal, LaGuardia Airport
Flushing, New York 11371
(718) 565-4100
www.FlightSafety.com

NOTICE

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At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

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THIS PUBLICATION CONSISTS OF THE FOLLOWING:

Page No.	*Revision No.	Page No.	*Revision No.
Cover	0.3	38-1—38-22	0.2
i—iii	0.3	45-i—45-iv	0.2
iv	0.2	45-1—45-6	0.2
v	0.3	49-i—49-iv	0.2
vi	0.2	49-1—49-50	0.2
32-i—32-vi	0.2	51-i—51-iv	0.2
32-1—32-56	0.2	51-1—51-4	0.2
32-57	0.3	52-1—52-26	0.2
32-58—32-78	0.2	53-1—53-8	0.2
33-i—33-iv	0.2	54-1—54-6	0.2
33-1—33-46	0.2	55-1—55-4	0.2
34-i—34-viii	0.2	56-1—56-10	0.2
34-1	0.3	57-1—57-4	0.2
34-2—34-82	0.2	71-i—71-vi	0.2
35-i—35-iv	0.2	71-1—71-126	0.2
35-1—35-22	0.2	WA-i—WA-ii	0.2
36-i—36-iv	0.2	WA-1—WA-38	0.2
36-1—36-26	0.2	WAI-1—WAI-40	0.2
38-i—38-iv	0.2	APP A-1—APP A-6	0.2

*Zero in this column indicates an original page.

CONTENTS
VOLUME 2

CHAPTER 32
LANDING GEAR

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 32
LANDING GEAR

INTRODUCTION

This chapter presents the Landing Gear and brake systems for the Citation 560XL/XLS/XLS+ aircraft. General maintenance considerations are included, accompanied by functional and operational checks. References for this chapter and further specific information can be found in Chapters 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 32—"Landing Gear," of the Aircraft Maintenance Manual (AMM).

Figure 32-1. Landing Gear

GENERAL

The landing, gear system for the 560XL/XLS/XLS+ is controlled electrically and hydraulically actuated (Figure 32-1). The nose gear assembly consists of a single wheel assembly and an oil-over-air (nitrogen) strut; while the main gear assemblies also consist of a single wheel assembly, but an air-over-oil strut.

The nose gear tire is the chined type for water and slush deflection.

The main gear doors are mechanically linked to the main gear trunnions, so they extend and retract with the main gear and require no separate actuators. The nose gear has three doors. The aft "spade" door is linked to the nose gear trunnion, so it extends and retracts with the gear. The two forward "side" doors for the nose gear are actuated through a system of linked rods and torque tubes, so they are mechanically opened and closed as the nose gear extends and retracts.

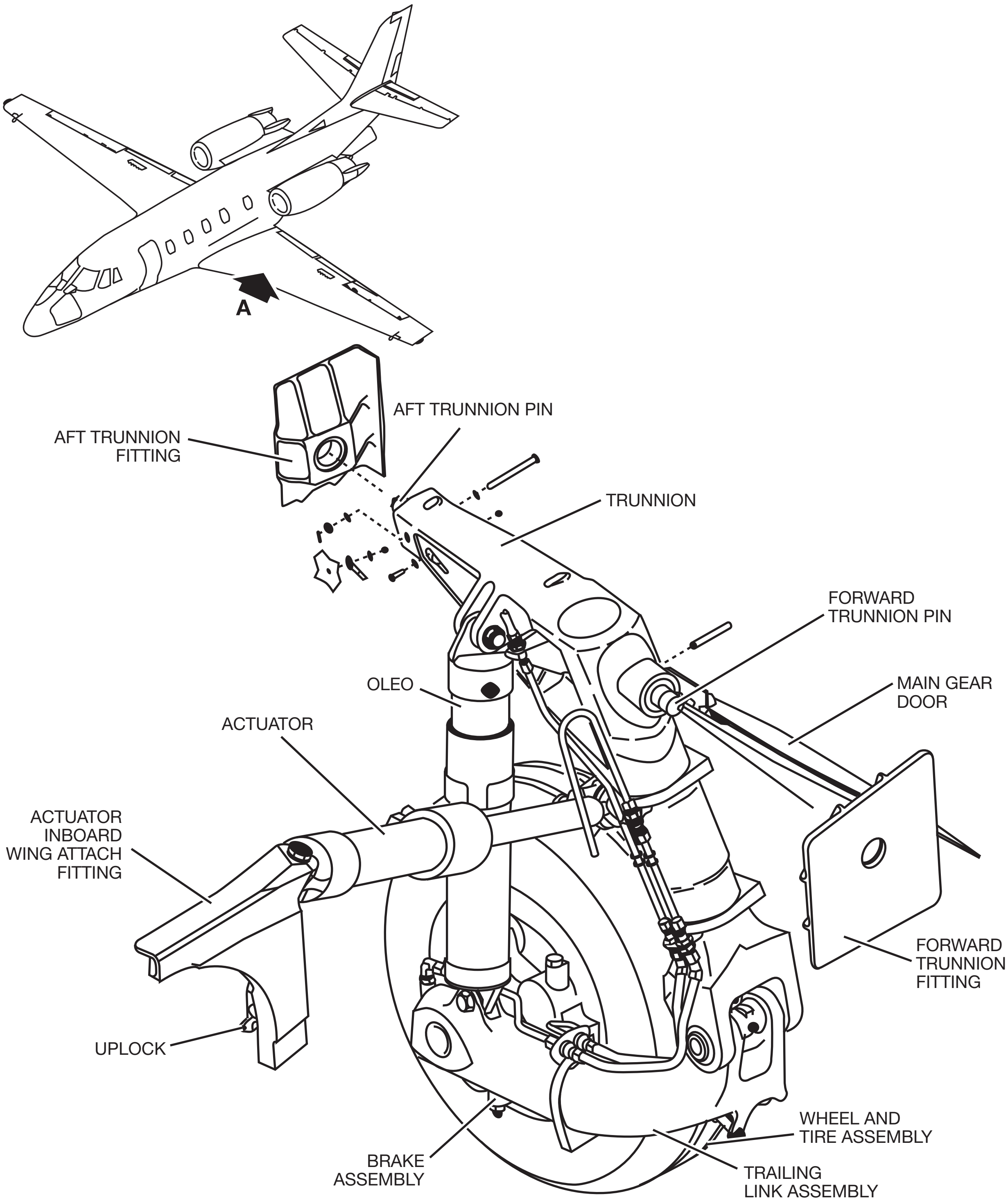
The actuators for each gear incorporate internal mechanical downlocks to hold the gear in the extended position. The gear is held retracted by mechanical uplocks that are spring-loaded to lock. They are hydraulically released.

CAUTION

The park brake must be released before lowering the aircraft from jacks. Failing to do so causes the aircraft to move forward as the main gear contacts the ground.

The main gear is equipped with hydraulically actuated disc brakes. An emergency (pneumatic) braking system is provided for use when hydraulic braking fails. Brake antiskid control prevents wheel skidding on wet, dry or icy runways after minimum wheel spin is attained.

NOTES



DETAIL A

Figure 32-2. Main Landing Gear

MAIN LANDING GEAR AND DOORS

DESCRIPTION

Each main gear consists of a trunnion, oleo assembly and trailing link (Figure 32-2). A wheel and tire assembly is attached to an axle/link assembly and an oleo assembly. The wheel/tire assembly is suspended by the trunnion and trailing link. The gear are hydraulically operated and retract inboard. The main landing gears are on two support attach fittings in the left and right wings. The trunnion aft attach fitting is on the wing rear spar and the trunnion forward attach fitting is on the wing main spar. The actuator assembly is inboard of the trunnion aft and forward attach-fittings. It supports the main landing gear assembly by connecting the gear actuator from the side brace to the main landing gear trunnion.

CAUTION

Be sure to observe all safety precautions in the *AMM* when removing or installing the main gear trunion, to prevent damage to the trunion bearings.

The main gear has a mechanical latching mechanism in the "retracted" position. The actuator has an internal downlock provision in the extended gear down position. The gear does not require hydraulic pressure, overcenter linkage or springs to hold the gear in the down and locked position. Hydraulic power is required to unlock the internal locking mechanism.

The oleo assembly is an air-over-oil assembly containing a metering pin and orifice fitting that varies the volume of fluid in the oleo assembly. Fluid movement in the oleo varies to compensate and resist shock according to severity.

COMPONENTS

Oleo Assembly

The oleo assembly is connected from the trunnion to the axle/link and trailing link assembly. When the oleo is being compressed (i.e., during landing) fluid is passed by the metering pin and orifice fitting. This varies the volume of fluid in the oleo. The volume of fluid metered by the metering pin controls the movement of the oleo. The oleo assembly contains nitrogen, which acts as a shock cushion.

Trailing Link Assembly

The trailing link assembly consists of an axle link on the aft end of the trailing link. The aft end of the trailing link assembly attaches to the base of the oleo assembly and connects the wheel and tire assembly. The forward end of the trailing link assembly is connected to the trunnion to absorb landing shock.

Main Landing Gear Doors

The main landing gear doors are designed to cover the wheel well (except for a portion of the tire) with the gear in the retracted position. The door is hinged to the lower portion of the wing structure and outboard edge of the wheel well; and is secured to the trunnion with a linkage rod. When the main landing gear is retracted, the door follows.

Brake Assembly

There is a multiple disc CARBON brake at each main landing gear wheel, operated by a power brake valve, which is controlled by master cylinders that connect to the lower portion of the rudder pedals.

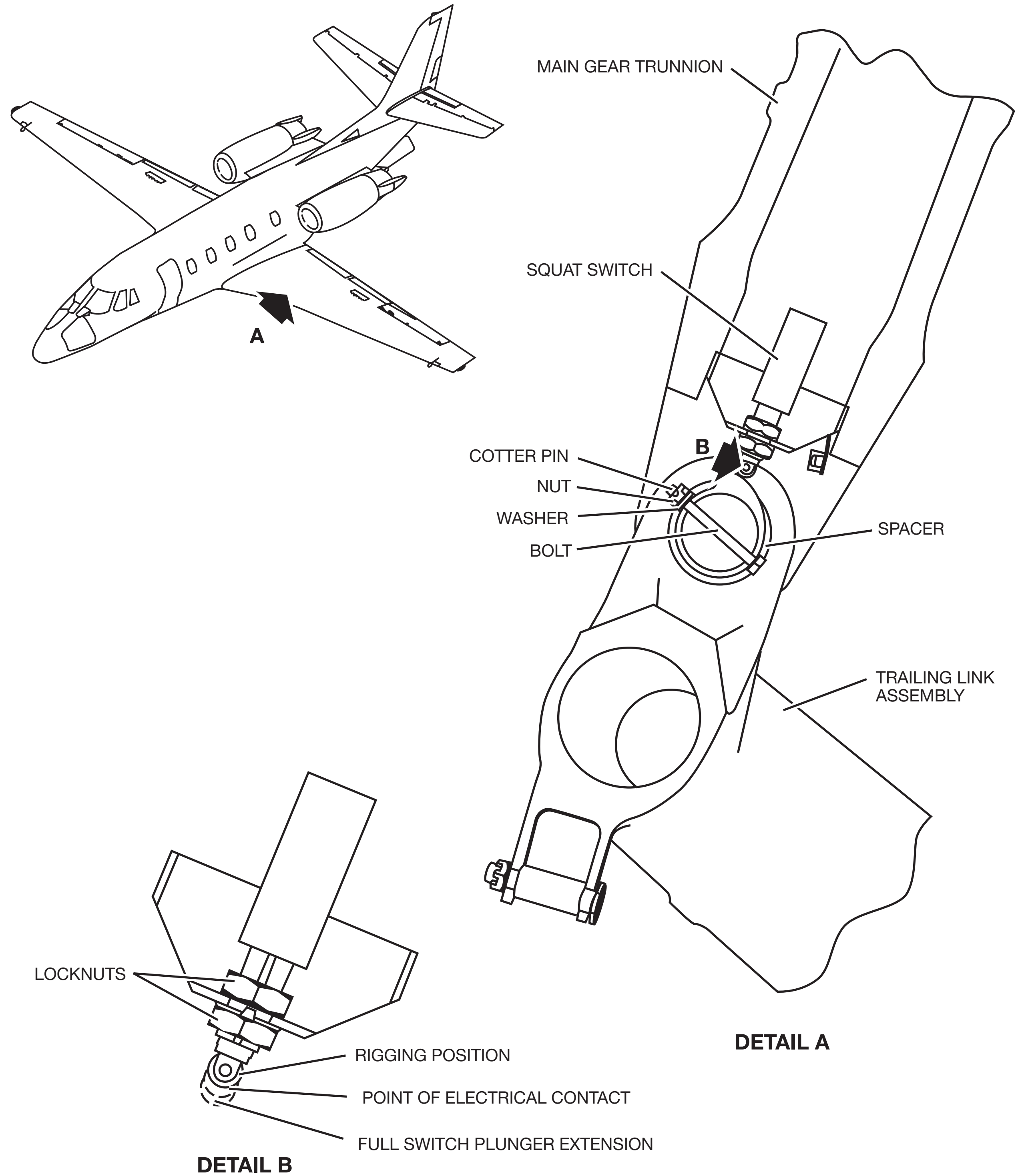


Figure 32-3. Squat Switch

Squat Switch

There is a main landing gear squat switch on each main gear trunion (Figure 32-3). As the aircraft lifts off the ground, the oleo strut extends, causing the main gear trailing link to rotate and actuate the squat switch.

DIAGNOSTICS

Squat Switch Adjustment/Test

1. Jack the aircraft.
2. Disconnect the squat switch electrical connector.
3. Make sure that main gear oleo is fully extended.

NOTE

Ensure the squat switch roller is resting on the top of the spacer ramp.

4. Adjust the switch with jamnuts until electrical actuation of the plunger occurs.
5. Continue to rotate the jamnuts one full turn past the electrical actuation point.
6. Secure the switch in this position and safety wire.
7. Connect the squat switch electrical connector.
8. Lower the airplane from jacks.

CAUTION

If the main gear oleo strut is removed do not allow the trailing link to be lowered enough to allow the squat switch plunger to drop on the front side of the cam. Raising the trailing link at that point causes damage to the switch.

NOTES

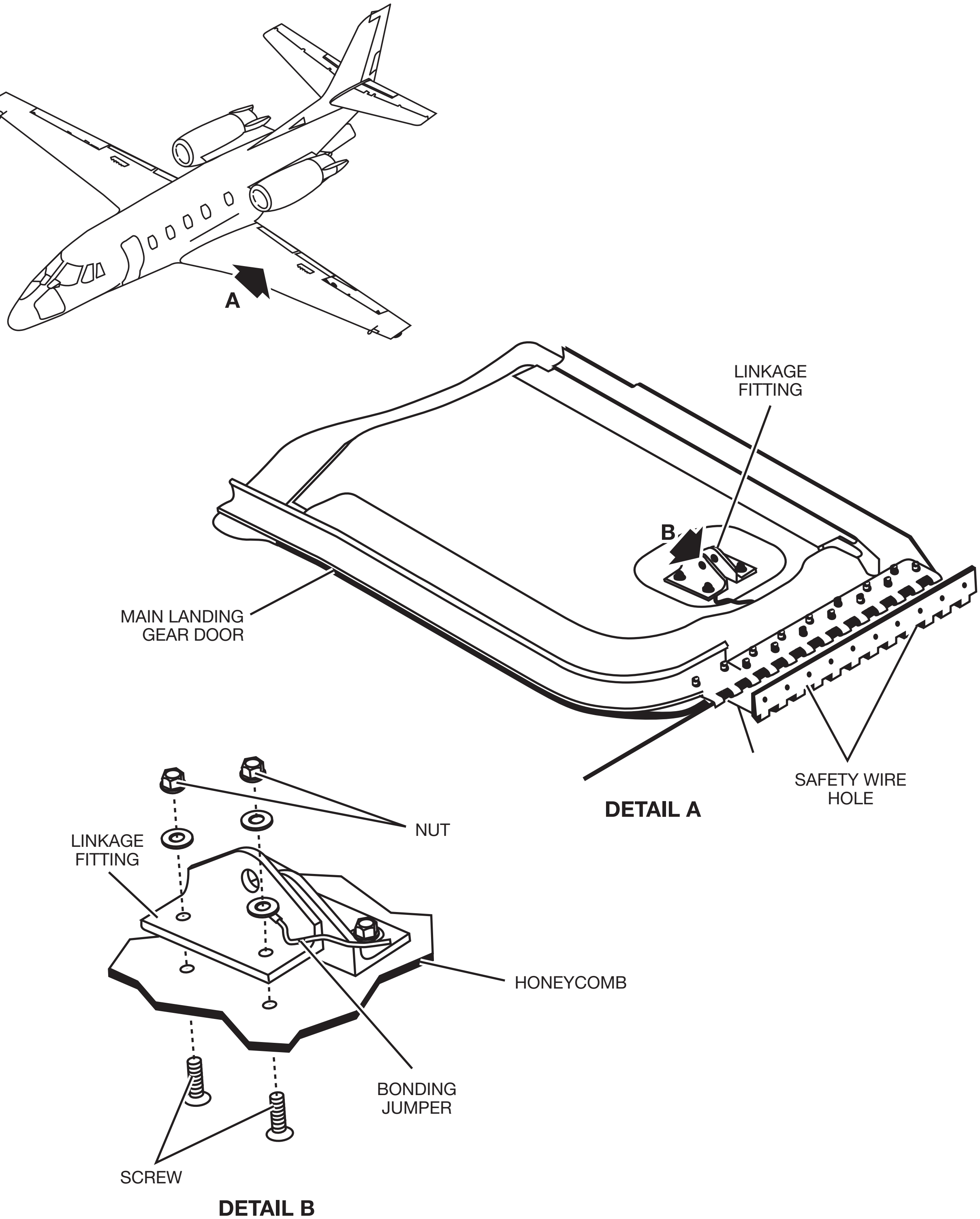


Figure 32-4. Main Landing Gear Doors

Adjust Main Landing Gear Door

Maintenance practices for the main landing gear and doors are identical (Figure 32-4). The main landing gear actuator has a self-locking provision in the gear down position. The locking action is maintained with no hydraulic pressure. Therefore, no ground downlock pins are required when servicing the airplane. The main landing gear doors are hinged to the airplane wing structure and are actuated by a linkage rod attached to the main landing gear.

1. Jack the airplane.

WARNING

Clear personnel away from the main landing gears and speedbrake areas before applying hydraulic power to the airplane.

2. Connect the hydraulic power unit to the airplane.
3. Connect the ground power unit to the airplane.
4. Make sure the main gear door linkage rods are properly installed between the door and main landing gear.
5. Raise the main landing gear. Make sure that the gear is in locked position.
6. Inspect the door for firm fit against wall and flush fit with lower wing surface. Door must be slightly preloaded to ensure the door does not gap in flight.
7. If the door requires adjustment, extend the gear and adjust the main landing gear door linkage rod.
8. Raise the gear and inspect the door fitting. Repeat this adjustment until the door fits firmly against the wall and fits flush with lower wing surface.
9. Lower the gear to the down-and-locked position.
10. Return electric systems to the OFF configuration.
11. Disconnect the hydraulic service unit and external electrical power.
12. Lower the airplane from jacks.

NOTES

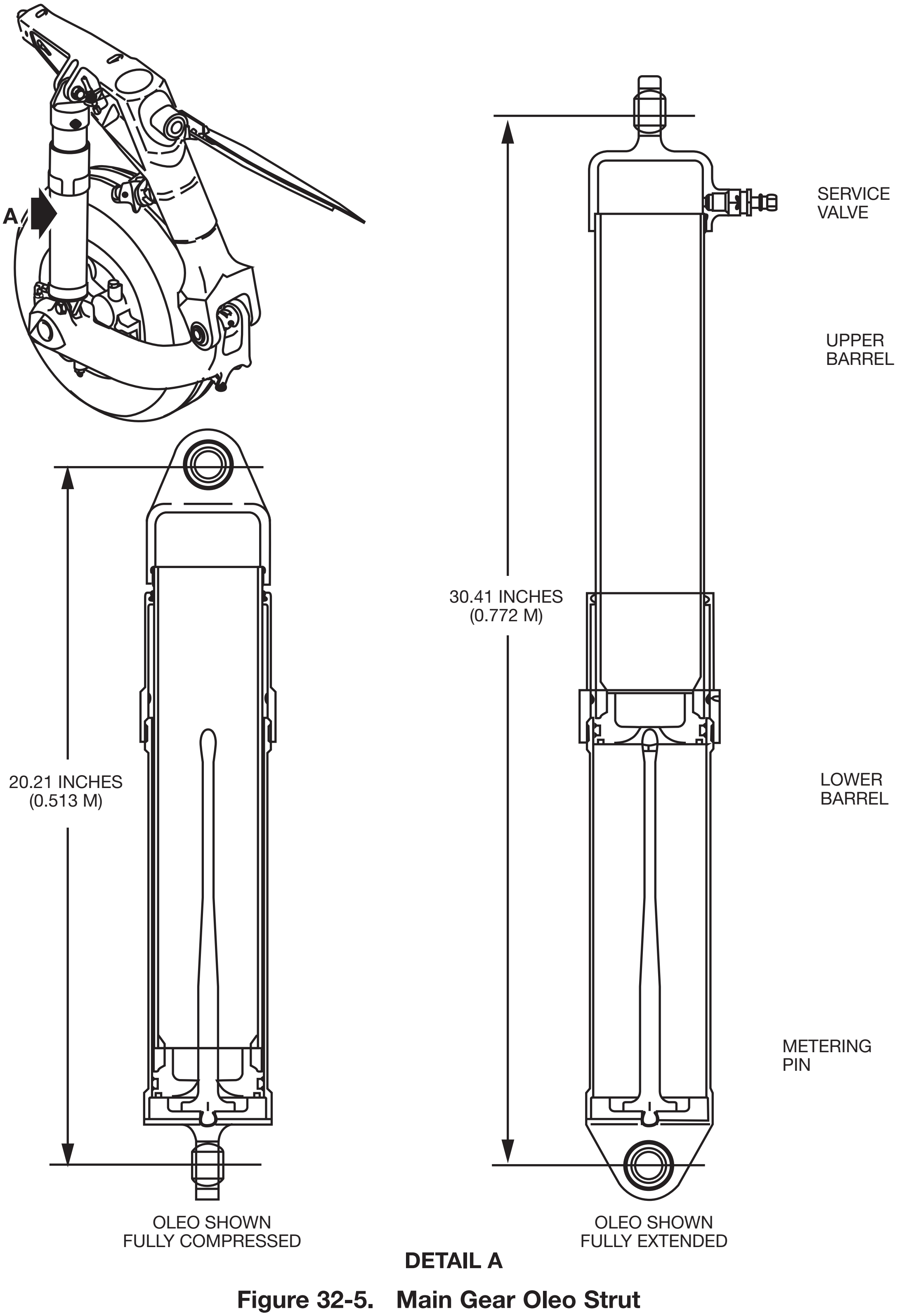


Figure 32-5. Main Gear Oleo Strut

Main Gear Oleo Servicing

The main gear oleo servicing is described below. Refer to Figure 32-5.

1. Jack the airplane until the tires clear the ground.

WARNING

High-pressure air is dangerous. Personnel must thoroughly understand required safety precautions when handling high-pressure air as outlined in Chapter 36—"Pneumatics."

2. Open the service valve and deplete pressure from the oleo. Remove the safety wire and service valve when pressure is released.
3. Connect the service hose from the hand pump service unit (containing approved phosphate ester hydraulic fluid) to oleo.
4. Open the check valve on the hand pump and slowly force the oleo into the collapsed position using a hydraulic jack.
5. Close the check valve on the hand pump and pump fluid into the oleo until it is fully extended.
6. Repeat the steps above a minimum of four times.
7. Finish the bleeding process with the oleo fully collapsed.
8. Disconnect the hand pump service line and install a service valve with new packing (O-ring.)
9. Safety wire the service valve.

NOTE

Keep oleo full of fluid and do not release from compressed position until the service valve has been installed. The fluid level must be at the service valve hole with the oleo in the fully compressed position.

10. Connect a nitrogen source to the service valve with a gauge/adapter assembly and service the oleo to 397 psig. Refer to Chapter 36—"Pneumatics" for servicing high-pressure gases through the service valve.
11. Slowly bleed off pressure to 297 psig, then remove the gauge/adapter assembly. Install the dust cap.

CAUTION

When replacing the main gear oleo strut, check for proper part number. The oleo strut for the excel aircraft looks identical to the oleo used on other citation models.

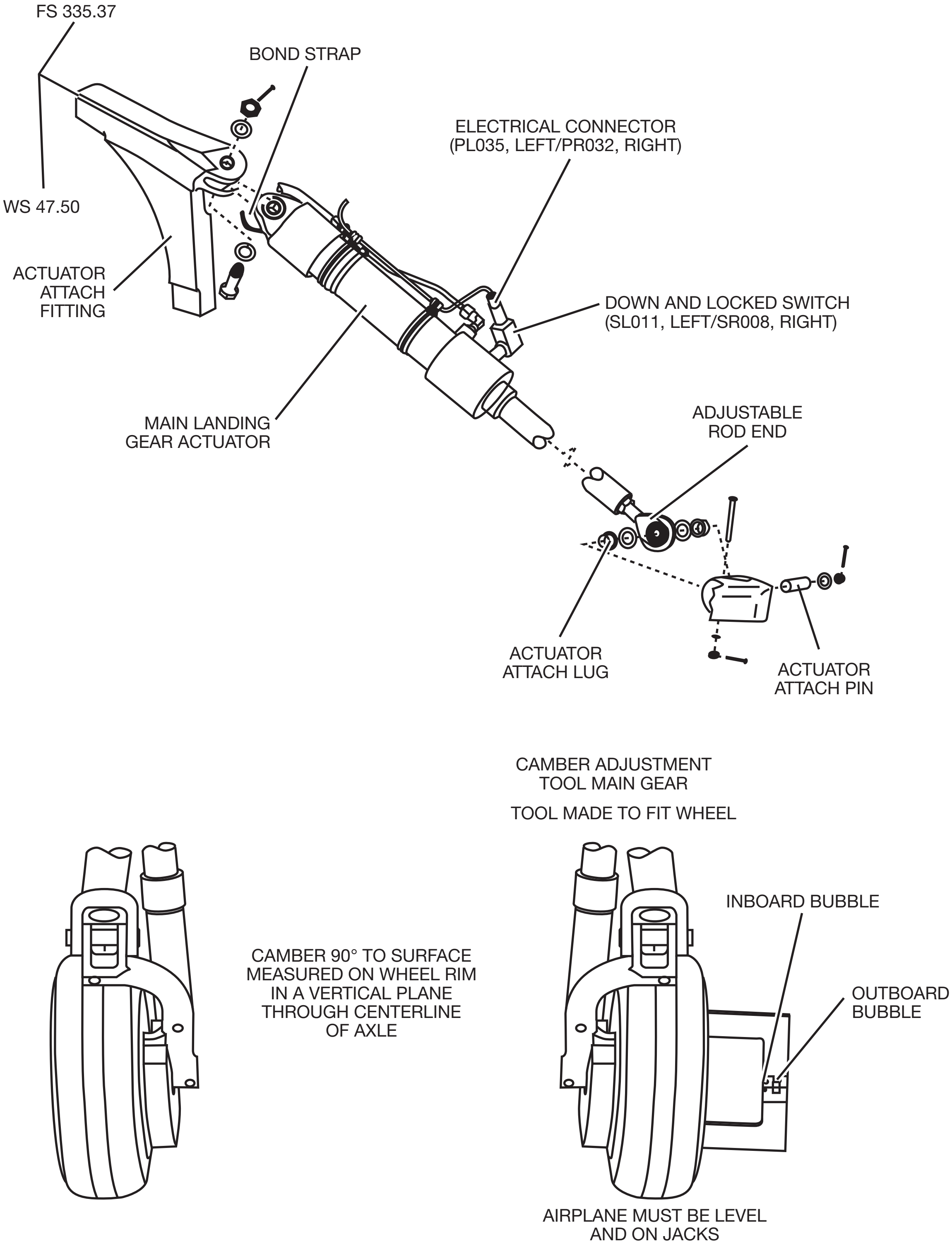


Figure 32-6. Main Gear Actuator

Main Landing Gear Actuators

The main landing gear actuators are inboard of the left and right main gears (Figure 32-6). The actuators are normally actuated hydraulically. During auxiliary/ emergency extension of the landing gears, the actuators are actuated pneumatically, with the use of a shuttle valve. When the actuator extends, the gear extends to the down-and-locked position.

The main landing gear actuator has an integral locking mechanism to lock the landing gear in the fully extended position. There is a spring-loaded plunger incorporated in the locking mechanism that presses inward (toward the actuator) on a key-lock. When the gear and actuator reaches full extension, the key-lock moves into a groove in the actuator piston and the plunger moves over the key-lock. When a hydraulic pressure of 300 psi (+75 or –50 psi) is applied to the actuator retract port, hydraulic pressure overcomes the plunger spring. This allows the plunger to move away from the key-lock, permitting the key-lock to expand out of the actuator piston groove to unlock the actuator for retraction.

An electrical down-and-locked indicating switch is on the top of the actuator. The switch is actuated by the locking plunger. Through electrical circuits it indicates to the landing gear control system and the pilot that the respective landing gear is down-and-locked.

Camber Adjustment

The main gear trunnion camber adjustment is accomplished by adjusting the actuator rod end.

NOTE

Incorrect wheel camber causes tires to wear unevenly, and drastically shortens tire life.

Adjust and /or Verify Camber as Follows

1. Jack the airplane.
2. Level the airplane.
3. Locate tool CJMDL32-001 on the outboard side of wheel (rim). Center the inboard bubble on the camber adjustment tool.
4. Determine the correct camber (1.5 degrees) by verifying inner and outer bubbles are centered.
5. If bubbles are not centered, remove the safety wire. Loosen the jamnut on the actuator rod end.

CAUTION

Ensure that there is sufficient thread engagement of the rod end into the actuator shaft. Engagement length must have sufficient threads to cover inspection hole.

6. Using wrench flats, rotate the shaft on the rod end to obtain a centered bubble indication.
7. Tighten the jamnut and install the safety wire.
8. Verify that inner and outer bubbles are centered on the camber adjustment tool CJMDL32-001. If not, readjust per above instructions.
9. Lower the airplane from jacks.

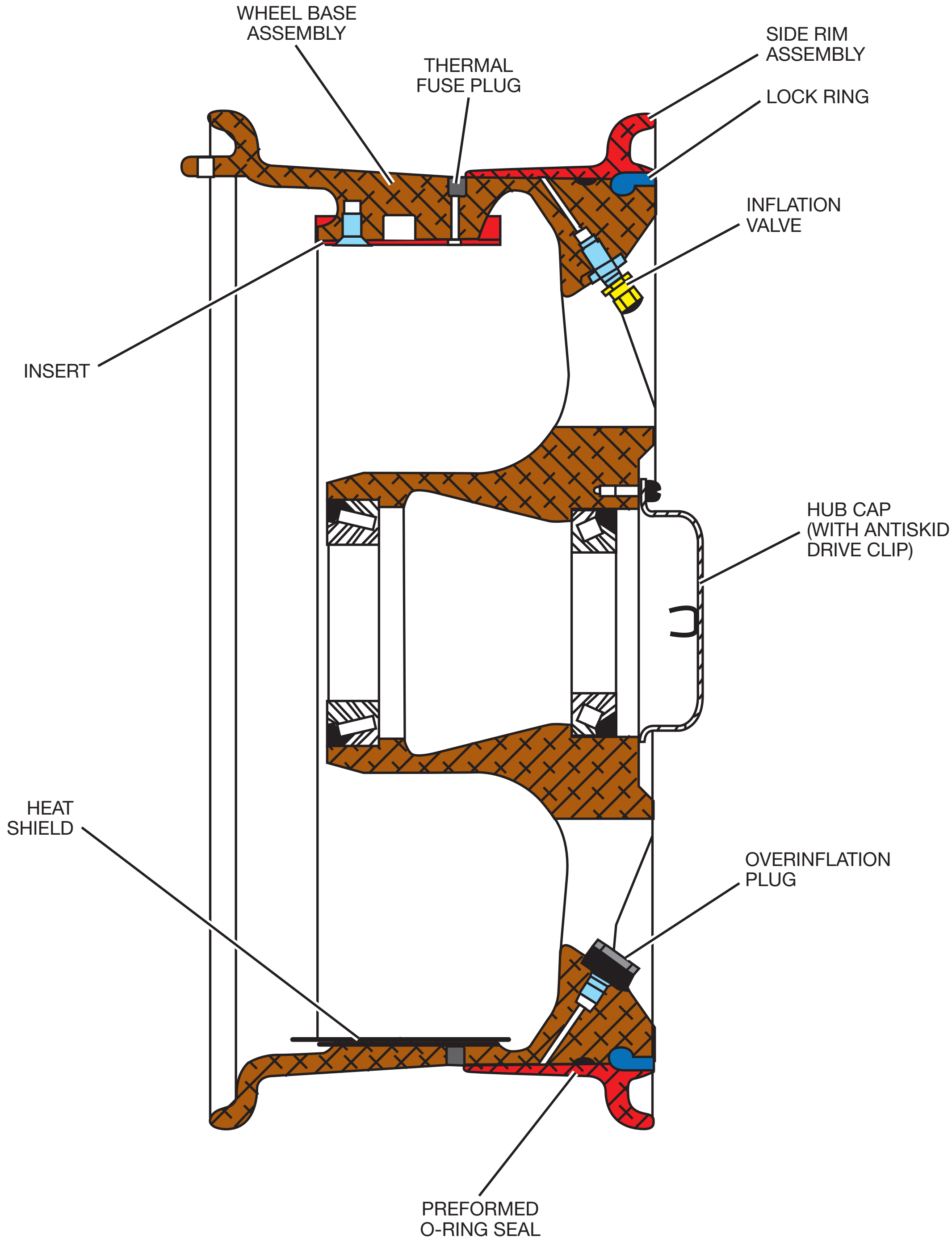


Figure 32-7. Main Gear Wheels

Main Gear Wheel

The main landing gear wheels are a tubeless lock-ring type assembly constructed of forged aluminum (Figure 32-7). Each wheel assembly consists of a wheel base assembly and a side rim assembly that are secured together with a lock ring. There is an inflation valve in the wheel base (outer) assembly. The wheels are designed for use with 23.5 x 8.0R12 tubeless 14-ply rating tire.

The main landing gear wheel base assembly and side rim assembly are sealed when assembled together by a preformed O-ring packing (in the mating surface of the wheel base assembly).

The main landing gear wheel rotates on two tapered bearings (cones and cups). The bearings are protected from dirt, moisture, and other contamination by seals on the outboard side of the inner/outer bearing assemblies. There is a hub cap on the wheel base (outer) assembly. The hub cap (with antiskid system drive clip) is attached to the wheel base assembly with screws.

There are heat shields in the wheel base assembly at the torque lugs to prevent heat (generated by the brake assembly) from damaging the wheel and tire assembly.

CAUTION

See vendor manual for approved cleaners. Flammable solvents must not be used for wheel and brake cleaning, since it becomes trapped in heat shields and causes wheel fire during brake application.

Inserts are installed over bosses on the inner side of wheel base assembly. The inserts engage slots in the brake assembly disks to rotate the brake disks as the wheel rotates.

An overinflation plug is installed 180° from the inflation valve on the wheel base assembly. Thermal fuse plugs are installed in the drive lugs of the wheel base assembly. The thermal fuse plugs melt at a predetermined temperature to prevent tire over-inflation caused by a brake overtemperature condition.

The main gear wheel assembly has inspection requirements at specific intervals. For inspection requirements and for more detailed information concerning the main landing gear wheel, refer to BF Goodrich Wheel and Brake publication.

NOTES

Main Gear Tire

The airplane utilizes a size 23.5 x 8 or 12 tubeless 14-ply rating tire (Table 32-1).

Tire Tread Examination

Examine tires with the following criteria for mounted tires:

- Uneven wear
- Cuts
- Bulges
- Fabric fraying/groove cracking
- Flat spots
- Bead damage

Tire preventive maintenance is outlined in the tire manufacturer's publication.

Servicing

Servicing the tire by maintaining correct inflation pressure is the most important job in any tire preventative maintenance program. Improper inflation pressure causes uneven tread wear.

Underinflation, indicated by excessive wear in the shoulder area, is particularly severe. It increases the chance of bruising sidewalls and shoulders against rim flanges. In addition, it shortens tire life by permitting excessive heat buildup.

Overinflation is indicated by excessive wear in the center of the tire. This condition reduces traction, increases tire growth and makes treads more susceptible to cutting.

Servicing the tire(s) requires maintenance personnel to handle compressed gas. Observe safety precautions.

Safety Precautions:

Allow the tire and brake to cool before attempting to service.

WARNING

Introducing relatively cooler nitrogen into a tire that is hot or when the brakes are hot may cause the tire to burst.

Stand at a 90° angle to the axle along the centerline of the tire during servicing.

WARNING

The tendency of a bursting tire is to rupture along the bead. Standing in front of either bead area could cause injury should the tire burst.

CAUTION

Applying a tire sealant on the tire may cause wheel corrosion.

Follow all local safety and technical directives while servicing tires.

Table 32-1. MAIN GEAR TIRE PRESSURE

TIRE PRESSURE	CORRECTIVE ACTION
>220 psig	Adjust pressure to max of correct pressure range
213–220 psig	None–correct pressure
201–212 psig	Adjust pressure to max of correct pressure range
191–200 psig	Adjust pressure to max of correct pressure range-recheck pressure after 24 hours
0–190 psig	With rotation-replace tire
0–190 psig after 24 hours	Check tire for damage-adjust pressure to max of correct pressure range-recheck pressure

Procedures:

Check tire pressure regularly.

Tire pressures should be checked with an accurate gage on a regular basis (daily, if airplane is operated daily). When practical, pressures should be checked before every flight.

Check only cool tires at least two to three hours after a flight. Use an accurate gage. Inaccurate gages are a major cause of improper inflation.

Use the recommended tire pressure.

CAUTION

Refer to Chapter 32—"Wheels—Troubleshooting," Table 101, when tire pressure falls below recommended limit, to determine proper corrective action.

Main Gear Tire Pressure:

Main gear tire pressure should be maintained at 210 PSIG, +2 or –5 PSIG (1,448 kPa, +14 of –34 kPa) unloaded, 218 PSIG, +2 or –5 PSIG (1,503 kPa, +14 or –34 kPa) loaded.

NOTE

The operating pressures are to be measured with the weight of the airplane on the wheels.

Adjust tire pressures for climate change.

Climate changes have an effect on tire pressure when flying from a hot climate to a cool climate and vice versa. When temperature change is extreme (changes in excess of 50°F) for example, a tire inflated/utilized in a warm climate will have a drop in air pressure when the airplane on which it is installed is flown to a cold climate; bringing an airplane out of a heated hangar into the cold winter will do the same.

In either circumstance, tires should be overinflated to compensate for the subsequent cooling and loss of pressure caused by extreme temperature changes. As a general rule, an ambient temperature change of 5°F produces a pressure change of about one percent.

Serviceing Characteristics

Loss of Tire pressure:

A slight amount of diffusion through the carcass in tubeless tires is normal. The sidewalls are purposely vented in the lower sidewall area to bleed off the diffused air preventing separation or blisters.

A tire may lose as much as 5% of the initial inflation pressure in a 24-hour period and be considered normal. A tire with an abnormally high leak-down rate shall be replaced. Applying an unapproved tire sealant to the tire may cause wheel corrosion or cause an out of balance condition.

Above Normal Brake Energies Have Been Exceeded (Rejected Takeoff or Emergency Braking).

Even though inspection may show no apparent damage, the tires may have sustained incipient damage that could result in premature failure.

Also, wheels shall be checked using the applicable wheel overhaul manual.

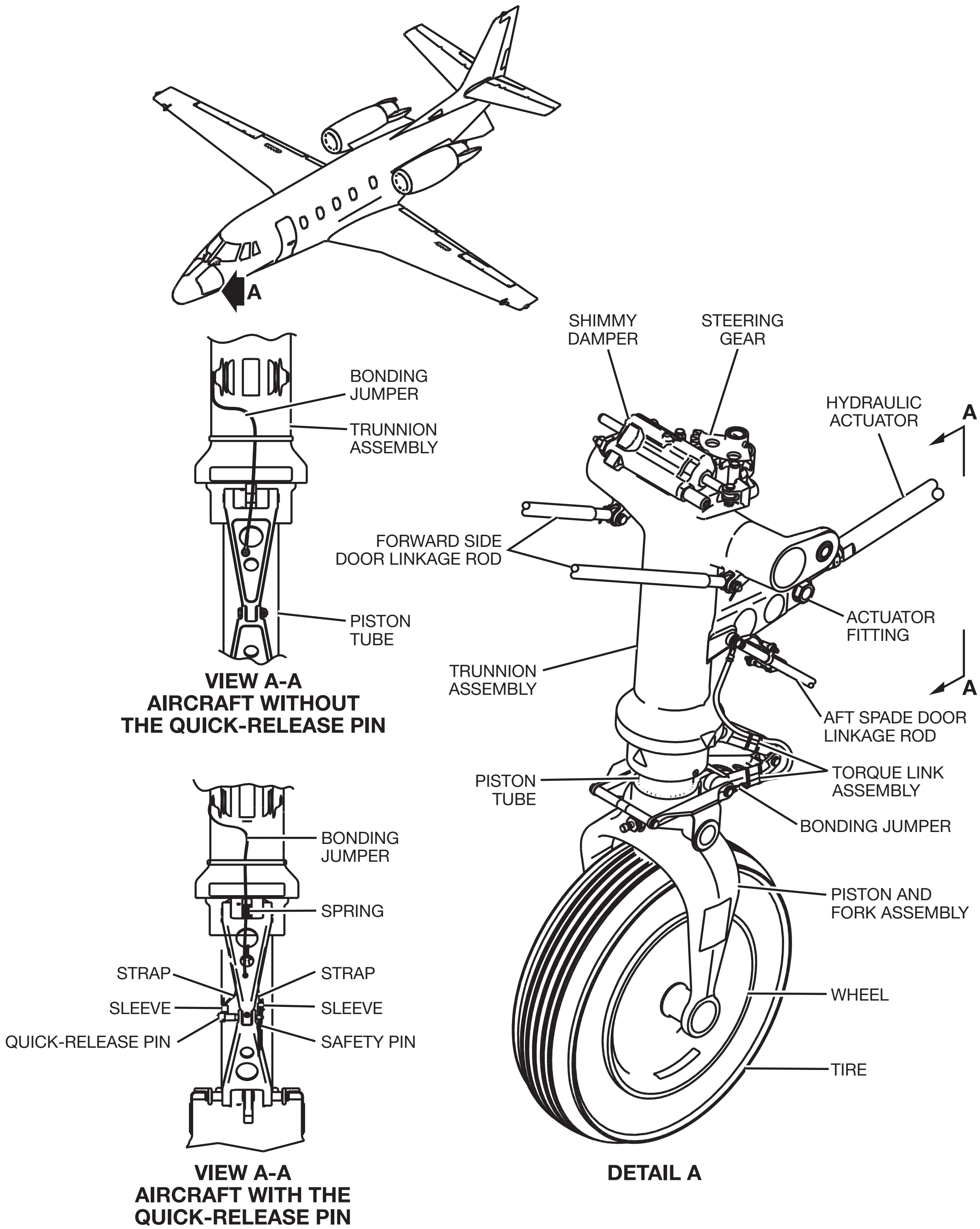


Figure 32-8. Nose Landing Gear

NOSE LANDING GEAR AND DOORS

DESCRIPTION

The nose landing gear assembly consists of a wheel and tire assembly attached to a shock strut and suspended by a trunnion (Figure 32-8). The nose landing gear assembly is hydraulically actuated and retracts forward. The nose gear wheel is automatically centered in the nose wheel well. Three attach points secure the nose landing gear in the airplane nose section: one attach point at each end of the trunnion and one at the hydraulic actuator attach fitting. The trunnion rotates on bearings.

The nose landing gear has a mechanical latching mechanism in the retracted position. The hydraulic actuator has an internal downlock provision in the gear extended position. The gear does not require hydraulic pressure, overcenter linkage, or springs to hold it in the down- and-locked position. Hydraulic pressure is required to unlock the internal downlock. The nose landing gear is utilized for steering and towing the airplane. There is a steering gear assembly on top of the trunnion to provide a tow turning radius stop.

Three doors enclose the nose landing gear wheel well when the nose landing gear is in the retracted position. Two forward doors cover the forward portion of the nose wheel well and the spade door covers the aft portion. The two forward doors are linked to a torque shaft and control rod assembly and are attached to the nose gear trunnion. The doors are open when the nose landing gear is fully extended and closed when the gear is fully retracted.

NOTES

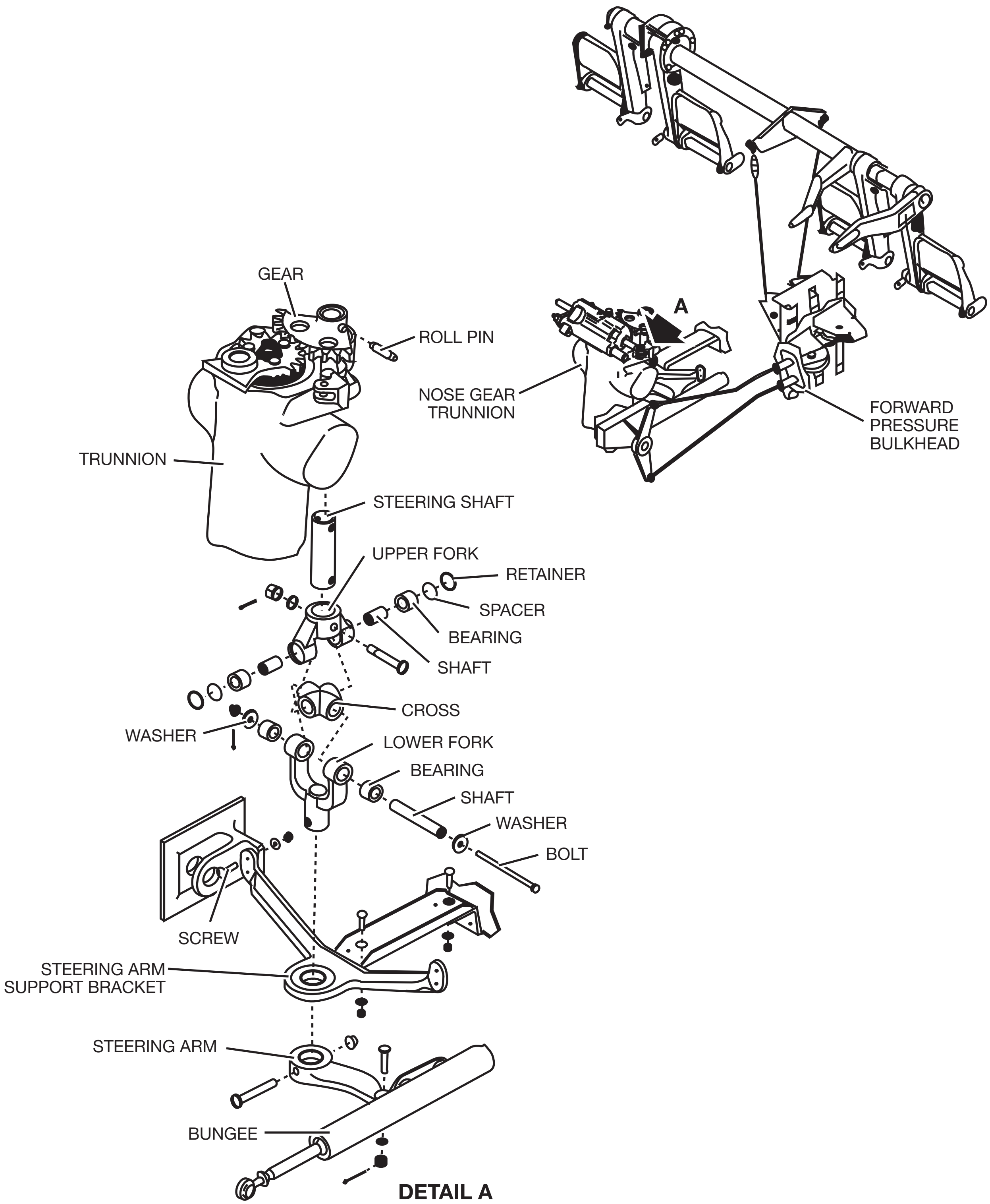


Figure 32-9. Nose Gear Steering

COMPONENTS

Torque Links

The torque links attach the trunnion assembly to the piston and fork assembly. The torque links keep the nosewheel aligned with the other components of the nose gear assembly. The torque link assembly is attached to the trunnion assembly and the piston and fork assembly with pins The top and bottom torque links are attached together with a bolt, a nut, a cotter pin, and washers. On airplanes that incorporate SB560XL-32-20, the top and bottom torque links are attached together with a quick-release pin and a safety pin. Each of the attach points on the torque link have a grease fitting for lubrication.

Nose Gear Steering

Steering the airplane while on the ground is accomplished by cables that are attached to the rudder pedals at one end and to a bellcrank bungee at the other end. The bungee is a spring-loaded rod which transmits steering control to a steering arm, universal joint and steering gear atop the nose gear strut. The universal joint automatically centers the nose gear during gear retraction (Figure 32-9).

NOTES

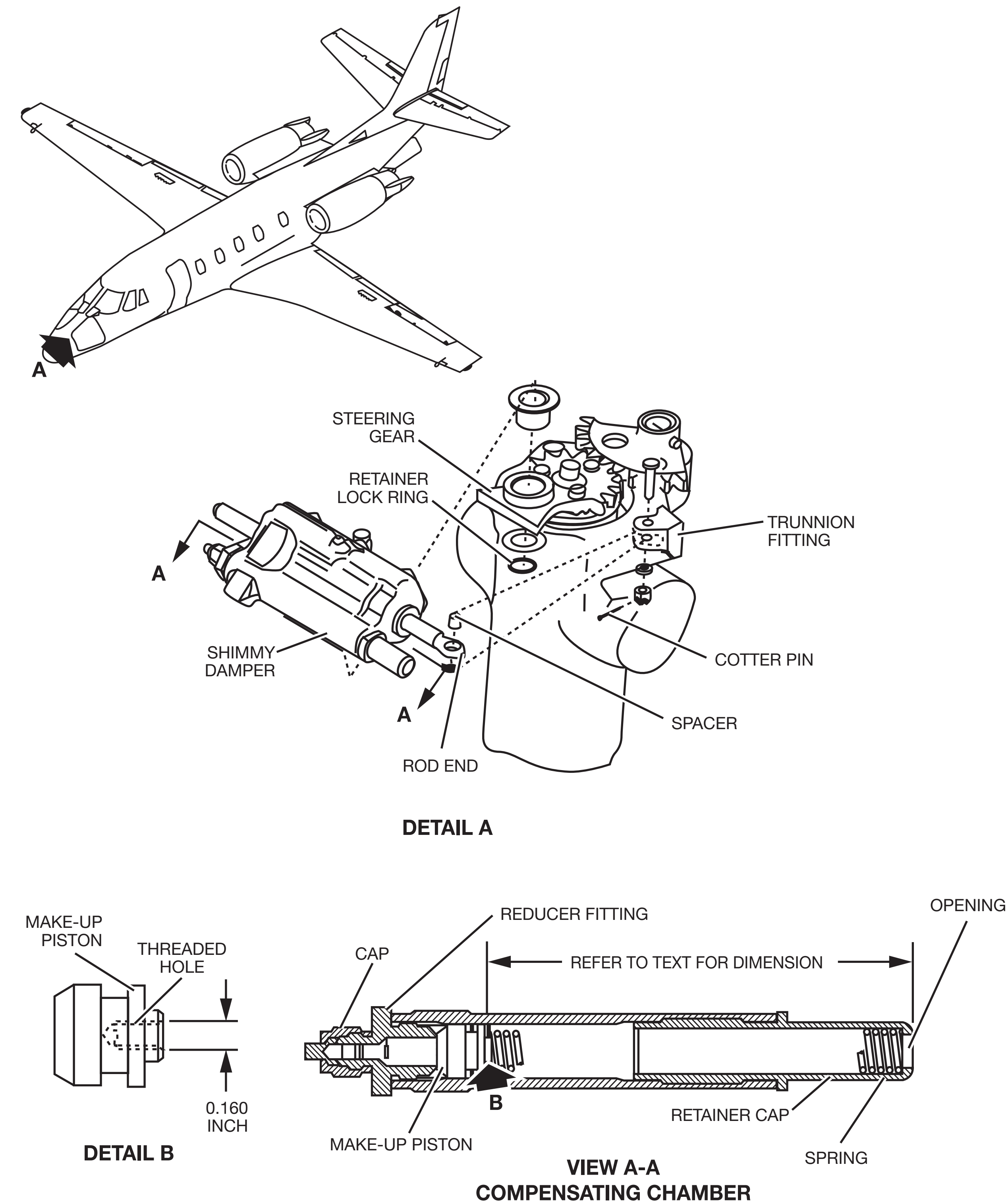


Figure 32-10. Shimmy Damper

Shimmy Damper

The nose landing gear shimmy damper incorporates a piston with an orifice to restrict fluid movement when the piston moves through a hydraulic fluid filled cylinder (Figure 32-10). The restricting action dampens rapid movements of the nosewheel. The damper also incorporates a compensating chamber which consists of:

- A makeup piston
- A spring
- A check valve
- A relief valve

The chamber is utilized to relieve thermal expansion of the hydraulic fluid.

The shimmy damper is on the nose landing gear steering gear with the shimmy damper rod end attached to the trunnion.

When the shimmy damper malfunctions or excessive leakage is noted, remove the shimmy damper from the nose landing gear. Refer to the Model 560LXL *Component Maintenance Manual,* Shimmy Damper, Chapter 32, for maintenance and repair.

WARNING

Observe all safety precautions while working on hydraulic systems or units that contain phosphate ester base fluids. Prolonged exposure can cause skin dehydration and chapping.

CAUTION

Observe technical precautions while working on hydraulic systems or units that contain phosphate ester base fluids. The fluid attacks a wide range of materials including rubber, copper, various plastics, and paints.

Shimmy Damper Servicing

Measure the depth of the makeup piston inside shimmy damper by inserting a measuring probe in the open end of the retainer cap. If the measurement is less than 4.5 inches (114.3 mm) or more than 4.6 inches (116.8 mm), service the shimmy damper.

Remove the service port cap.

NOTE

It helps to make sure that there is no unwanted air in the hydraulic service pump.

NOTE

It helps to use a 90° adapter fitting to connect the hydraulic service pump to the service port.

Use a container to catch the spilled hydraulic fluid. Connect a hydraulic service pump with Skydrol to the service port. Refer to Chapter 29, Main Hydraulic System-Maintenance Practices, for safety precautions.

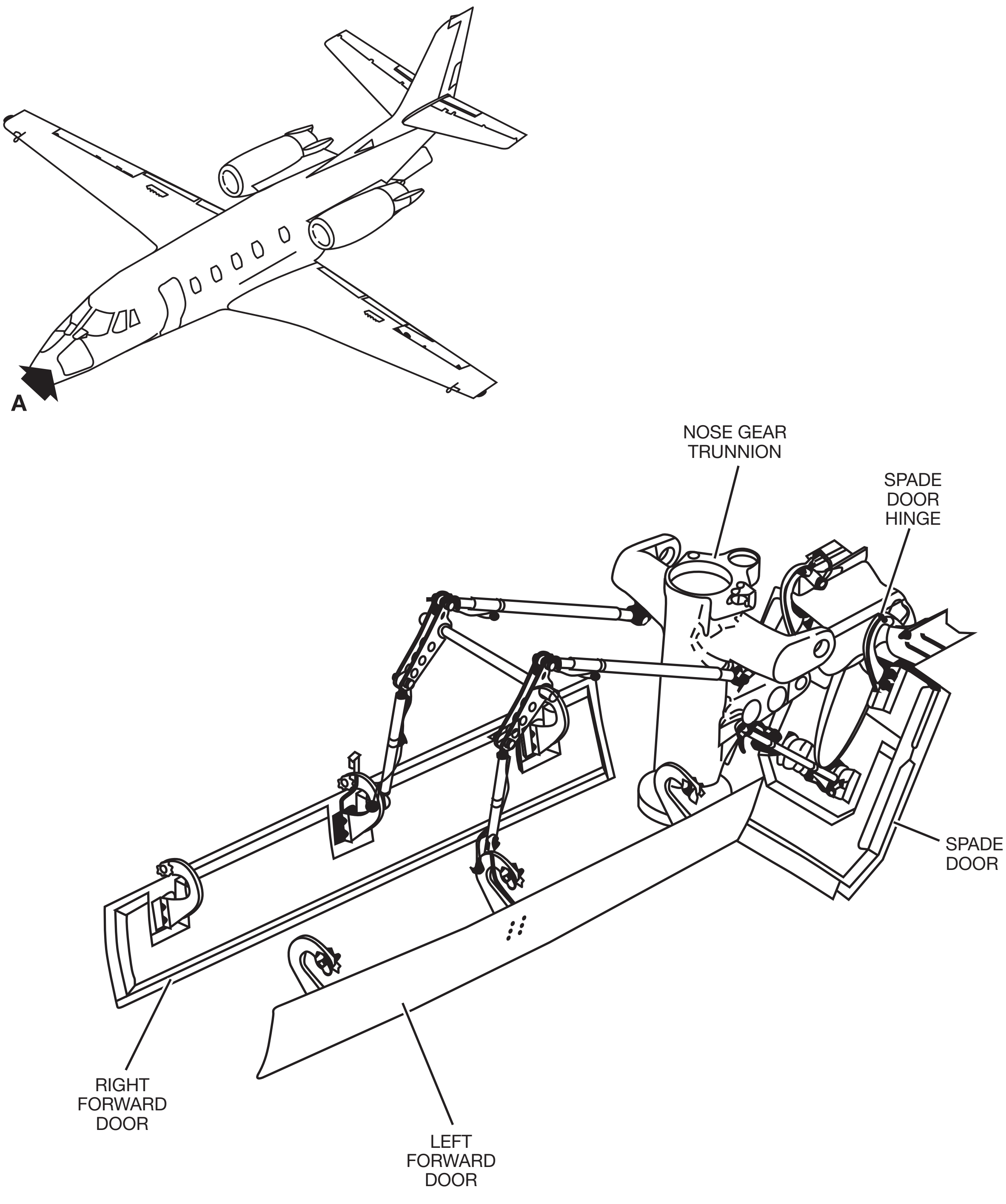
Use a hydraulic service pump to fill the shimmy damper.

NOTE

Fluid can bleed out of the retainer cap during this procedure.

Loosen the bleed-screw as necessary on the shimmy damper to release unwanted air. Tighten the bleed-screw on the shimmy damper. The measurement of the make-up piston must be more than 4.5 inches (114.3 mm) after you bleed unwanted air.

Loosen the bleed-screw as necessary on the shimmy damper. The measurement is between 4.5 inches (114.3 mm) and 4.6 inches (116.8 mm). Disconnect the hydraulic service pump and install the service port cap. Fully flush the retainer cap with alcohol to remove all the unwanted hydraulic fluid.



DETAIL A

Figure 32-11. Nose Gear Doors

Steering

Nosewheel steering is accomplished by cables that are connected to the rudder pedals. Nosewheel steering turning is limited by the rudder stops. The turning limit is approximately 20° either side of center.

The centerline of the steering universal joint is in alignment with the centerline of the trunnion supporting bolts. When the nosewheel is retracted, the lower half of the steering universal joint remains in position; while the upper half, pivoting with the strut moves to the center position, automatically centering the nosewheel. With the nosewheel fully retracted, the upper half of the steering universal joint and the nosewheel remain stationary, while the lower half of the steering universal joint can move freely to permit normal operation of the rudder pedals.

Bellcrank

The bellcrank is in a bracket on the left side of the wheel well. Cables from the rudder pedals connect to the top and bottom ends of the bellcrank with the steering bungee connected to the upper end.

Steering Bungee

The steering bungee is a spring-loaded rod that transmits steering control from the bellcrank to the steering arm. The spring allows the nose gear to turn past the limits of the control cables when the airplane is being towed, up to 90° either side of center with the control lock disengaged, or 60° with control lock engaged.

Steering Universal Joint

The steering universal joint transmits steering control to the wheel. Upon retraction, the steering universal joint automatically centers the nose gear. With the nose gear retracted, the rudder pedals are free to operate without moving the nosewheel.

Steering Gears

The steering gears on top of the trunnion transmit steering control to the strut. Bolts in the trunnion stop the gears when the turning limit is reached. A shimmy damper connected to the front gear prevents steering oscillations.

Nose Gear Doors

Three doors cover the nose landing gear in the retracted position (Figure 32-11). Two forward doors cover the forward portion of the wheel well, and a spade door covers the aft portion. These doors are linked mechanically to the nose gear trunnion. They operate in conjunction with the extension and retraction of the landing gear. The doors are open when the gear is fully extended; and closed when the gear is fully retracted. The left and right forward side doors are opened and closed through linkage rods and a torque assembly. The spade door is attached to the aft side of the trunnion with two linkage rods.

Operation

When the nose landing gear is retracted or extended, the aft spade door follows the gear through the linkage rods. The torque assembly also rotates, causing the forward side doors to open and close. The three-nose landing gear doors are operated through mechanical linkages that are connected to the landing gear. When the landing gear control handle is selected to the up position, the nose gear enters the nose wheel well, followed by the spade door. As the landing gear locks in the retracted position, the two forward doors close and overlap the forward edge of the spade door (completely enclosing the nose wheel well compartment).

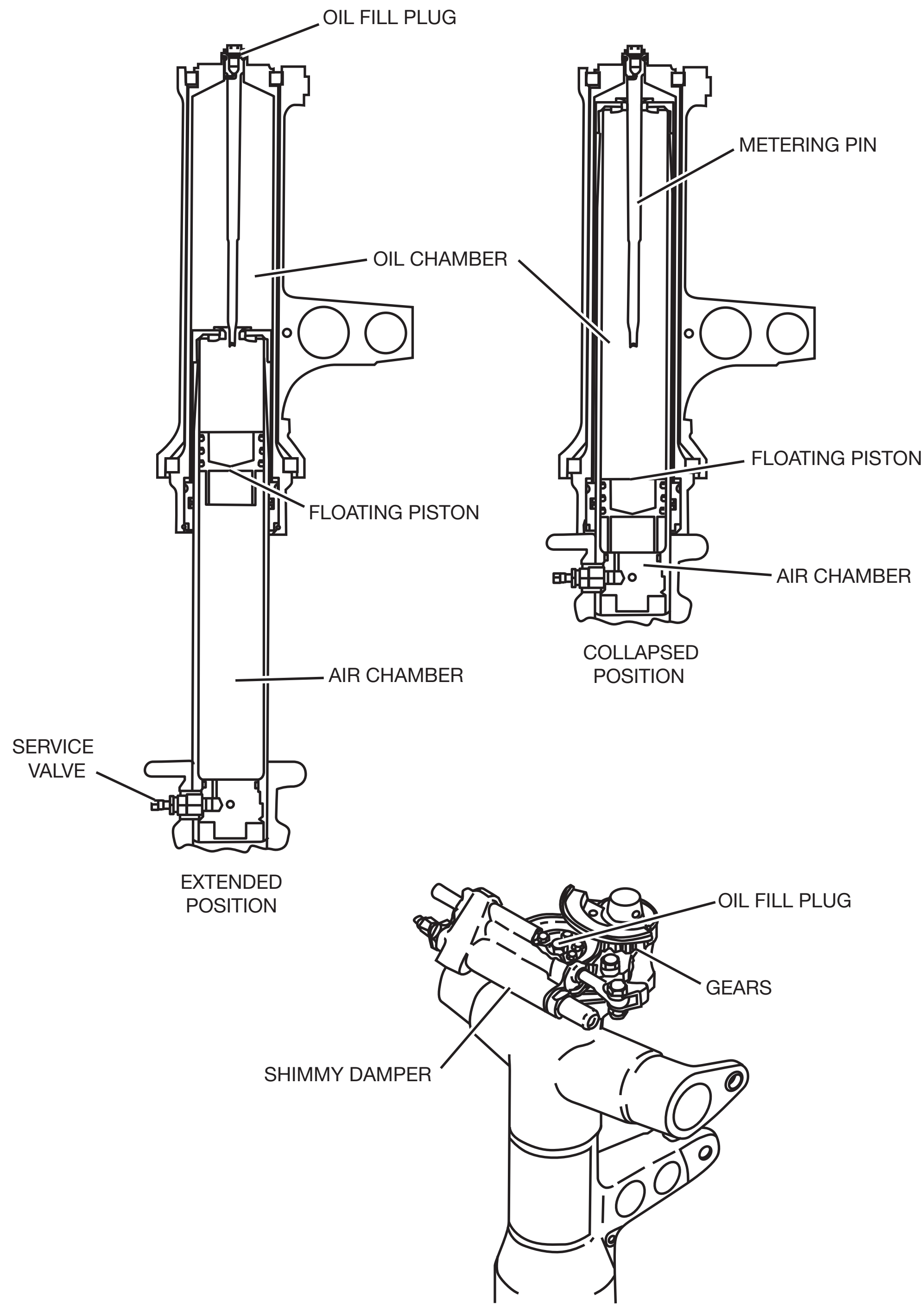


Figure 32-12. Nose Shock Strut

Nose Gear Shock Strut

The nose landing gear shock strut consists of (Figure 32-12):

- Trunnion
- Barrel piston
- Isolation (floating) piston
- Oil filler plug
- Metering pin
- Air filler valve
- Torque link assembly
- Actuator attach fitting

The metering pin extends through an orifice at the top of the piston. As the strut is compressed, the tapered metering-pin progressively decreases the flow of hydraulic fluid through the orifice. As the volume of fluid permitted through the orifice decreases, the movement of the shock strut (piston) is slowed. As the strut is decompressed, the volume of fluid permitted back through the orifice is progressively increased which allows rapid extension of the shock strut.

The shock strut contains a nitrogen chamber positioned below a floating piston. The floating piston separates the hydraulic fluid from the nitrogen chamber and moves up and down within the piston. The compressed nitrogen serves as a cushion to support the airplane and absorb shock.

Shock strut bottoming during normal landings and/or taxiing is evidence of low nitrogen pressure in the shock strut nitrogen chamber.

Diagnostics

Nose Gear Shock Strut (Piston) Extension Check

1. Jack the airplane until the nose gear is clear of the floor.
2. Measure the strut extension dimension.
3. If the dimension is less than 9.10 inches (231 mm) no additional action is required.
4. If the dimension is 9.10 inches (231 mm) or greater, the landing gear must be disassembled and checked for proper security between the bearing/orifice assembly and the piston.

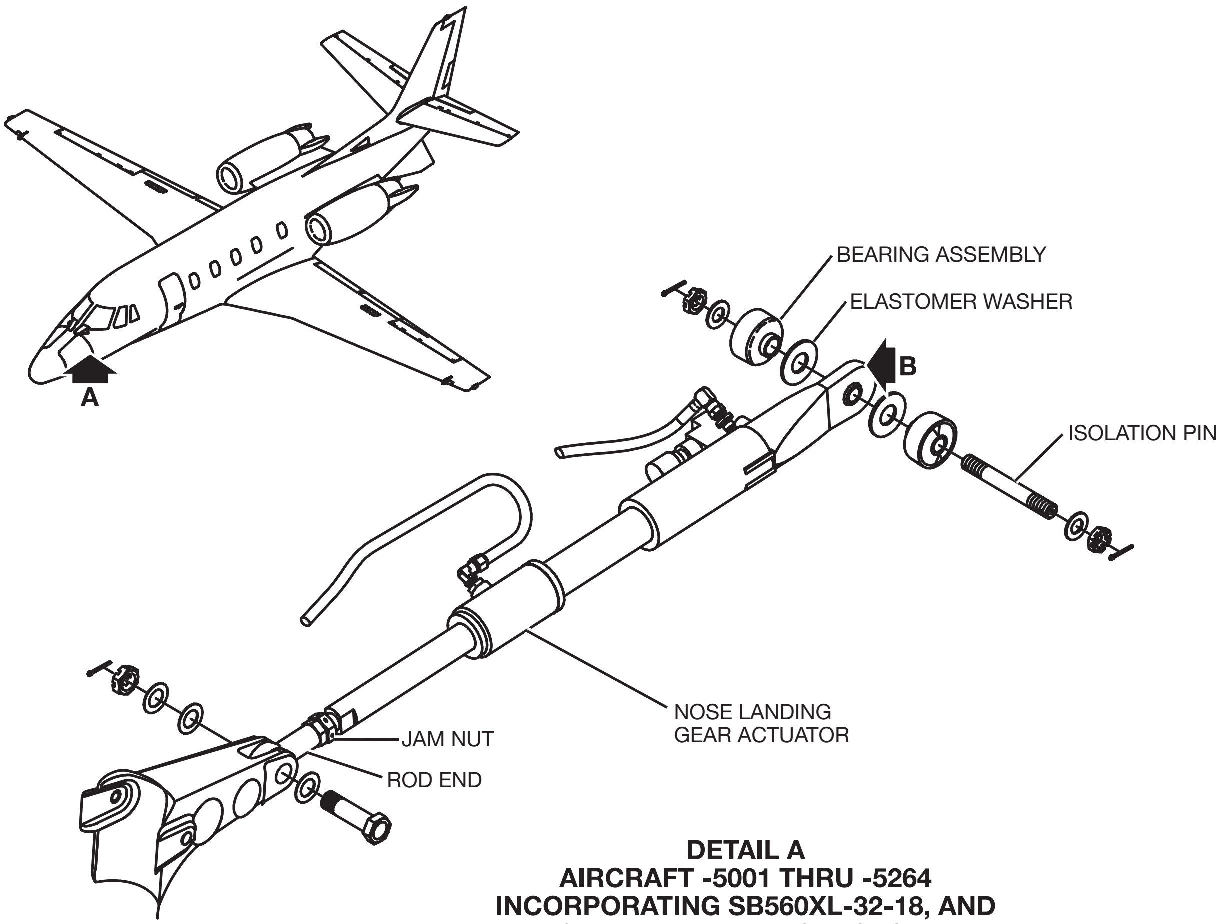
Nose Gear Strut Servicing

NOTE

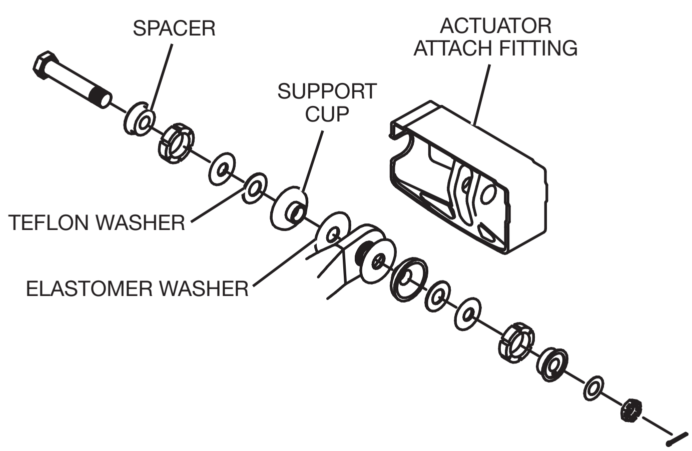
When a small amount of nitrogen is present in the oil chamber, the strut must be serviced; but if a large amount of oil is present in the nitrogen chamber, it is cause for overhaul.

Leaking nitrogen at the filler valve/core and/or faulty floating piston seals is indicated by the presence of excessive oil in the nitrogen chamber. This causes low pressure in the nitrogen chamber.

1. Jack the airplane until the tires clear the ground.
2. Open the service valve and deplete pressure in the lower chamber. Drain any fluid in the chamber by removing service valve.
3. Remove oil fill plug from top of strut.
4. Connect the service hose from the hand pump service unit containing phosphate ester base hydraulic fluid to the top of the strut.
5. Close the check valve on the hand pump and pump fluid into the strut until fully extended and pressure is 200 ± 50 psig.
6. Open the check valve on the hand pump and slowly force strut to the collapsed position using a hydraulic jack.
7. Repeat the steps above until no air is returned to the service unit reservoir.
8. With the strut in the fully collapsed position, disconnect the hand pump service line and install oil fill plug with new O-ring and safety wire plug.



DETAIL A
AIRCRAFT -5001 THRU -5264 INCORPORATING SB560XL-32-18, AND AIRCRAFT -5265 AND SUBSEQUENT



DETAIL B
AIRCRAFT -5002 THRU -5264

Figure 32-13. Nose Gear Actuator

9. Install service valve with new O-ring and safety wire.
10. Connect nitrogen source to service valve with the gauge adapter assembly and service the oleo to 130 psig.
11. Slowly bleed off pressure to 100 psig, remove the gauge/adapter assembly, and install the dust cap.

Nose Landing Gear Actuator

The nose landing gear actuator is aft of the nose gear (Figure 32-13). The actuator is normally actuated hydraulically. During emergency extension of the gear, the actuator is actuated pneumatically through the use of a shuttle valve. When the actuator piston retracts, the nose gear extends to the down-and-locked position.

The nose gear actuator has an internal locking mechanism to lock the gear in the fully extended down position. Operation of the locking mechanism is the same as the main gear locking mechanism, except that locking occurs with the actuator piston retracted.

NOTE

The nose landing gear actuator is encompassed in a rectangle-shaped box that extends aft from the forward pressure bulkhead. A bolt securing the aft end of the actuator is accessed through the actuator tunnel area (under the aircraft).

An electrical down-and-locked indicating switch is on the top of the actuator. The switch, (through electrical circuits) indicates to the landing gear control system and to the pilot that the nose gear is down-and-locked.

NOTES

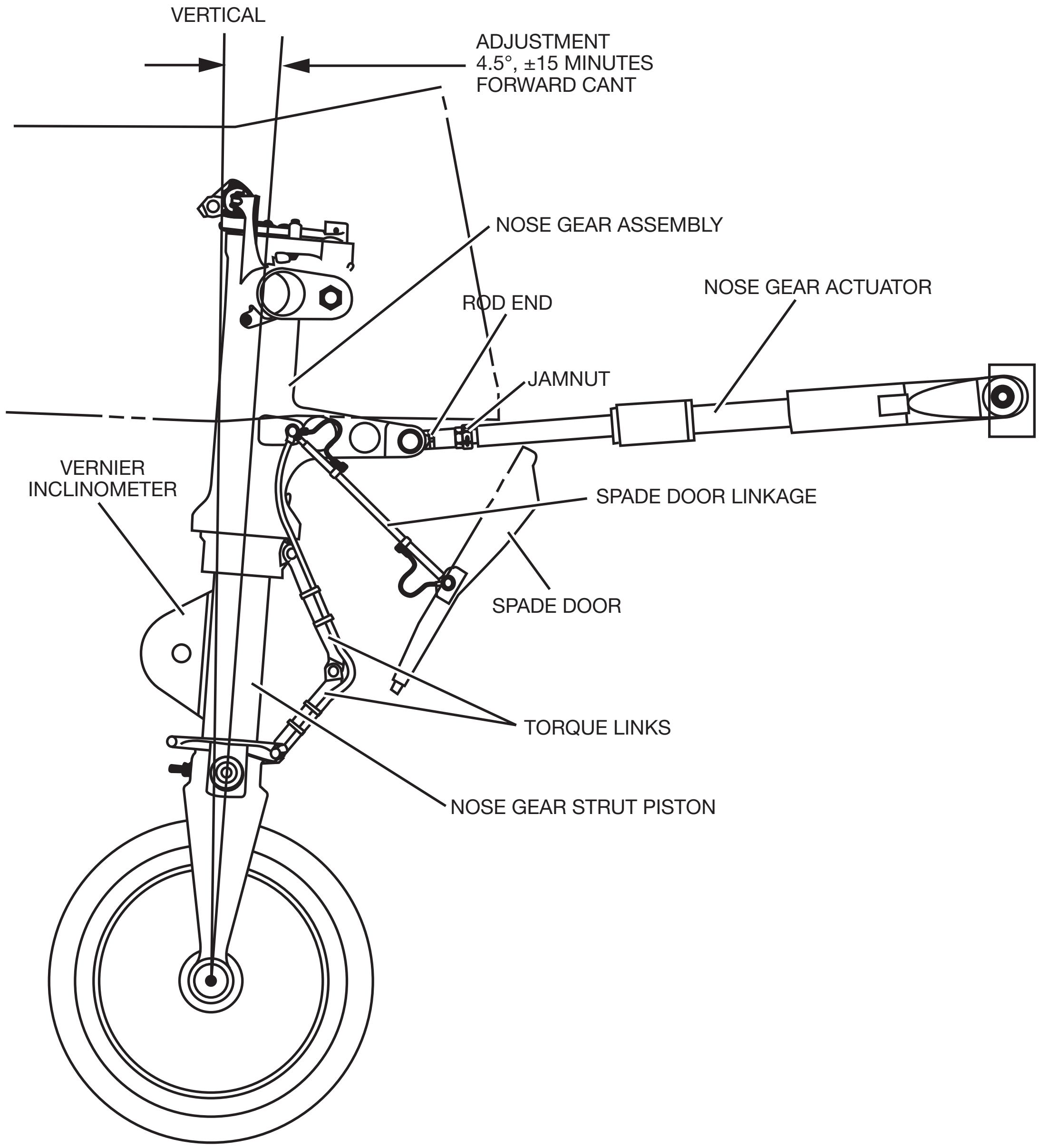


Figure 32-14. Angle Adjustment

Nose Gear Forward Angle Adjustment

Adjust the nose gear strut (piston) to a forward angle of 4.5° in relation to the fuselage station plane (Figure 32-14). The angle is obtained by adjustment of the nose gear actuator rod end.

NOTE

Verify that the nose landing gear is down and locked before adjustment.

1. Jack and level the airplane.
2. Cut the safety wire and loosen the actuator rod end jam nut several turns to free the rod end lock-key.
3. Position and set a Vernier inclinometer (capable of reading degrees and minutes), to the nose gear strut piston surface.
4. Adjust the actuator rod end in/out to obtain 4.5° strut angle forward on the inclinometer.
5. Tighten the actuator rod end jam nut, mating the lock-key with a slot on the rod end while maintaining the adjustment within tolerance.
6. Safety wire the rod end jam nut.
7. Remove the inclinometer.

NOTE

If the nose gear forward angle requires adjustment, the nose gear door-control rod linkages must be adjusted/checked for proper door closing.

8. If adjustment of the door control rod linkage is not required, lower the airplane and remove jacks.

NOTES

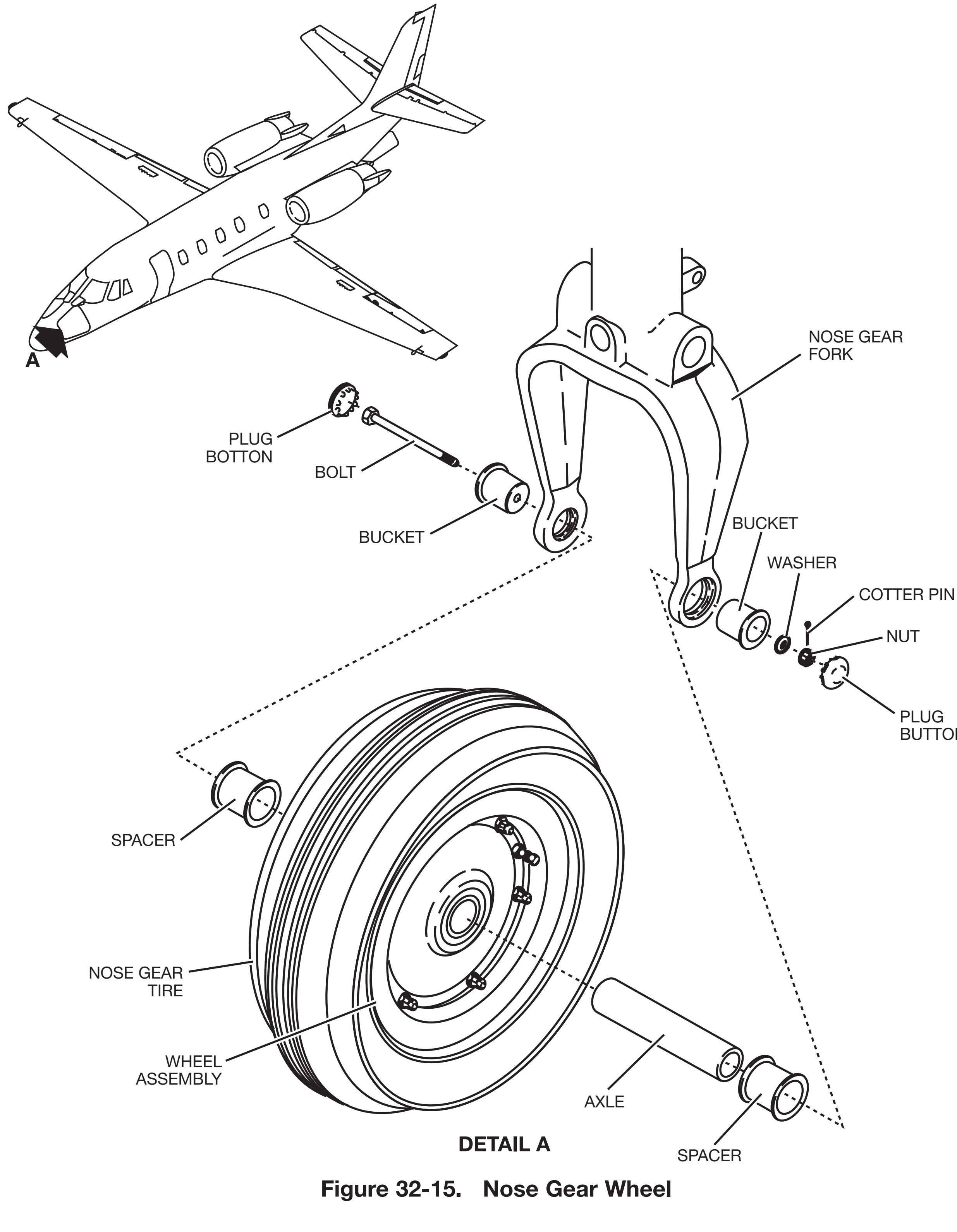


Figure 32-15. Nose Gear Wheel

Nose Gear Wheel

The airplane utilizes a size 18 x 4.4 tubeless 10-ply rating tire.

CAUTION

Use only authorized nose gear tire. Refer to *Model 560XL/ XLS/XLS+ Illustrated Parts Catalog*, Chapter 32.

The wheel is a divided-type wheel to facilitate tire removal and installation (Figure 32-15). The two wheel halves are held together with:

- Bolts
- Washers
- Countersunk washers
- Self-locking nuts

There is an O-ring in a groove on one wheel half that provides an air seal at the junction of the two wheel halves. One wheel half has an inflation valve for inflating or deflating the tire. Each wheel half is individually balanced at the time of manufacture. This permits the wheel halves to be assembled in any position relative to one another and allows the wheel halves to be interchanged without the need for wheel rebalancing.

The wheel rotates on two tapered roller bearings. The bearing cup is shrink-fitted into the hub of each wheel half. The bearings are protected from the dirt and moisture by bearing seals that are built into the bearing.

NOTE

When a tire is removed from a wheel and is to be remounted on the same wheel, the tire must be mounted in the same clock position on the wheel as when removed.

Nose gear tires must be balanced after remounting.

Observe proper inflation and deflation procedures and specified torque values (Table 32-2). Do not use impact or power wrenches to remove or tighten any threaded parts.

The nosewheel has inspection requirements based on the number of tire changes. Therefore, tire changes must be recorded in the airplane maintenance records. Refer to the BF Goodrich Installation, Nose Landing Gear Wheel Assembly Manual—"Inspection" section, for specific inspection requirements and intervals.

At each tire change, the wheel halves must be examined for corrosion. Corrosion may be removed, primed and touched up. For tire removal and/or installation, and for other detailed breakdown and maintenance information concerning the wheel and tire assembly, refer to BF Goodrich Installation—Nose Landing Gear Maintenance and Overhaul Manual.

Table 32-2. NOSE TIRE PRESSURE

TIRE PRESSURE	CORRECTIVE ACTION
>140 psig	Adjust pressure to max of correct pressure range
130–140 psig	None–correct pressure
123–129 psig	Adjust pressure to max of correct pressure range
117–122 psig	Adjust pressure to max of correct pressure range-recheck pressure after 24 hours
0–116 psig	With rotation-replace tire
0–116 psig	Check tire for damage-adjust pressure to max of correct pressure range-recheck pressure after 24 hours

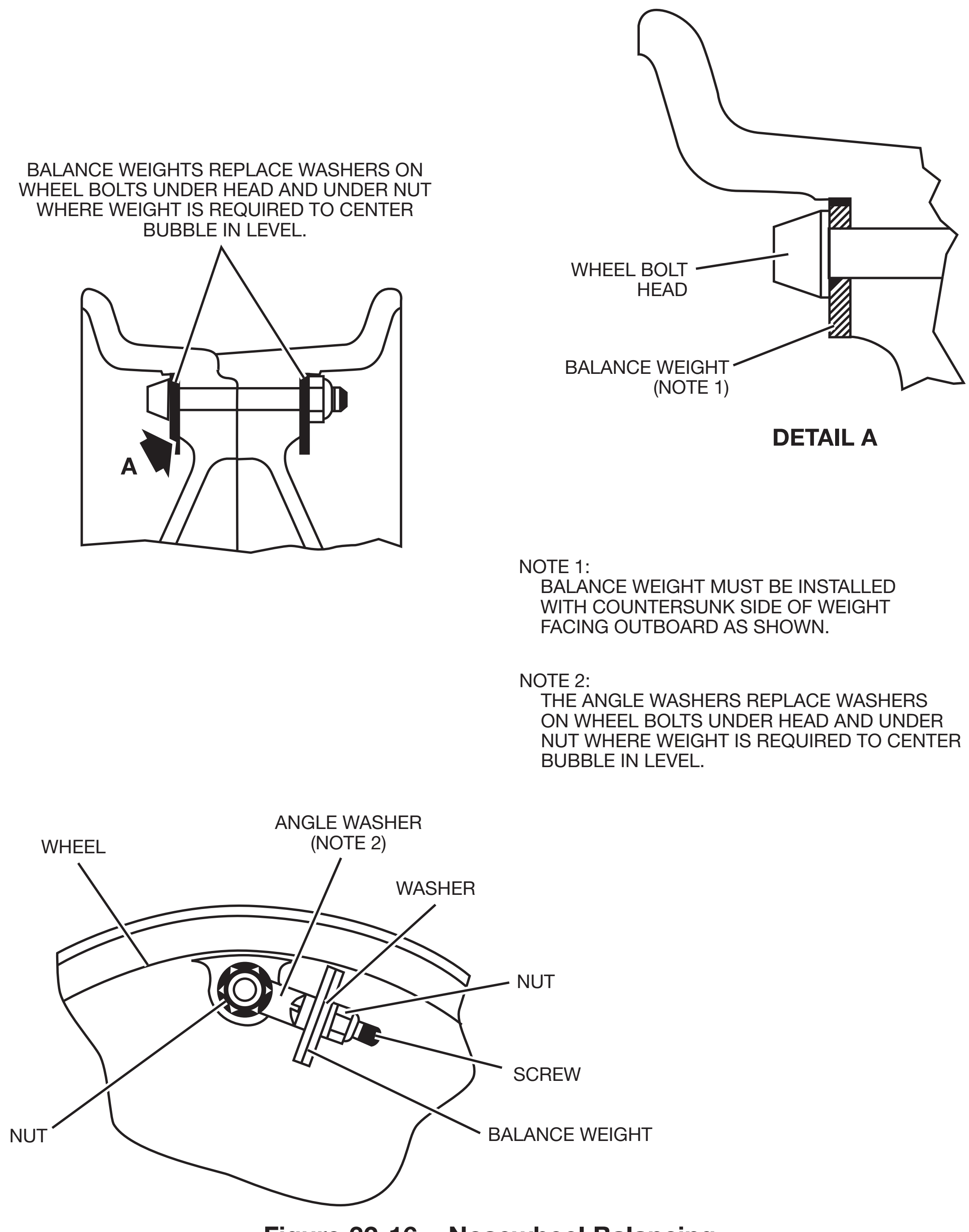


Figure 32-16. Nosewheel Balancing

Servicing

Refer to pg 32-17 for tire servicing.

Nose Gear Tire Pressure.

Nose gear tire pressure should be maintained at 130 PSIG, +5 or –5 PSIG (896 kPa, +34 or –34 kPa) unloaded, 135 PSIG, +5 or –5 PSIG (930 kPa, +34 or –34 kPa) loaded.

Nose Landing Gear Wheel Balancing

A wheel balancing kit (containing weights) is available to statically balance the nose landing gear wheel assembly. Kit 5002532, the weight mounts flush against the wheel half (Figure 32-16).

Part numbers of the individual weights in the 5002532 kit and associated Cessna part numbers are listed in the following table:

Weight Reference	Part Number
1. 5002533	6241108-24
2. 5002534	6241108-25
3. 5002535	6241108-27
3. 5002536	6241108-26

Four balance weights (three, half-ounce weights and one, three-quarter-ounce weight) are available for balancing the wheel and tire assembly. The weights, when installed, replace the washer (s) under the tie bolt head and (if a second weight is used at a given tie bolt) under the tie bolt nut. Weights may also be placed at two adjacent tie bolts, either under the bolt head or under the bolt head and nut.

The nose wheel must be balanced within 2.0 inch-ounces (0.014 N.m) of perfect balance. This means that if, on checking wheel balance, even a single half-ounce weight improves the balance of the wheel, it should be installed. (One half-ounce multiplied by 4.3 inches (109.22 mm)–the diameter from axle center to tie bolt center–equals 2.15 inch-ounces (0.015 N.m).

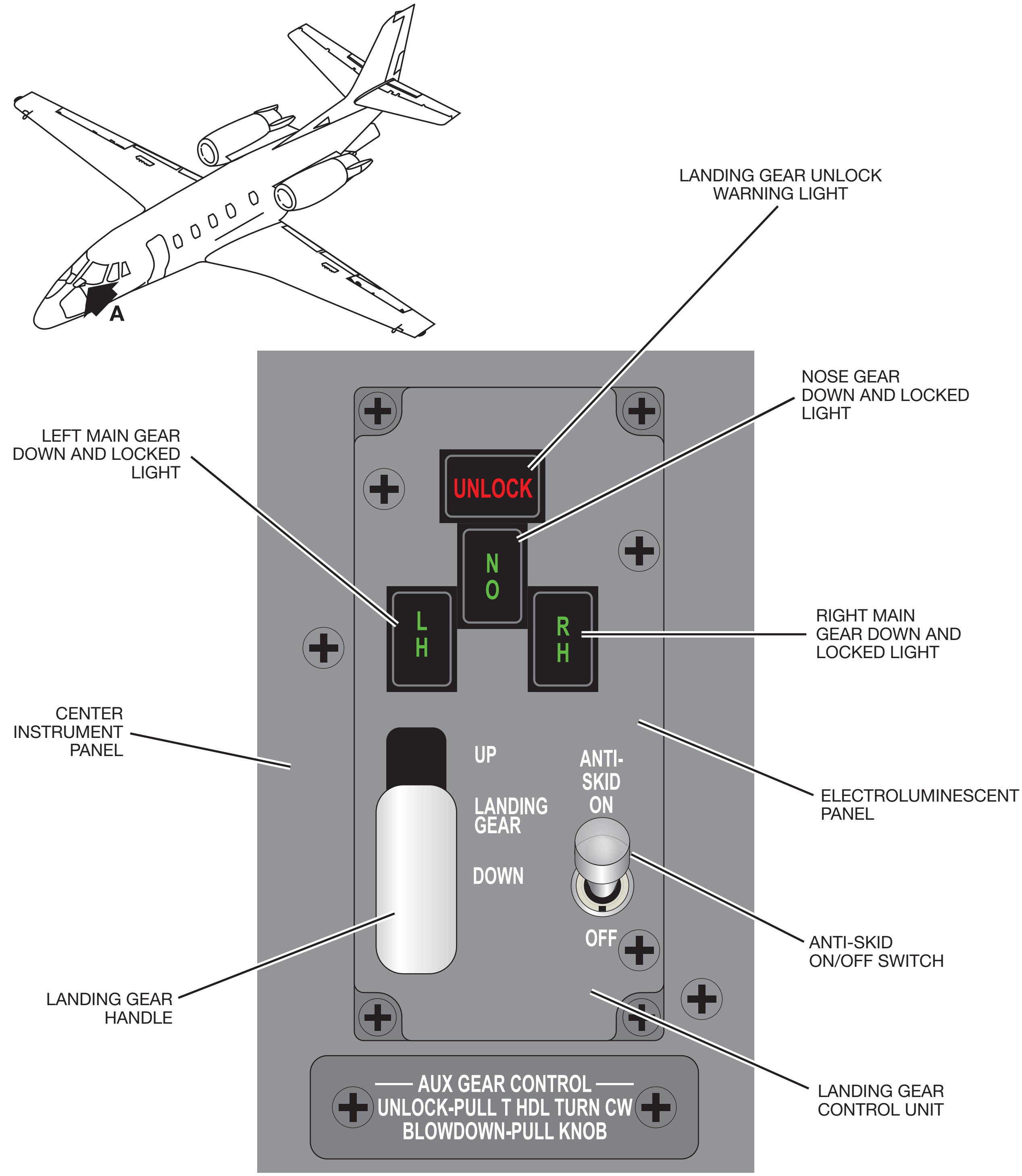
Under the tie bolt head the weight must always be installed with the weight designation facing out; the countersunk hole of the weight receives the tie bolt head, replacing the countersunk washer.

CAUTION

Permanent wheel balance weights (installed inside the wheel halves) should not be removed for assembly balance.

The nose wheel assembly shall be placed on the static balance with the bearing in place and the valve stem is up. It is recommended that a permanent scribe line be placed on the vertical balancer's scribe disc.

NOTES



DETAIL A

Figure 32-17. Landing Gear Handle

Landing Gear Extension and Retraction

Normal extension and retraction of the landing gear is accomplished by the airplane hydraulic system.

Normal retraction of the gears is accomplished when the gear control handle (Figure 32-17) is set to retract. This in turn energizes the retract solenoid on the hydraulic landing gear control valve. Hydraulic flow is routed to unlock the actuator internal downlock, while directing flow to the retract side of the gear actuator piston. The landing gears are held in the retracted up position by a mechanical uplock hook. The mechanical uplock hook is normally unlocked hydraulically.

During normal extension of the gears, the extend solenoid on the hydraulic landing gear control valve is energized when the gear down switches are actuated by the landing gear control handle. Hydraulic flow is routed to an uplock and sequence actuator for each gear. The actuators unlock the gear uplock hooks, directing hydraulic flow to the extend side of the gear actuator piston, extending the gear. The gear is held in the down-and-locked position by an internal downlock mechanism in the gear actuator. The internal downlock can only be unlocked with hydraulic pressure that is directed to the retract port.

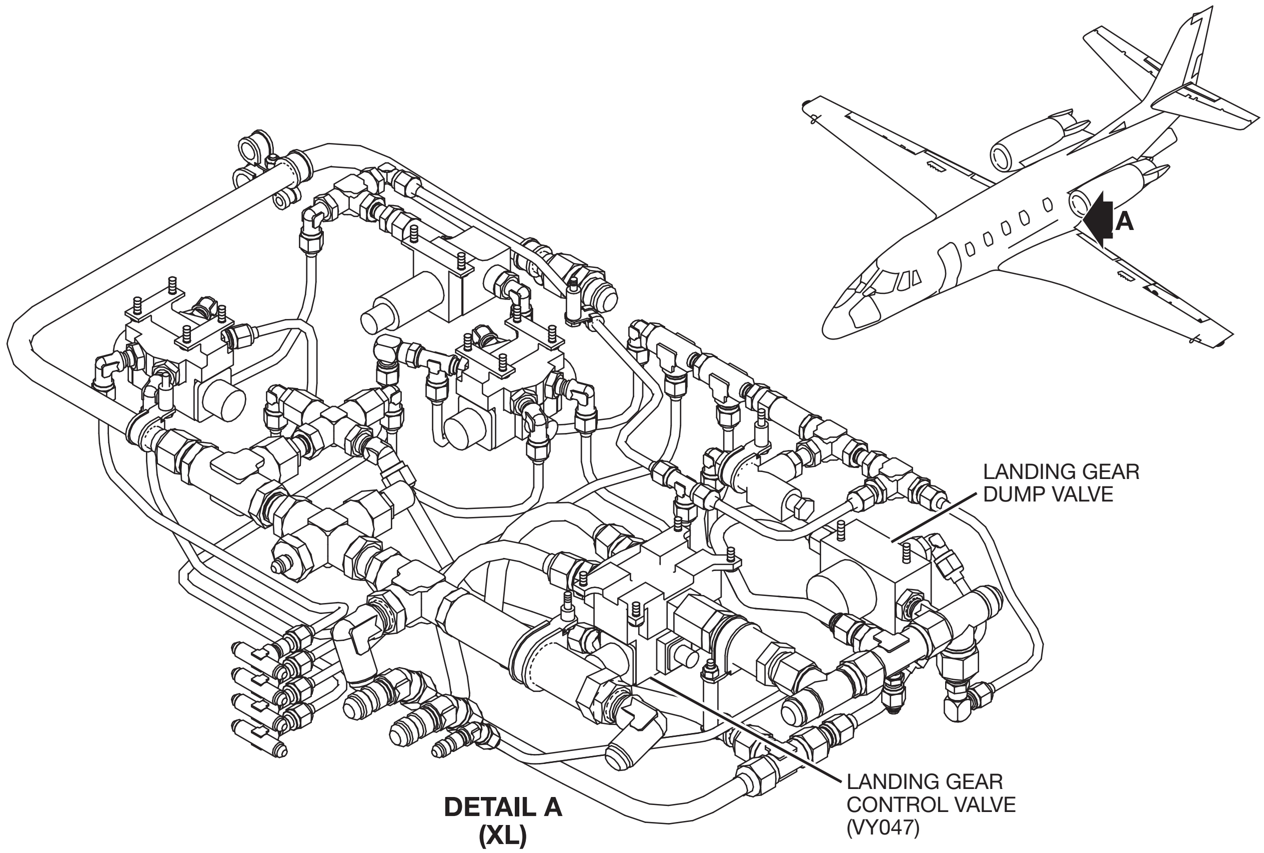
Landing Gear Control Unit

The landing gear normal retraction/extension system is electrically controlled by the action of the landing gear control handle. The landing gear control unit and gear handle assembly is in the bottom left corner of the center instrument panel (right installation is optional). The landing gear control unit/handle assembly includes:

- Retract and extend switches
- Warning lights
- Gear selector switch
- A control handle locking solenoid

The landing gear control handle locking solenoid prevents the handle from moving to a gear-up position while the airplane is on the ground.

NOTES



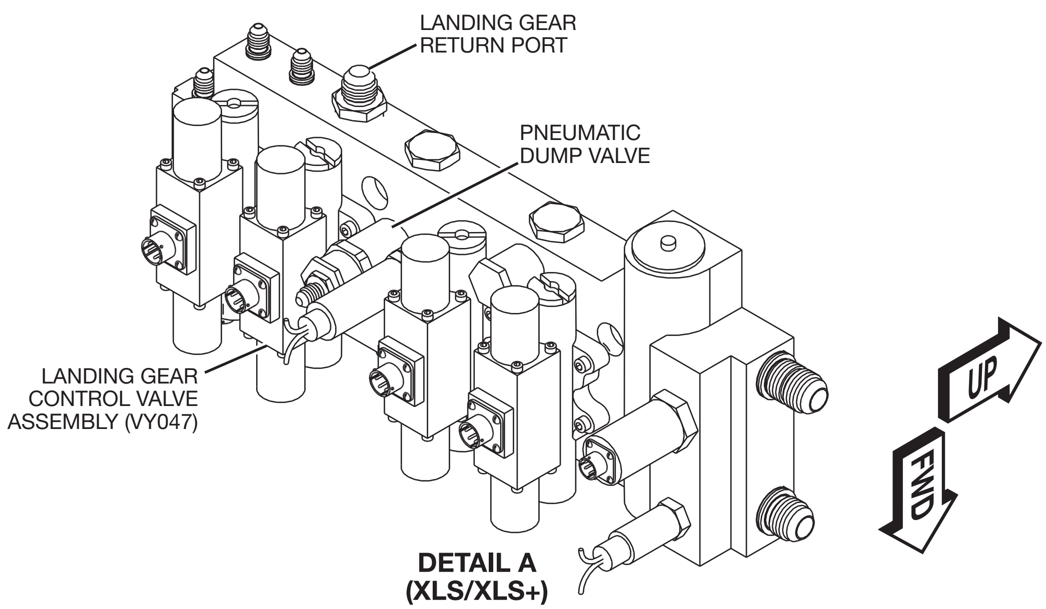


Figure 32-18. Gear Control Valves

Gear Control Handle

The landing gear control handle actuates electric switches. The switches open and close electrical circuits to the gear logic PC board that controls the retract/extend solenoid on the landing gear control valve. The control handle is spring-loaded in either the retract or extend position. To move the lever from one position to another, it must be pulled out so a pin on the handle can move over a cam. The pin actuates either the retract or extend switch, depending on which detent the control handle is in.

A gear selector solenoid at the landing gear control handle assembly has a spring-loaded plunger that prevents movement of the control handle while the airplane is on the ground. The solenoid is activated or deactivated by a landing gear squat switch. It receives electrical power from the gear position and warning electrical system. A gear-selector switch is also on the landing gear control handle assembly. The gear-selector switch is used to change the landing gear up or down warning lights, as selected by the gear control handle. The gear-selector switch is a primary part of the gear position and warning electrical system.

Landing Gear Control Valve

The landing gear control valve is attached to the hydraulic control sub panel (XL) or hydraulic manifold assembly (XLS/XLS+) below the tail section under the fairings (Figure 32-18).

The landing gear control valve is connected in the system by elbows, tee, and reducer fittings. A quick disconnect connector connects electric power to the solenoid windings.

The landing gear control valve is a four-way valve, operated by a three-position spool valve. The three-position spool valve has two independent windings. While deenergized, the solenoid maintains a neutral position that blocks off the hydraulic inlet port, and connects both the retract and extend outlet ports to the return port. This prevents trapping fluid under pressure in the lines. The landing gear control valve functions with a power source of 18 to 30 VDC. The spool valve maintains a neutral position until electrical power is supplied to either solenoid and hydraulic inlet pressure repositions the spool. The position of the spool depends on which solenoid is energized. With the extend solenoid winding energized, the hydraulic inlet port is connected to the extend outlet port; and the retract outlet port is connected to the return port. With the retract solenoid winding energized, the hydraulic inlet port is connected to the retract outlet port; and the extend outlet port is connected to the return port.

NOTES

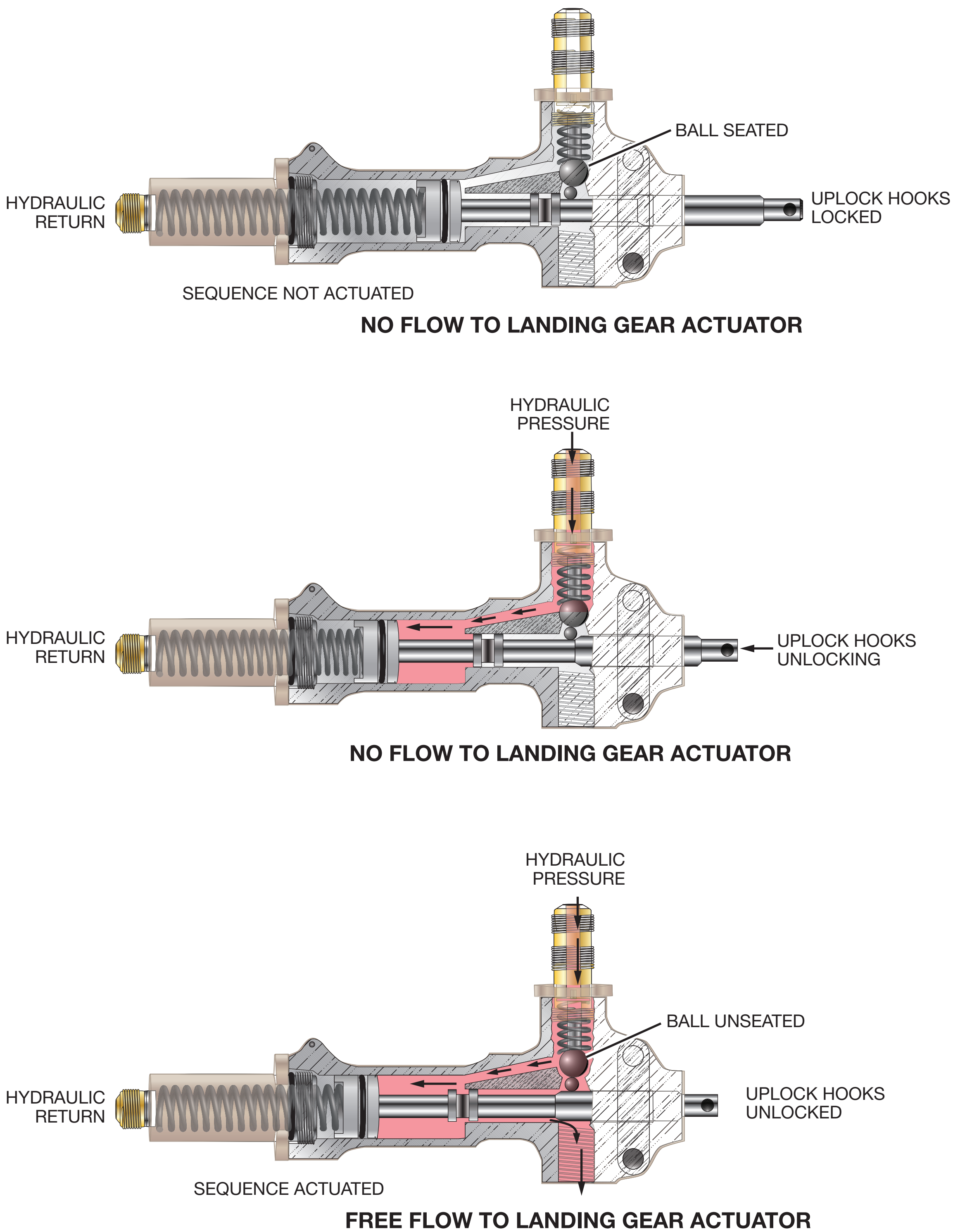


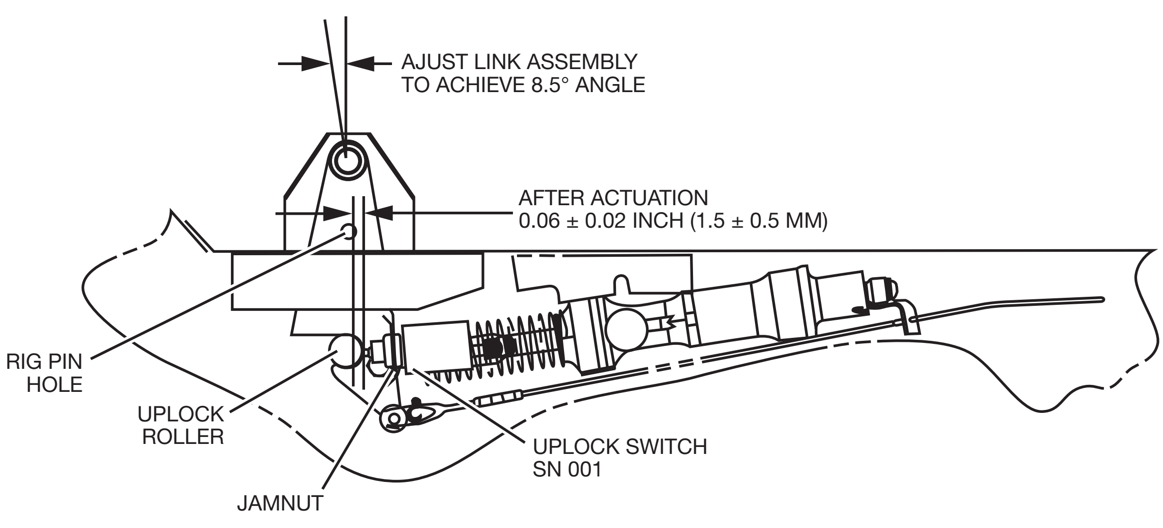
Figure 32-19. Uplock Sequence Actuator

Uplock and Sequence Actuator

An uplock and sequence actuator is at each landing gear uplock hook to fully release each uplock hook, before allowing hydraulic pressure to the respective gear actuator (Figure 32-19). A check valve in the uplock and sequence actuator prevents the passage of hydraulic fluid until the uplock hook is fully released and the sequence actuator rod has retracted far enough to unseat the check valve. Reverse hydraulic flow during gear retraction unseats the check valve, permitting passage of returning fluid.

NOTES

VIEW LOOKING FORWARD
MAIN GEAR UP LOCK



VIEW LOOKING INBOARD (LEFT SIDE)
NOSE LANDING GEAR UP LOCK

Figure 32-20. Uplock Hooks and Switches

Uplock Hook Assemblies

The main and nose landing gear uplock assemblies are mechanically locked and hydraulically powered to the unlocked position. The uplock hooks automatically lock the gear mechanically when the gear uplock rollers contact the uplock hooks (Figure 32-20). Releasing the gear from the uplock hooks is normally accomplished through an uplock and sequence hydraulic actuator (attached to the uplock mechanism). Manual uplock hook release is accomplished by a cable attached to the uplock mechanism.

Up and Locked Switches

There are three up-and-locked switches on the uplock hooks: right, left, and nose.

The uplock switches are used in the gear control system to monitor when the gear is up and locked. A signal is sent to the gear control PC board which turns off hydraulic pressure to the gear in the up position. The switches also make inputs to the gear monitoring system to extinguish the gear UNLOCK annunciator when the gear is up-and-locked.

Adjust the switches by positioning the switch for an open circuit in the up-and-locked positions with a 0.04 to 0.08-inch overtravel.

Circuits are closed in all other positions.

CAUTION

Do not overtorque nut on aluminum threads of the uplock switch. After the switch is properly mounted, the switch case must not be rotated with excessive force. Excessive force causes the tab washer to cut into and destroy the aluminum threads on the switch.

NOTES

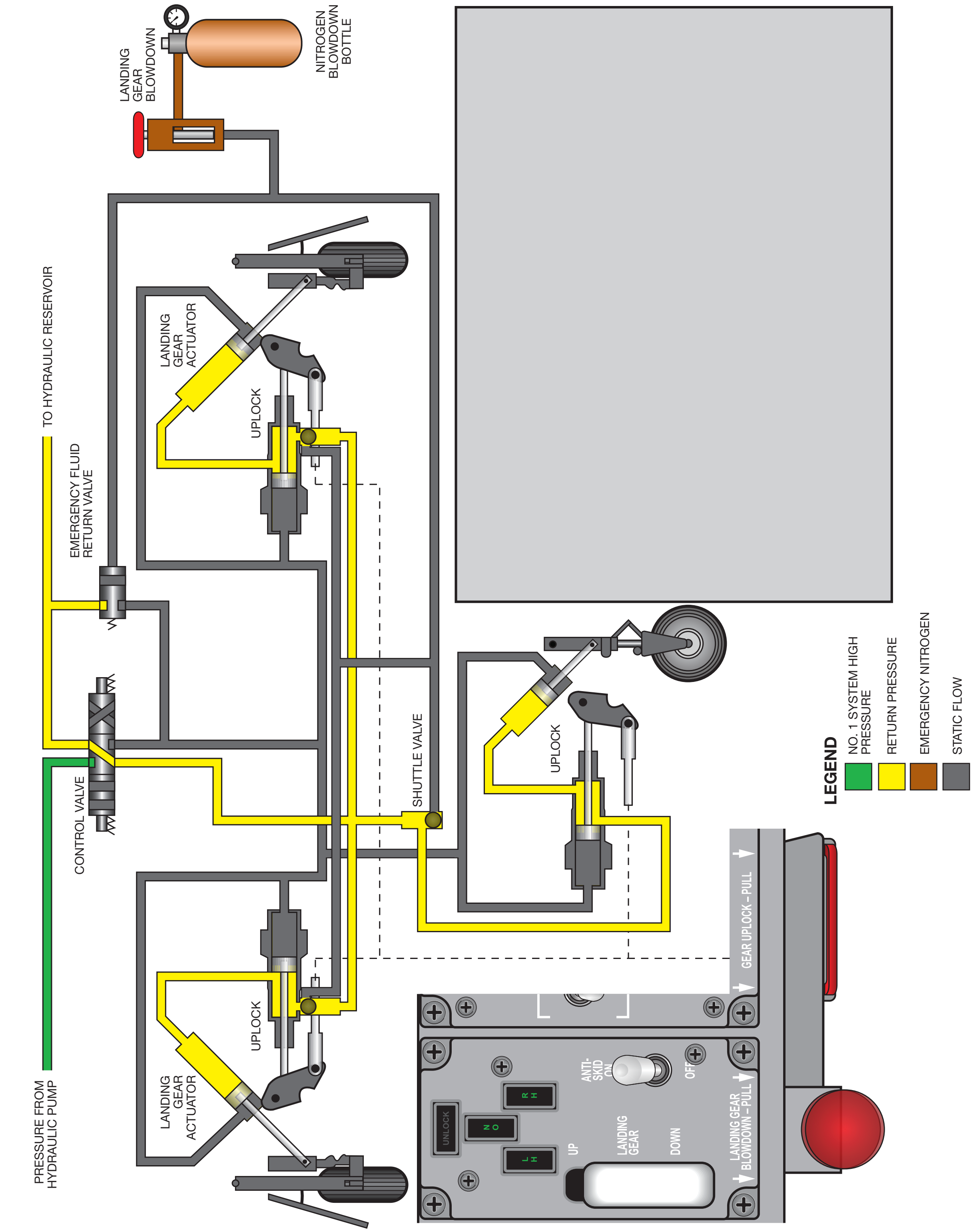


Figure 32-21. Hydraulic Diagram

OPERATIONS

Hydraulic Operations

When the landing gear control handle is placed down to extend the landing gear, an electrical circuit is completed through each of the following:

- The gear downlock switch
- The extend solenoid of the gear control valve
- The hydraulic system bypass valve (Figure 32-21)

When energized, the bypass valve solenoid closes the valve and routes the full flow of hydraulic fluid to the landing gear control valve. When it is energized, the extend solenoid landing gear control valve positions the flow ports in the control valve, directing the hydraulic pressure to the individual uplock and sequence actuators. The uplock and sequence actuators unlock the uplock hook and direct the hydraulic pressure to the extend side of the landing gear actuator pistons until each gear is fully extended and the gear downlock switches are actuated.

When all downlock switches are actuated, the electrical circuit is opened, which removes power from the extend solenoid of the landing gear control valve and the hydraulic system bypass valve solenoid. The hydraulic system bypass valve opens. While the landing gears are down-and-locked, only the internal locks in the gear actuator hold the gear extended.

When the landing gear handle is placed up to retract the gear, an electrical circuit is completed through each of the following:

- Gear uplock switch (right, left, and nose)
- The retract solenoid of the gear control valve
- The hydraulic system bypass valve

The bypass valve solenoid, when energized, closes the valve and routes the full flow of hydraulic fluid to the landing gear control valve. The retract solenoid winding of the landing gear control valve, when energized, positions the flow ports in the control valve and directs hydraulic pressure to the retract side of the gear actuator piston. This also releases the actuator internal downlock. Hydraulic pressure continues until each gear is retracted and gear uplock switches are actuated. When all gear uplock switches are actuated, the circuit is opened, which removes power from the retract solenoid winding of the landing gear control valve and to the hydraulic system bypass valve solenoid. The hydraulic system bypass valve opens. While the landing gears are up-and-locked, only the uplock hooks hold the gear retracted.

Summing Device Assembly

The summing device is under the baggage compartment floor near the forward side. It is used to increase the mechanical advantage of the uplock cable release system.

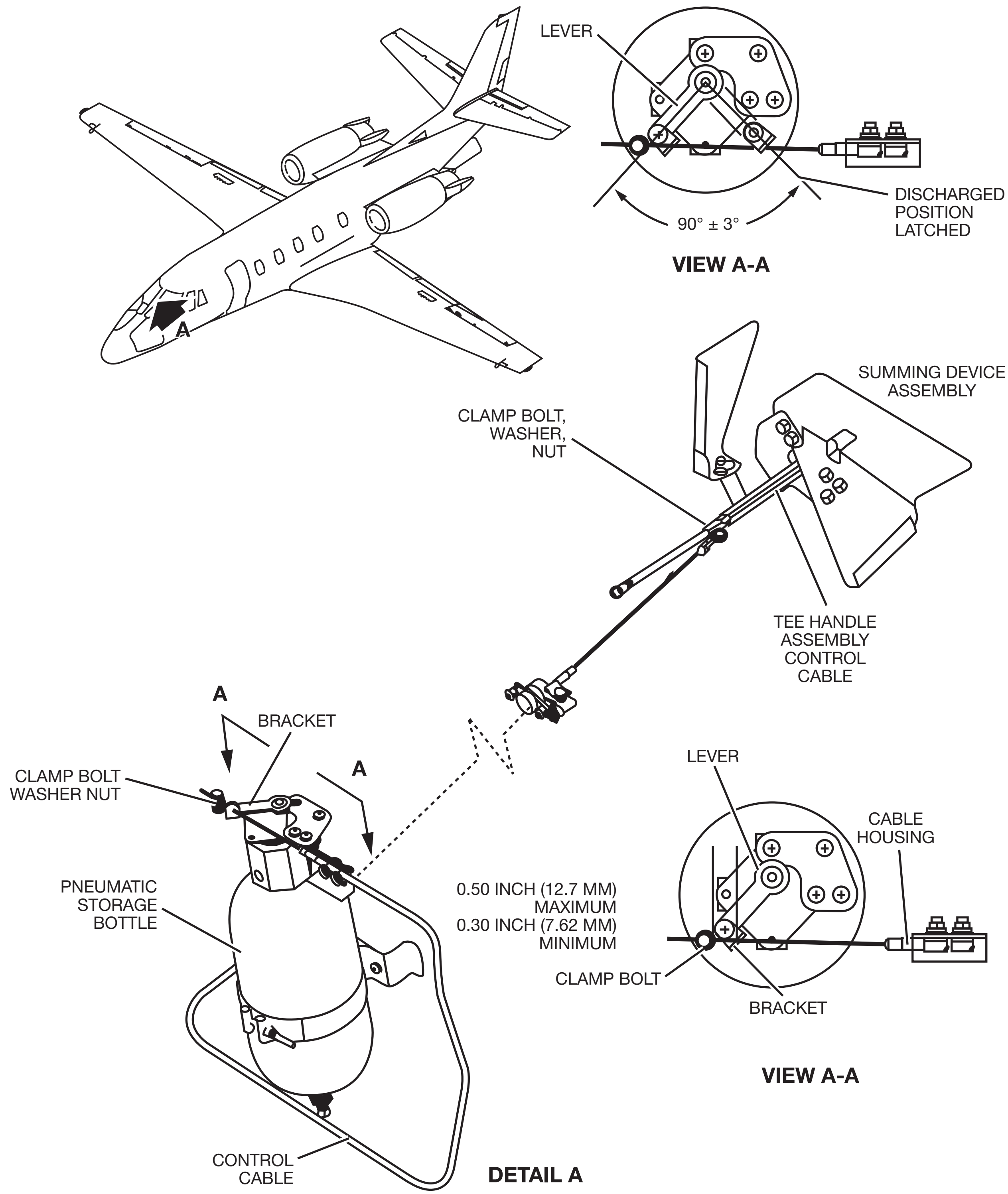


Figure 32-22. Auxiliary Gear Blow Down Bottle

Gear and Brake Emergency Air Storage Blow Down Bottle

There is an emergency air storage blow down bottle on the left forward side of the forward pressure bulkhead (Figure 32-22). The storage bottle holds 75 to 100 cubic inches of nitrogen at 2050 psi. A relief valve is provided to rupture and release excessive pressure at 4000 psi. The air storage bottle stores air pressure used to operate the gear emergency extension system as well as the emergency brake system. The gear extension control lever on the air bottle is operated by the round knob in the cockpit through a pull cable assembly. The control lever on the bottle latches in the release position. Therefore, the control lever must be reset by ground maintenance in order to return the handle to the normal position.

Emergency gear extension lines are connected to a vent line while the air storage bottle control lever is in the normal position. The vent line provides a route for venting the air extension chambers of the main gear actuators during hydraulic operation of the gear. The thermal relief valve releases at 4000 ± 250 psi. The relief valve is not reusable after rupture and either the relief valve and/or the complete bottle must be replaced.

A moisture bleed valve is rotated counter-clockwise to remove any moisture from the air blow down storage bottle.

NOTES

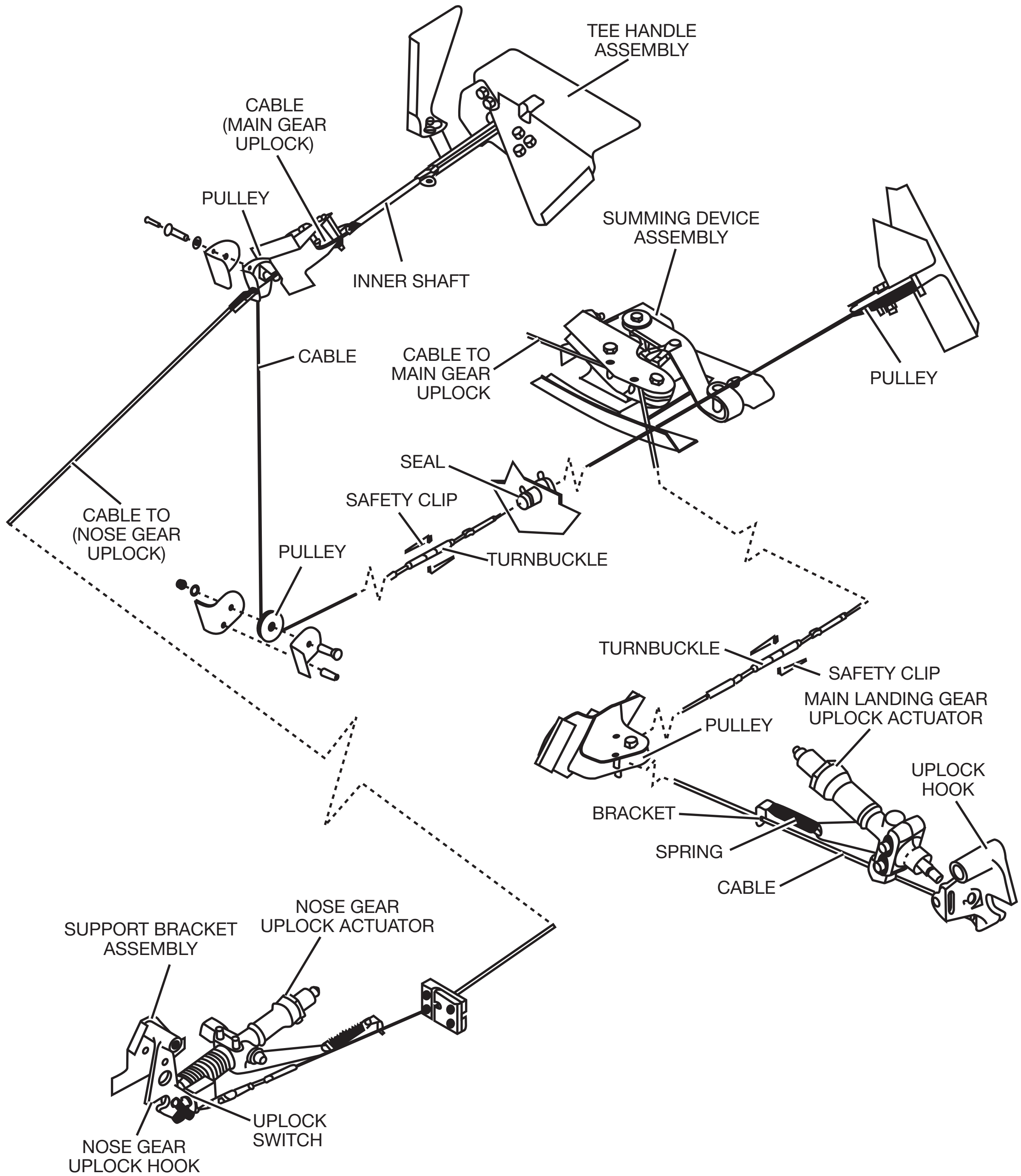


Figure 32-23. Auxiliary Gear Control

Auxiliary Emergency Gear Control

The auxiliary emergency gear control consists of a pull handle to unlock the gear uplock hooks that allows the gears to free fall from the gear wheel wells to the down-and locked-position, and a round knob to release high pressure air from an air storage bottle (see Figure 32-23). The round release knob cannot be pulled until the tee handle has been pulled and turned clockwise to the locked position on the XL/XLS. The full handle is connected to the uplock hook by cables. The round knob is connected to the valve control on the emergency air blow down storage bottle (Figure 32-24).

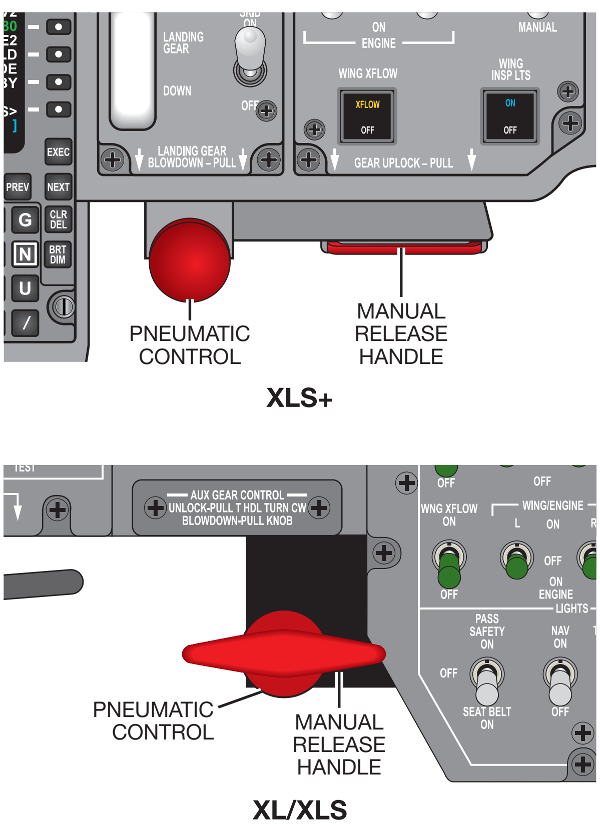


Figure 32-24. Control Handles

NOTES

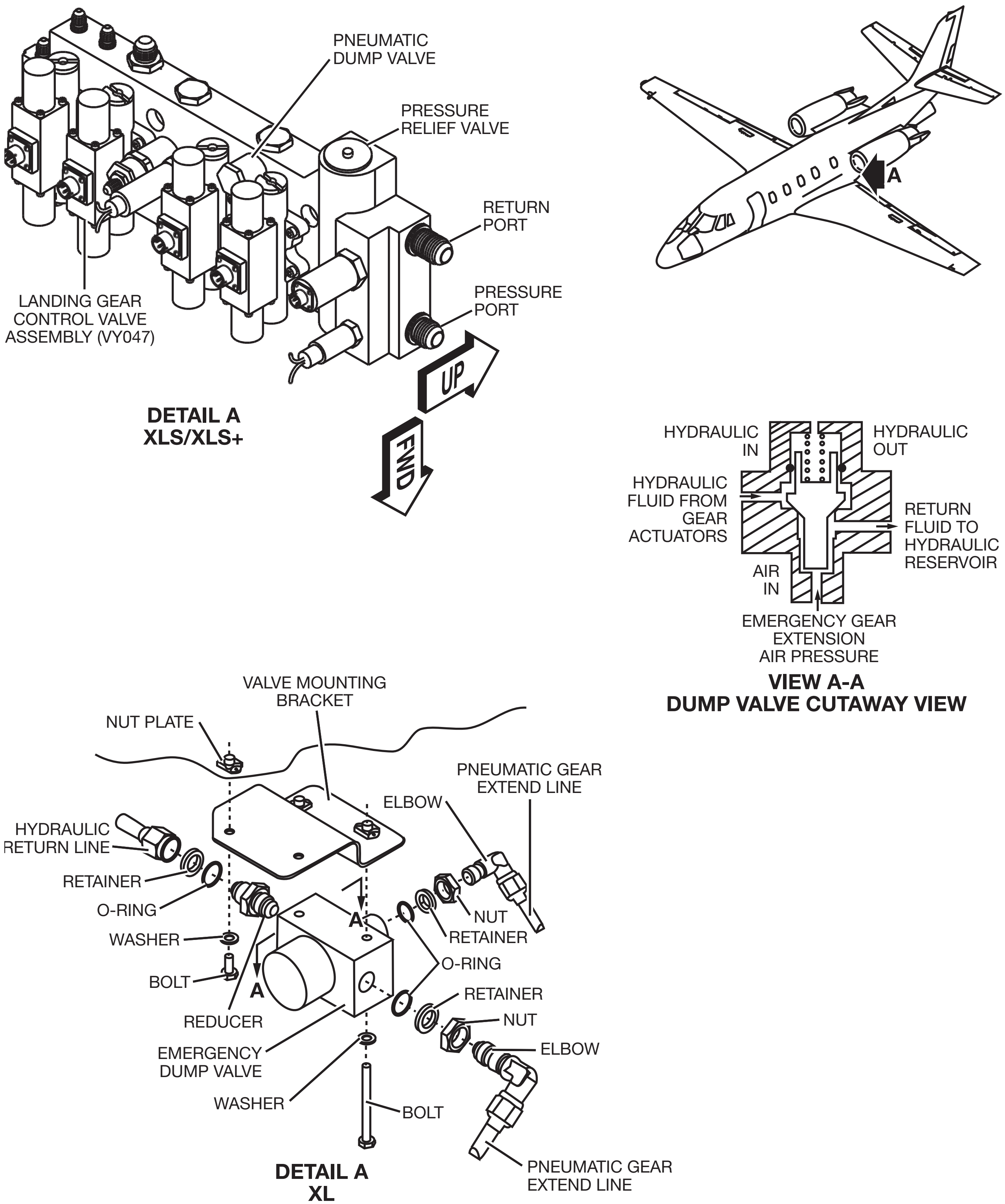


Figure 32-25. Gear Dump Valve

Emergency Dump Valve

The purpose of the emergency dump valve is to release trapped hydraulic fluid from the retract side of the landing gear actuator during emergency (pneumatic) extension of the landing gears only (Figure 32-25).

The dump valve is connected to the hydraulic gear retract line, the hydraulic system return line, and the pneumatic gear extension line. Pneumatic gear extension pressure of 200 psi or more actuates (opens) the dump valve. At the same time, pneumatic pressure on the extend side of the landing gear actuators forces hydraulic fluid on the retract side of the gear actuators to flow through the open dump valve and return to the hydraulic system reservoir.

Shuttle Valves

Three shuttle valves are incorporated in the landing gear hydraulic system, one for each landing gear. They are in the wheel well of their respective gear. The gear actuators have only one common gear extension chamber. Use of the shuttle valve in the system admits pressure from either hydraulics or pneumatics.

NOTES

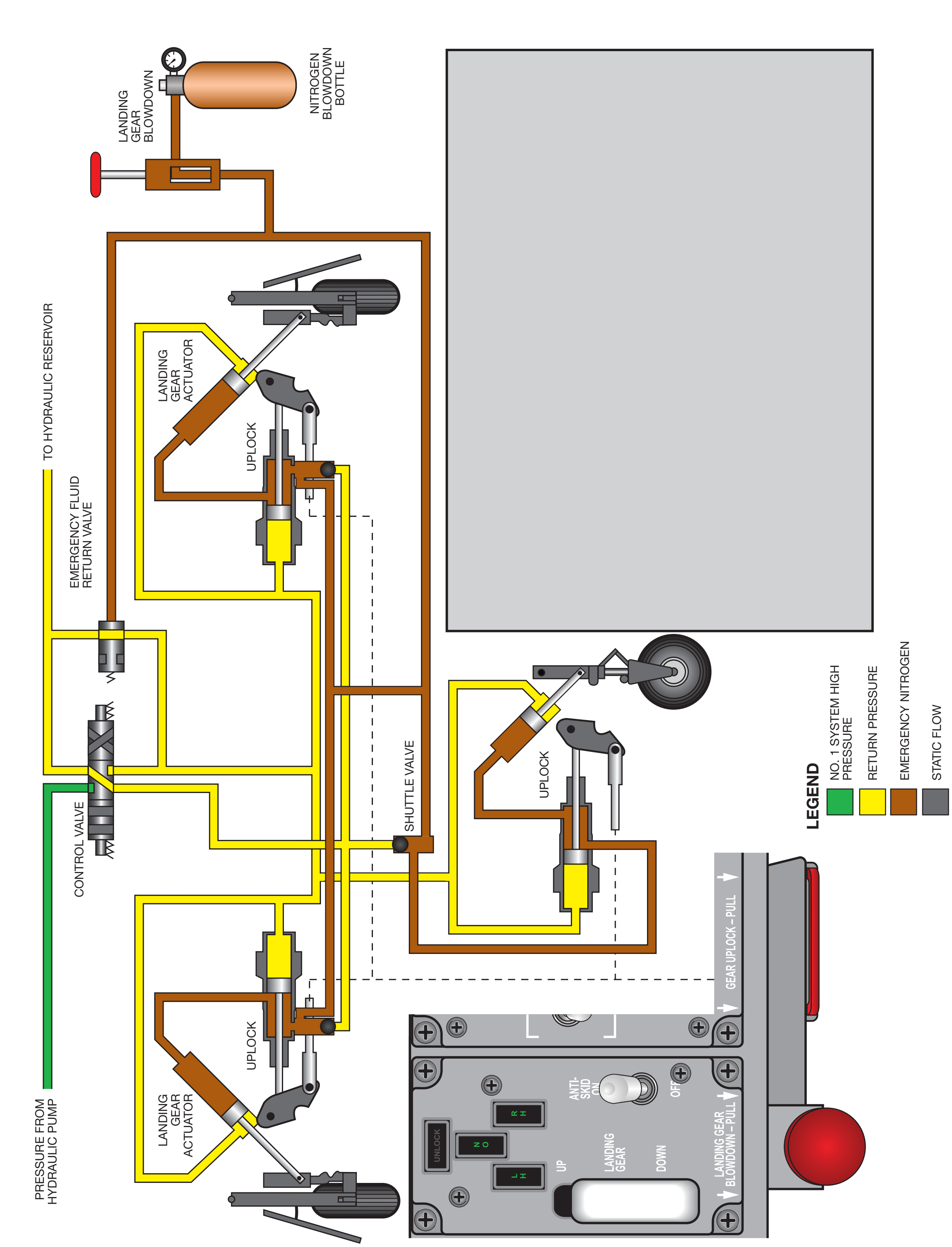


Figure 32-26. Gear Emergency Extension

Emergency Gear Extension Operation

Pneumatic landing gear operation is restricted to extending the gear and is identified “emergency gear extension” (Figure 32-26). Operation of two manual controls is required. Pull the emergency gear release pull tee handle and rotate it in order to lock the pulled out handle. Pulling the handle overrides (compresses) the springs on the gear uplock hooks, releasing the hooks and allowing the landing gears to free fall from the wheel wellss. When rotating the tee handle (XL/XLS) to the lock position, it aligns a groove in the tee handle control assembly mounting shaft, which permits pulling the round pneumatic control knob. Pulling the round knob actuates the emergency air blow down storage bottle control lever; and a pin latches it open. High pressure air is released through an orifice check valve to the shuttle valves which direct the pressure to the extend side of the landing gear actuating pistons. This drives the landing gear to the down-and-locked position. At the same time, air pressure positions the dump valve to permit hydraulic fluid from the actuators to return directly to the hydraulic reservoir. When the control lever closed, residual pneumatic pressure is vented overboard.

The emergency control cable rotates the control valve lever. At full open the valve lever latches. It must be manually released by pushing a pin on the control valve arm, before the lever can be repositioned back to the closed (normal) position.

The emergency brake system utilizes the same pneumatic storage bottle as the auxiliary extension system. Operating an emergency handle in the cockpit activates the emergency brakes.

NOTES

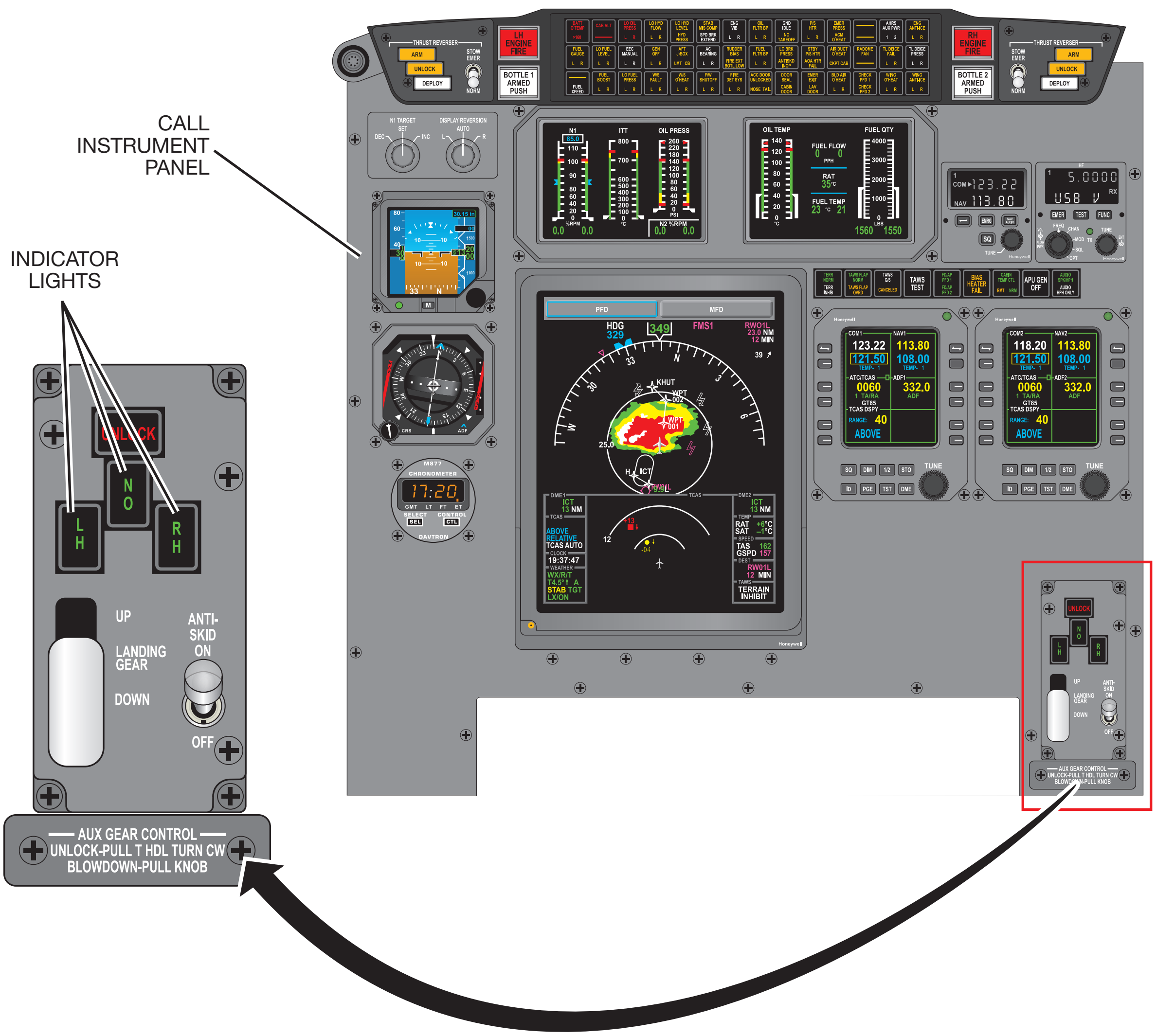


Figure 32-27. Gear Control Handle

POSITION AND WARNING SYSTEM

DESCRIPTION

The landing gear position and warning system provides a visual and audible indication of the locked or unlocked positions of the landing gear (Figure 32-27). Three green safe lights and a red UNLOCK light are in a group adjacent to the gear control handle at the bottom left of the center instrument panel. Each green light corresponds to one gear and indicates gear safe down-and-locked position. The red light indicates an unlocked gear position. The unlock light has two bulbs in parallel for continued operation if either burns out.

The landing gear warning system sounds an audible warning if:

- The gear is not down-and-locked, the flaps are extended beyond the approach (15°) position, and both throttles are retarded below 45° TLA.
- The gear is not down-and-locked, the radar altimeter senses that the aircraft is less than 500 ft AGL, and both throttles are retarded below 45° TLA.

If the altitude sensing function is lost, the system uses airspeed below 150 knots in place of the altitude.

OPERATION

Electrical power is present at each landing gear indicator light when the landing gear circuit breaker is closed. An individual ground circuit causes the light to come on when the landing gear actuator locks in the down position. The test switch and two landing gear test relays provide a separate ground for each light in order to check lamps for proper operation.

The unlock light is illuminated when any of the following conditions exist:

- With the gear selector switch in the down position and one or more gear are not in the down-and-locked position.
- With the gear selector in the up position, and one or more not up-and-locked while one or more gears are not down-and-locked. At least one gear has unlocked from the down position, but has not moved to the up-and-locked position.
- When the aircraft is on the ground with the gear down-and-locked and the gear control handle is in the up position.
- With the test switch in the gear lights test position. The test switch provides a separate power source to the unlock light and a gear test relay provides a ground regardless of the position of other switches in the system.

NOTES

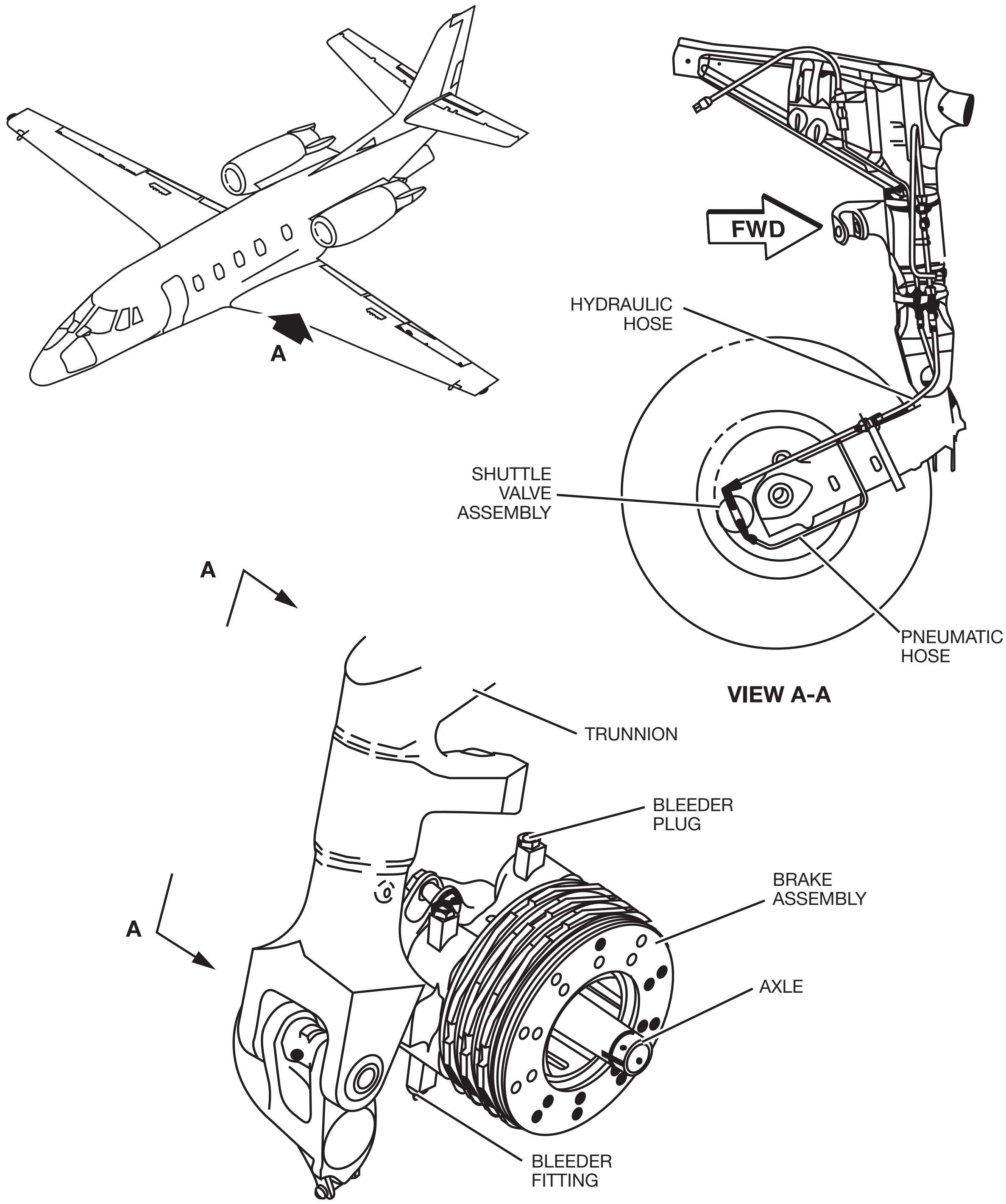


Figure 32-28. Brake Assemblies

WHEELS AND BRAKES

DESCRIPTION

The main landing gear wheels are equipped with disc-type brakes (Figure 32-28). The brakes are hydraulically actuated by a power brake valve, controlled from a master cylinder that is connected to each rudder pedal. The master cylinders, at the pilot and copilot positions, are connected in series to permit either pilot or copilot control of the brakes.

A parking brake valve is incorporated in the brake system which, when manually operated, prevents the return of hydraulic fluid pressure after the brakes are applied.

An emergency (pneumatic) braking system is provided for use when hydraulic braking fails. Auxiliary braking is controlled with a hand-operated valve that directs equal nitrogen pressure to each brake during emergency braking conditions. High-pressure nitrogen is supplied from the emergency gear and brake pneumatic storage bottle.

The main gear brake antiskid control is used to prevent wheel skidding on wet, dry or icy runways after a minimum wheel spin is attained.

The system consists of:

- Electrical control box
- Power brake valve
- Antiskid servo valve
- Motor/pump assembly
- Hydraulic accumulator
- Pressure switches
- Brake reservoir
- Mode switch
- Circuit breakers
- Indicator lights
- Self-test system

NOTES

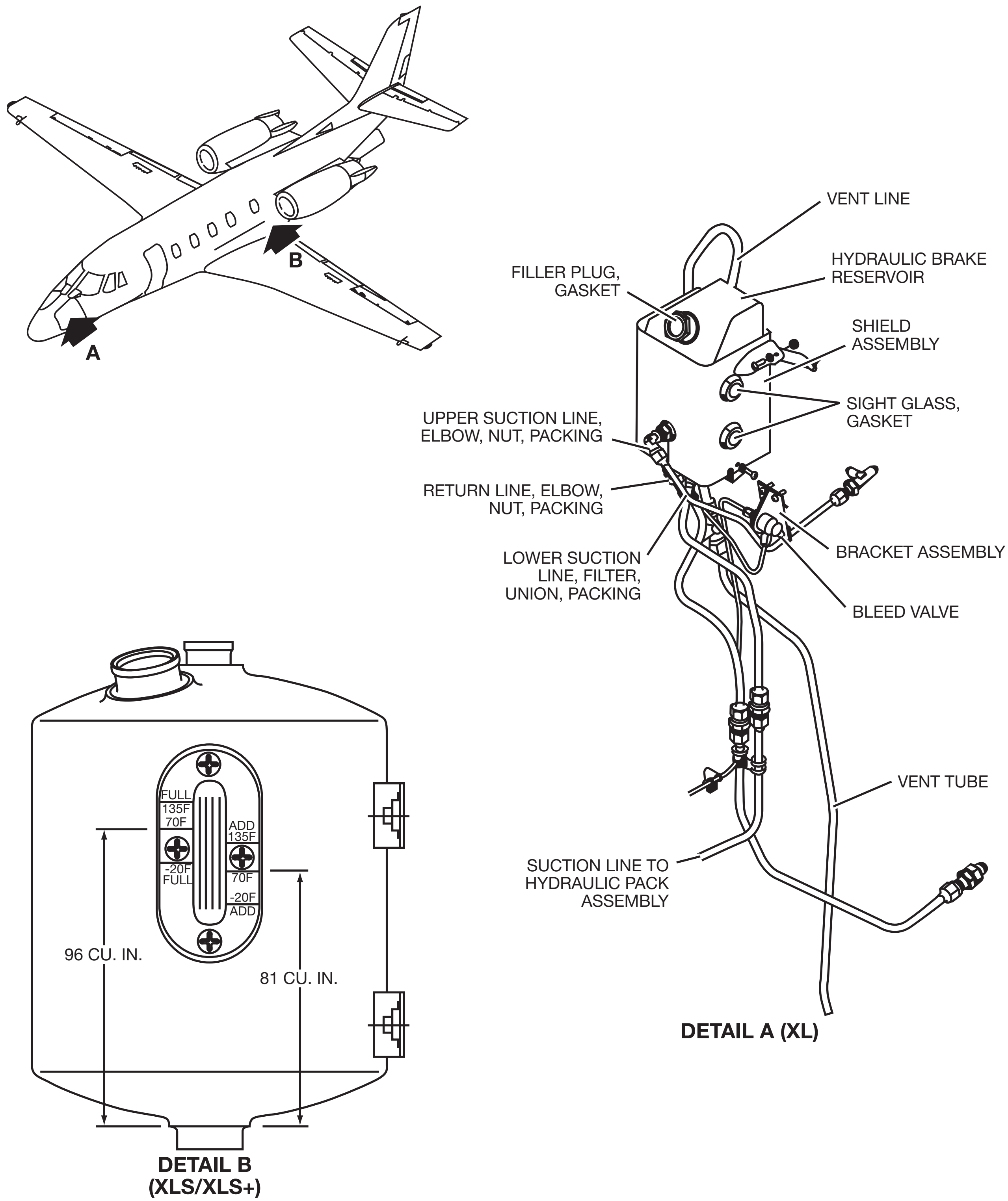


Figure 32-29. Brake Reservoir

COMPONENTS

Brake Fluid Reservoir

The brake system hydraulic fluid reservoir is on the forward side of the forward pressure bulkhead on the left side of the nose compartment (XL) or just forward of the battery compartment (XLS/XLS+) (Figure 32-29).

The reservoir consists of:

- Reservoir tank
- Sight glass
- Filler plug
- Connections for brake master cylinders supply line
- Vent line
- Bleed return line
- Supply line for the high pressure side
- Bracket for mounting the hydraulic bleed valve

Caution shall be exercised during maintenance practices as the brake system utilizes phosphate-ester based fluid.

The reservoir is pressurized by cabin pressure through a filter assembly and a check valve, teed into the reservoir overboard vent line. The filter prevents contamination of the brake fluid from cabin air (tobacco tars, etc.) The check valve prevents hydraulic fumes or fluid from entering the cabin.

A relief valve in the overboard vent line restricts cabin pressure from going overboard, permitting pressurization of the reservoir. The relief valve depends on differential pressure for the amount of flow through the valve. However, a small orifice incorporated in the relief valve permits a continuous bleed overboard.

The reservoir is pressurized to prevent cabin pressurization air from entering the low-pressure brake lines as they route through the cabin. As cabin pressure increases, the pressure on the brake low-pressure fluid also increases proportionally, preventing a pressure differential, which could allow air to enter the low-pressure fluid and cause spongy or flat brake pedals.

NOTES

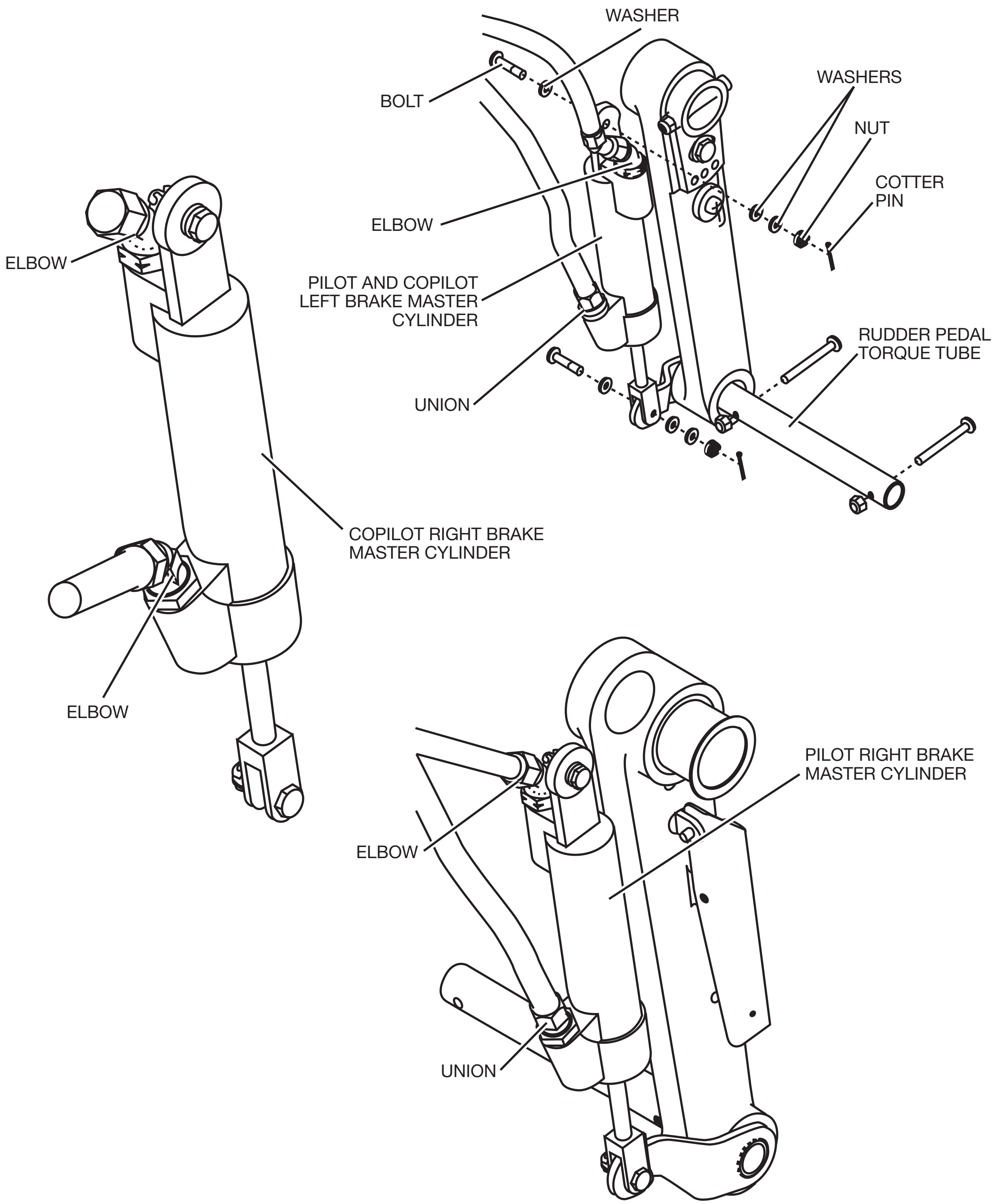


Figure 32-30. Brake Master Cylinders (XL)

Brake Master Cylinders

The pilot and copilot master cylinders are identical. However, the inboard and outboard master cylinders are not interchangeable (Figure 32-30). Maintenance practices for the master cylinders are typical.

The master cylinders connect to the power brake valve under the left forward belly fairing. The input pressures from the master cylinder control the output pressure from the power brake valve. The XLS/XLS+ does not use master cylinders, but incorporates pushrods and cables to the power brake valve.

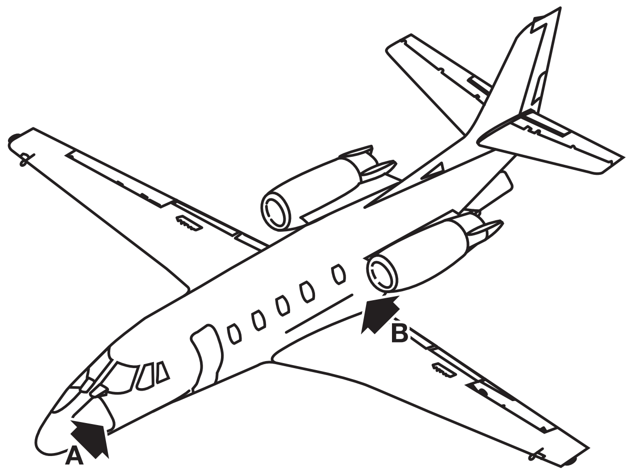
NOTE

The hoses must be connected for the fluid to flow from the reservoir to the top of the pilot master cylinder, out the bottom of the pilot master cylinder, to the top of the copilot master cylinder, and out the bottom of the copilot master cylinder to the brakes.

NOTE

When performing maintenance or conducting inspections near the master brake cylinders, particular attention must be given to maintaining proper clearances between the hydraulic lines, flex hoses, and the rudder cables. After installing a new flex line or tightening a fitting, the rudder pedals must be cycled through their full travel to ensure that there is no physical contact between the two systems.

NOTES



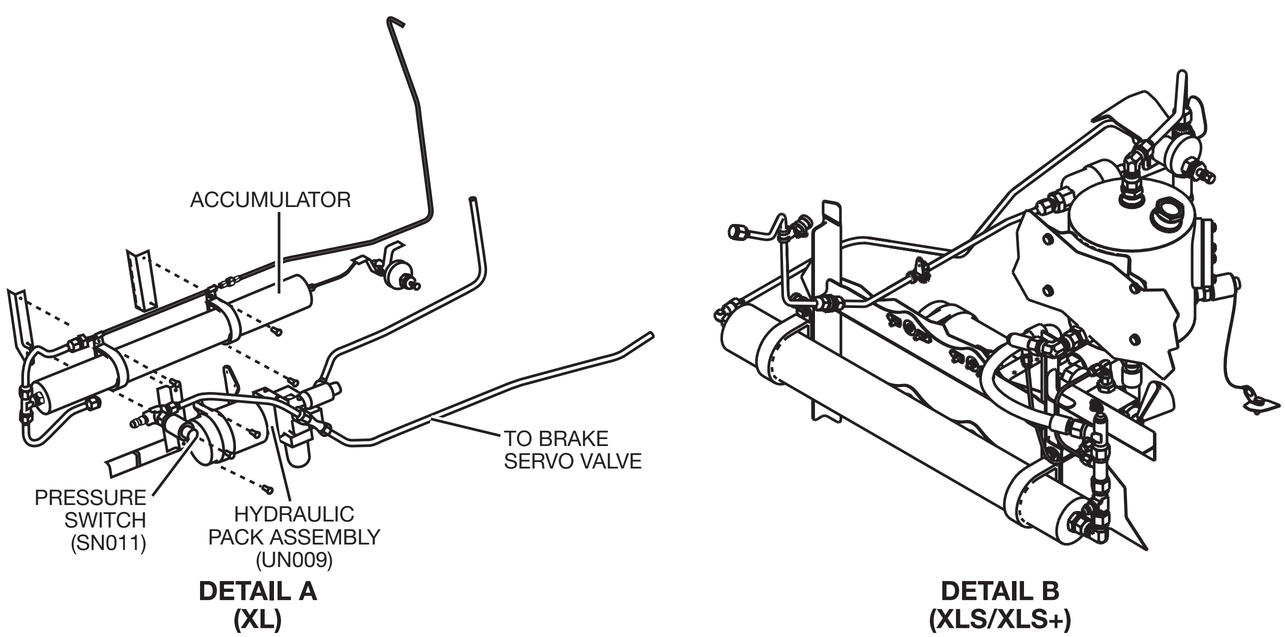


Figure 32-31. Brake Power Packs

Motor/Pump Assembly

A DC electric motor drives a hydraulic pump to provide high pressure for the brake system (Figure 32-31). System pressure is normally maintained between approximately 1,230 and 1,500 psi with a pressure switch in the brake system. The motor/pump assembly includes:

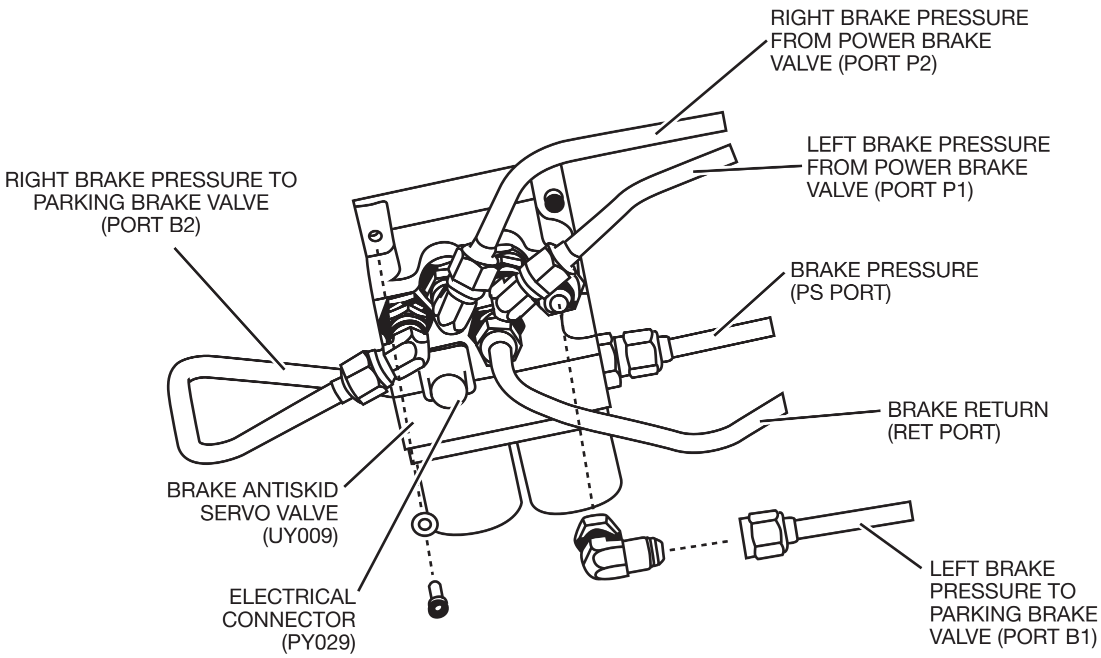
- A relief valve to prevent overpressurization
- A filter assembly to prevent contaminating brake hydraulic fluid in case of a component failure
- A pressure switch used to indicate a low brake pressure condition
- A check valve to prevent reverse flow of the hydraulic fluid

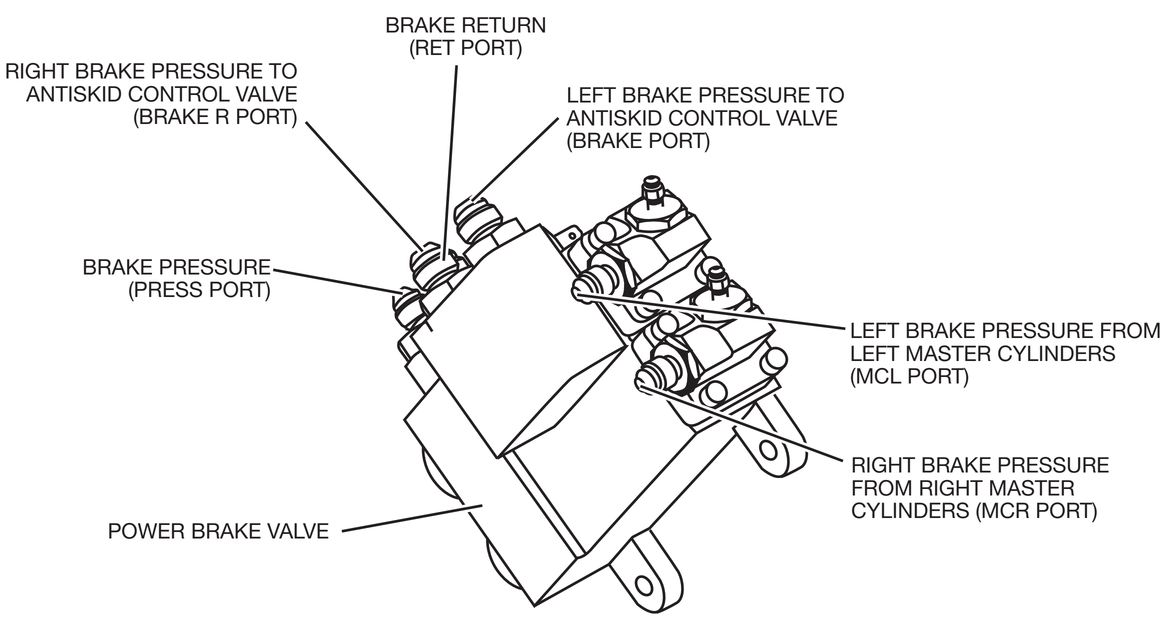
The motor is controlled by brake control motor relay in the left nose avionics compartment. The brake control motor relay closes when the landing gear control unit is selected to the gear extend position and a pressure switch in the nose avionics compartment is closed. The pressure switch opens, deactivating the motor at 1500 ± 50 psi on an increasing pressure and closes on a decreasing pressure of 1230 psi, +100 or –0 psi. The landing gear extend switch disables the control relay, preventing the motor from operating unless the landing gear control handle is in the extend position.

Accumulator

The accumulator provides a fluid reserve of 50 cubic inch capacity under pressure for the power brake function. An accumulator charging valve and pressure gauge are components of the accumulator system. The pressure gauge and charging valve are adjacent to the brake reservoir. The charging valve is the same type of valve used for servicing the air in the landing gear strut. The accumulator pressure gauge has numerical markings. There is a “temperature vs. pressure” placard for servicing on the left nose avionics bay door.

NOTES





(XL)

Figure 32-32. Antiskid Servo Valve and Brake Metering Valve

Power Brake Valve

The power brake valve is behind the belly fairings on the forward left side of the first bulkhead forward of the wing (Figures 32-32 and 32-33). The power brake valve regulates a maximum of 1000 psi, +50 or –20 psi to the brakes based upon pilot/copilot input to the left/right brake master cylinders.

Antiskid System

With the ANTI-SKID ON–OFF switch the operator has the option of power brakes with antiskid protection; or power brakes with no antiskid protection. In the ON position with aircraft speeds above 15 knots, the operator has power brakes with antiskid protection.

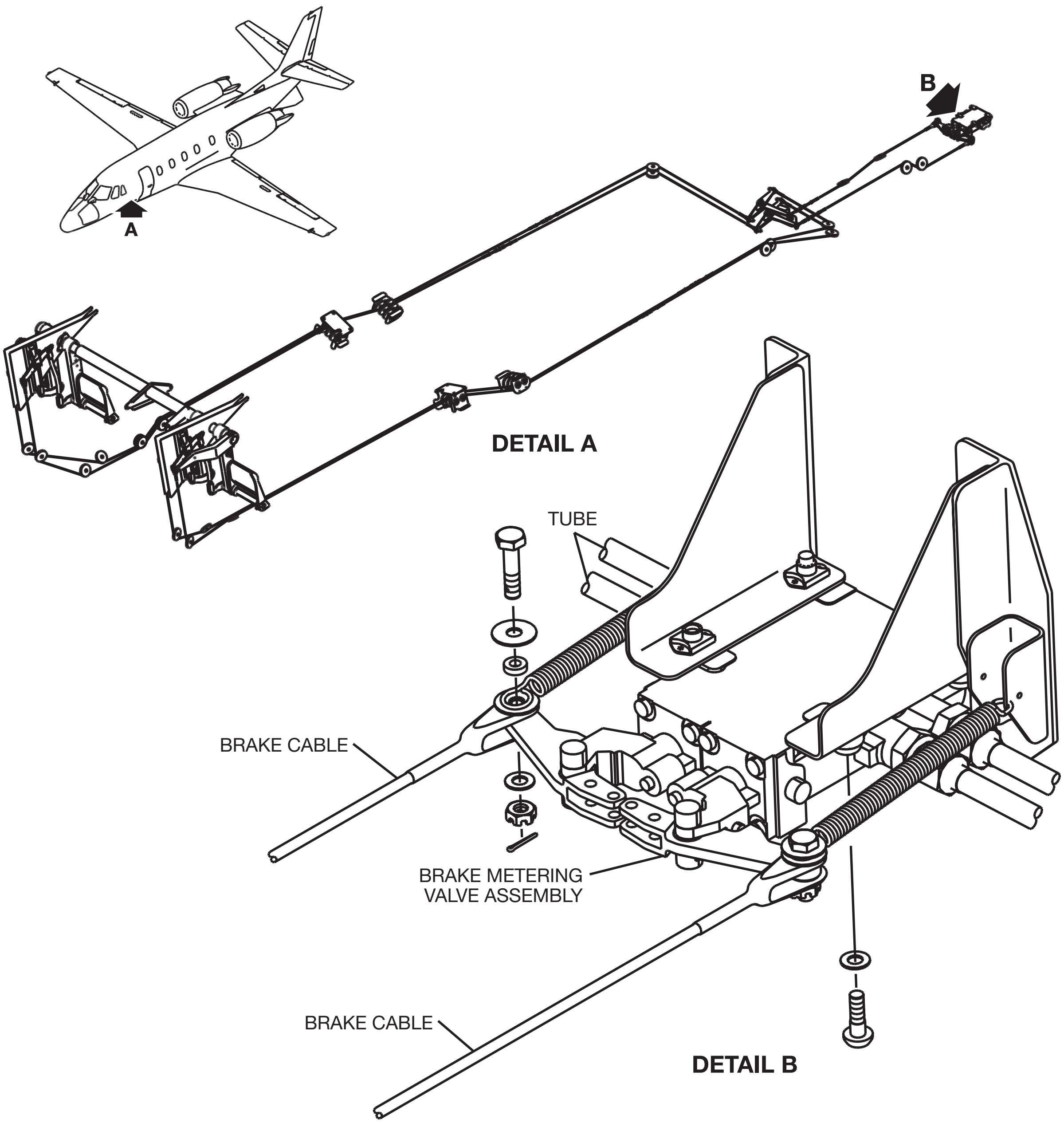


Figure 32-33. Power Brake Valve (XLS/XLS+)

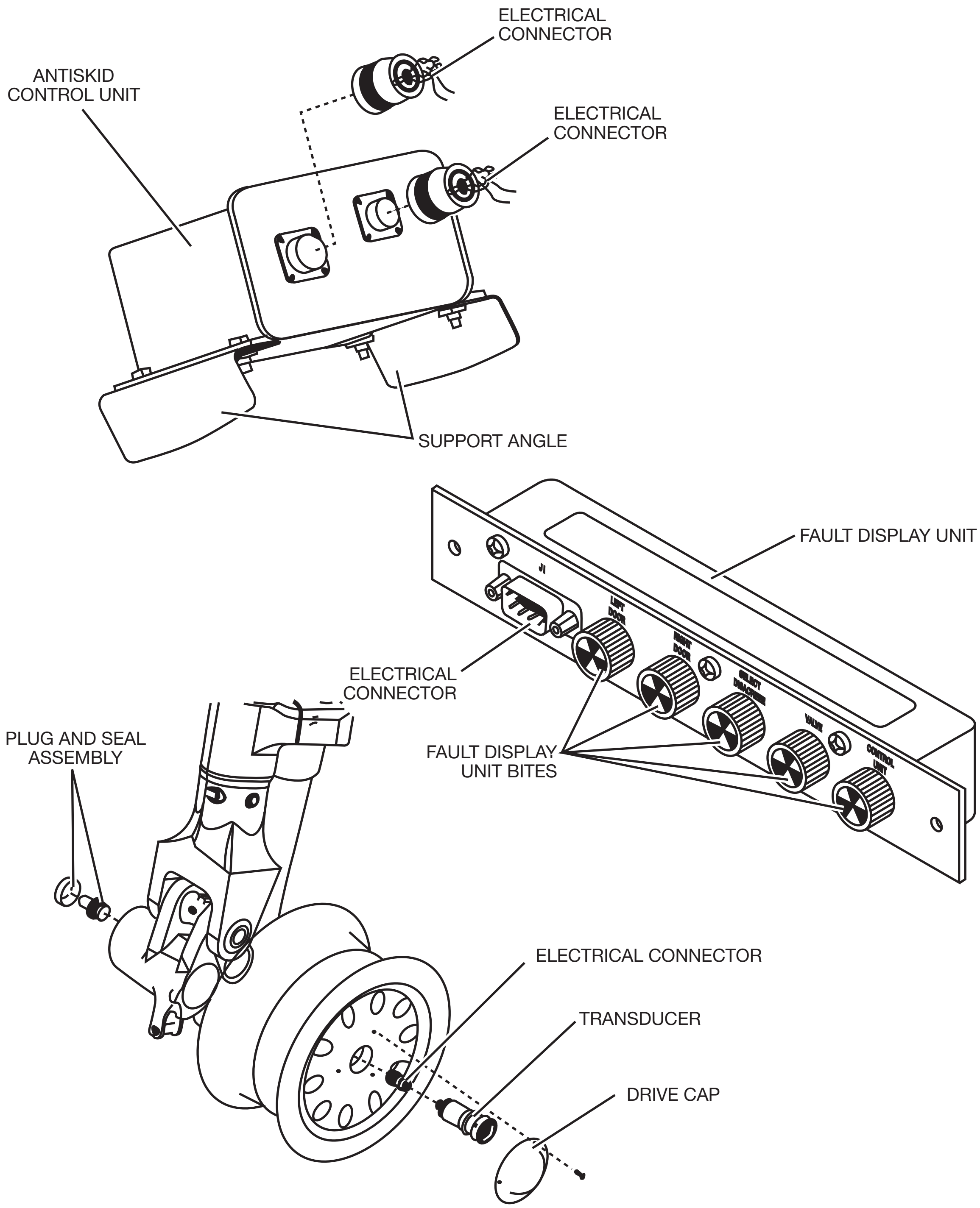


Figure 32-34. Antiskid System Components

The antiskid system is included as part of the power brake system to provide maximum braking on wet or icy runways. The system detects an incipient skid by measuring each landing wheels deceleration, then reduces the brake pressure (proportional to the deviation from the normal braking deceleration). This prevents the individual tire from continuing into a skid and modulates the brake pressure to achieve the most efficient braking.

The antiskid system consists of (Figure 32-34):

- Two wheel speed transducers
- A digital antiskid control unit
- Antiskid control servo valve
- ANTISKID ON/OFF switch
- ANTISKID INOP annunciator
- A fault display unit

There are wheel speed transducers in each main gear wheel axle. Each transducer generates an electrical signal by wheel rotation and sends the signal to the digital antiskid control unit. The digital control unit (which also receives signals from the left/right main gear squats) switches and the landing gear handle extends. This, in turn, retracts the control switches, providing signals to the antiskid control servo valve. The antiskid control servo valve reduces pressure to each brake assembly independently, as required, to prevent the tires from skidding.

The landing gear control handle extend/retract switches provide a ground circuit to the digital antiskid control unit when the landing gear control handle is positioned to the extend position.

The antiskid system provides two additional safety features:

- Touchdown protection prevents landing with hydraulic pressure applied to the brakes. The antiskid servo valve does not allow pressure to the brakes until both main gear are on the ground as sensed by the squat switches.
- Locked wheel crossover protection is provided between left and right wheels respectively. For speeds greater than 40 knots, locked-wheel protection provides a pressure-dump command to the slow wheel circuit when the velocity of the slow wheel is 50% or slower than that of the fast wheel. The pressure is completely dumped.

CAUTION

In the event an electrical failure should occur causing either total or partial brake system component failure, the brake system does not operate as a manual brake system even though the pedals in the cockpit are firm. The emergency brake system must be utilized.

Antiskid Wheel Transducer

The wheel speed transducers consist of:

- A rotor shaft
- Coil assembly
- Bobbin assembly
- Bearings contained in a housing assembly

A drive coupling is attached to the rotor shaft, which is driven directly by the drive assembly rotation.

A magnetic flux variation of 36 Hz per wheel revolution is produced and is supplied to the antiskid control unit as the necessary wheel speed data input.

CAUTION

Do not scratch the inner surface of the axle while removing sealant and expansion plug.

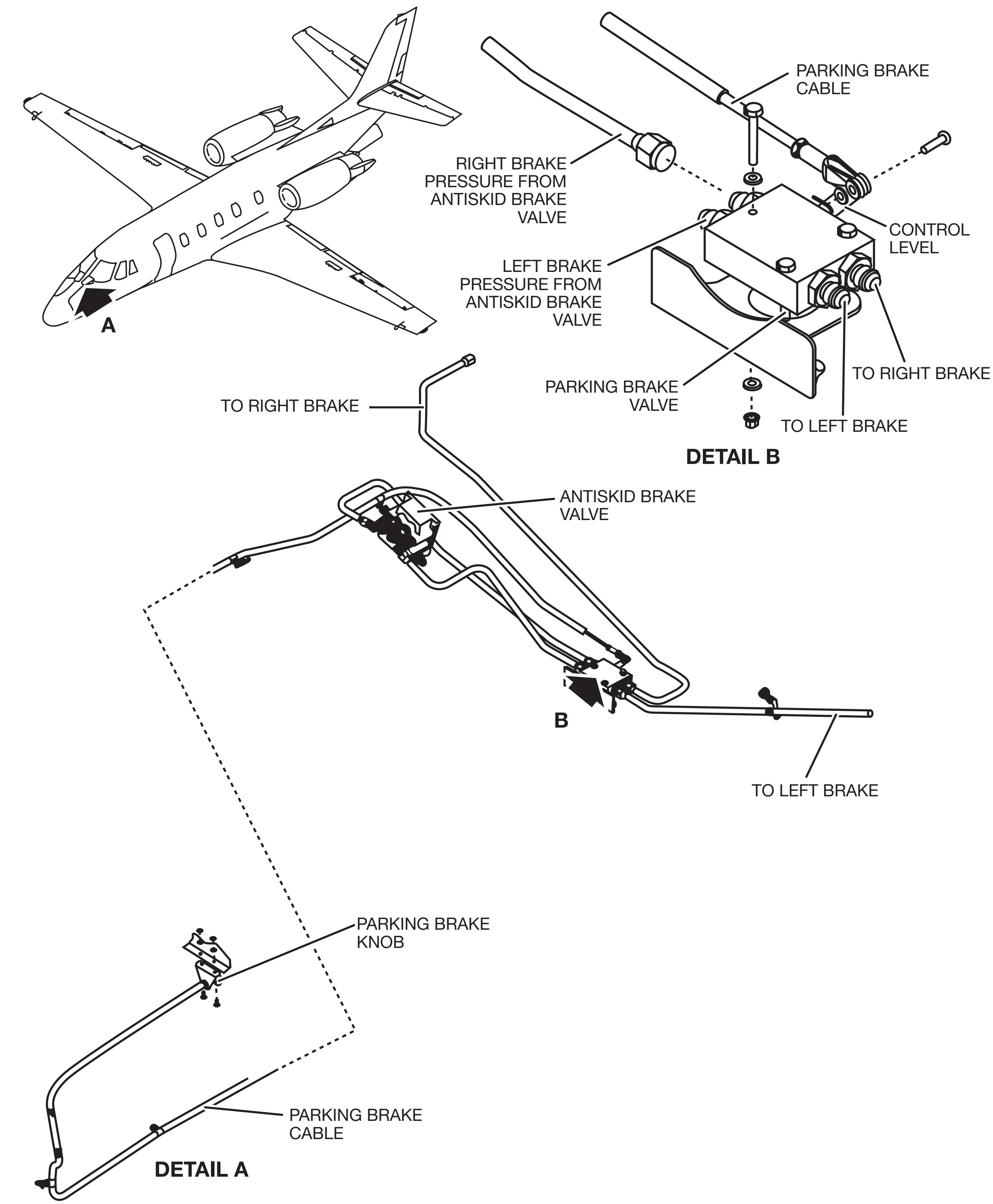


Figure 32-35. Park Brake Components

Antiskid Control Unit

The antiskid control unit consists of circuit board circuitry in a cast aluminum box. All electrical connections are made through wire bundles, which attach to two mating connectors on the antiskid control unit.

The antiskid control unit receives the output signals from the left and right wheel speed transducers and converts these signals to a DC voltage directly proportional to wheel speed. An accurate simulation of airplane velocity is available in addition to instantaneous wheel velocity.

The left and right wheel speed velocity voltages are then averaged to provide a composite signal that is used through the remainder of the control circuit functions. The wheel velocity signal, in conjunction with the reference deceleration control, continuously updates the reference velocity circuit. A velocity comparator circuit monitors the difference between the slowly changing reference velocity and the instantaneous wheel velocity signal. If either or both tires start to skid (in excess of the optimum skid required for effective braking), an error signal is generated.

The antiskid control unit, reacting to the error signal, supplies an output current signal (0.5 to 38.0 milliamperes) to the antiskid control valve in proportion to the severity of the wheel skid.

BIT System

The antiskid system has a built-in-test (BIT) system with BITE indicators, on a fault display unit on the left side in the nose compartment. The indicators latch to a tripped position when a fault is noted in the associated component. It is a function of the antiskid control unit to trip the respective BITE indicator on the fault display unit. The BIT system performs essentially three levels of testing:

- Power-up system initialization
- Continuous monitoring
- An initiated dynamic BIT

Functions monitored by the fault display unit are:

- Left/right wheel speed transducer transmitter
- Left/right main gear squat switches disagreement
- Antiskid servo valve fault
- A digital antiskid control unit fault.

The BIT system operates on a high-low voltage detection principal. When a short or open fault occurs in the servo or wheel speed generator circuits or an antiskid control circuit power fault is sensed, the ANTISKID INOP annunciator illuminates, warning the operator of the condition, and the respective BITE indicator on the fault display unit flags.

Parking Brake Valve

The parking brake valve is in the brake hydraulic pressure line behind the belly fairings on the aft side, center area of the first bulkhead forward of the wing (Figure 32-35).

Depress the brake pedals and set the parking brake control and release brake pedals. The parking brake control valve captures and holds hydraulic pressure on the main gear wheel brake assemblies. To release the parking brake, release the parking brake control. This releases the hydraulic pressure.

CAUTION

Electrical power must be turned on and the power brake system must be operative to set the park brake. This ensures that hydraulic pressure is available to the brake assemblies.

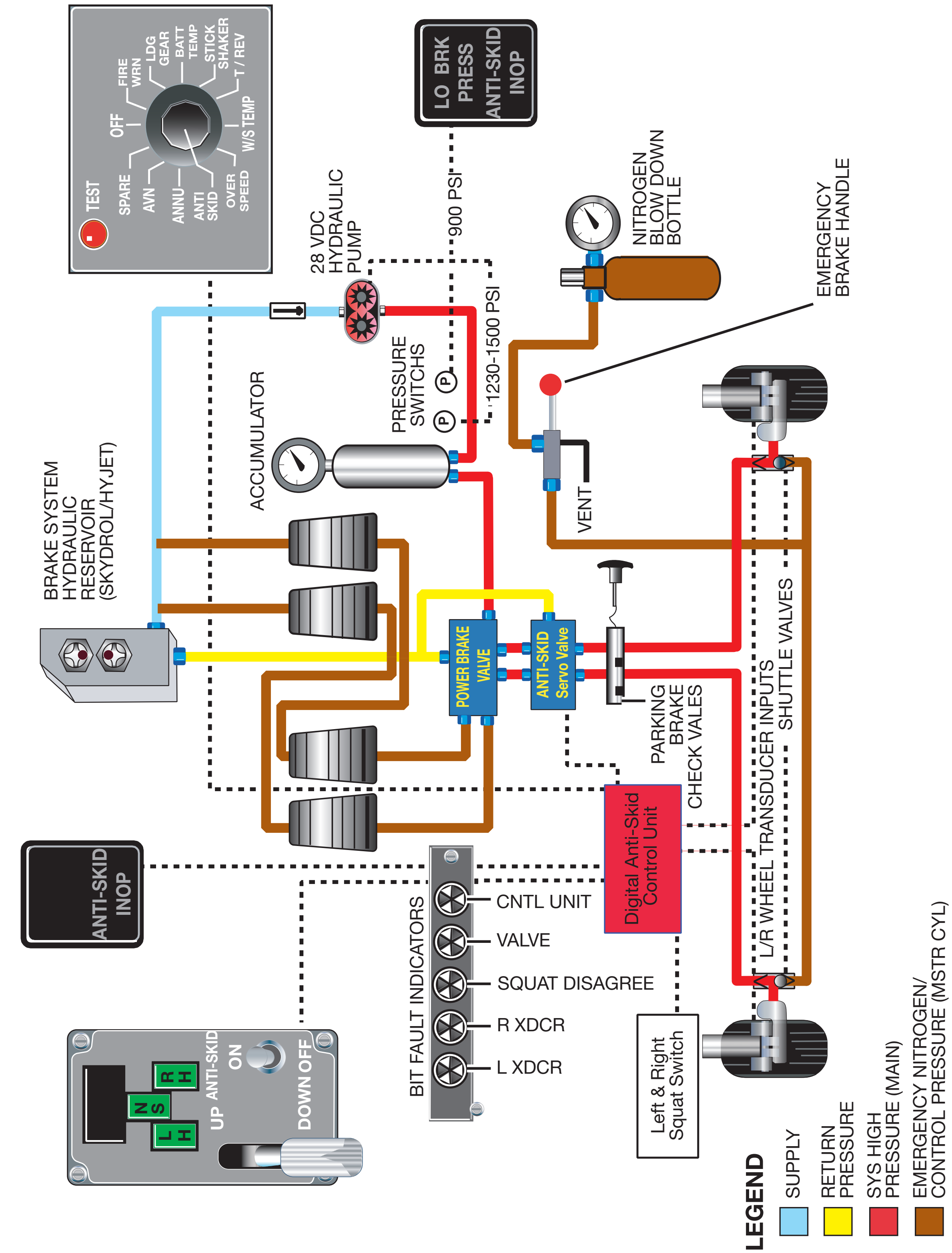


Figure 32-36. Power Brake/Antiskid System—XL

Brake Assemblies

The brake assemblies are piston-operated multiple disk brakes. The brake assembly consists of three primary parts:

- The piston housing assembly
- The heat sink
- A torque plate assembly

The aluminum piston housing assembly contains:

- A shuttle valve assembly
- Four adjuster assemblies
- Five piston sleeves and pistons
- Two bleeder plugs

A hydraulic shuttle valve assembly is attached to the piston housing assembly, which provides for emergency operation of the brake if the hydraulic pressure fails.

BRAKE SYSTEM OPERATION

XL

With the aircraft DC electrical bus energized, the POWER BRAKES circuit breaker engaged, and the landing gear handle in down position, the brake hydraulic pump operates until the brake accumulator pressure reaches 1500, ± 50 psi, as indicated on the accumulator pressure gauge (Figure 32-36). When the accumulator pressure falls below 1230 psi, the brake hydraulic pump energizes and increases brake accumulator pressure to 1500 psi, ± 50 psi as indicated on the accumulator pressure gauge.

The brake master cylinders initiate all braking action, with the exception of emergency braking. Displacement of the pistons in the master brake cylinders transfer hydraulic pressure to the power brake valve. If brake system high pressure is available, the master cylinder input pressure causes the power brake valve to apply a high-pressure output proportional to the master cylinder input. The hydraulic pressure is transferred to the antiskid valve and through the parking brake valve to the main landing gear brake assembly.

The antiskid function becomes operational when the ANTI-SKID ON–OFF control switch is in the ON position. This activates the antiskid valve. The antiskid wheel speed transducers supply pulse signals to the digital antiskid control unit. The control unit transmits signals to the antiskid valve, which controls and reduces the hydraulic pressure applied to the main landing gear wheel brake assemblies when required to avoid skidding.

NOTES

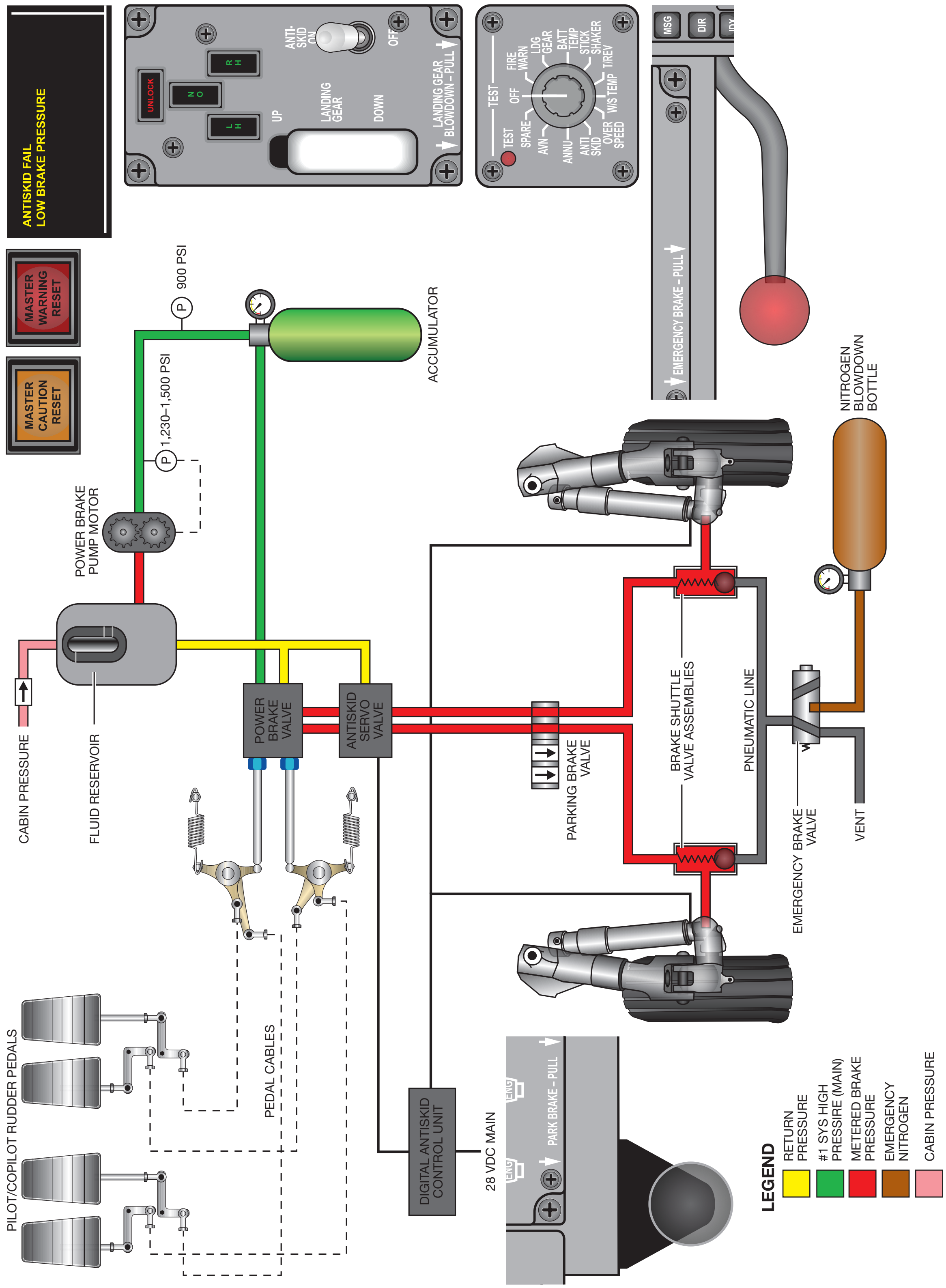


Figure 32-37. Power Brake/Antiskid System—XLS/XLS+

XLS/XLS+

When electrical power is supplied to the brake system, the hydraulic pump assembly comes on and charges the accumulator and pressurizes the brake system (Figure 32-37). This pressure stays upstream of the brake metering valve until the pilot or copilot actuates one of the brake pedals.

The brake control system moves the correct cables to actuate the left or right lever on the brake metering valve (refer to Brake Control System—Description and Operation). When the levers are actuated, the pressurized fluid is sent to the antiskid valve and out to the wheel at a pressure that is determined by the amount of pressure applied by the pilot or copilot.

If the antiskid senses a skid in one or both of the wheels, the antiskid valve will receive a signal from the antiskid controller to reduce the fluid pressure to that wheel. They hydraulic fluid is sent back to the brake reservoir until it enters the system again.

The parking brake valve is open until the handle in the cockpit is actuated. When the parking brake valve is closed, it operates like a check valve. Fluid can be sent out to the brake assemblies but can not return until the parking brake valve is opened.

NOTES

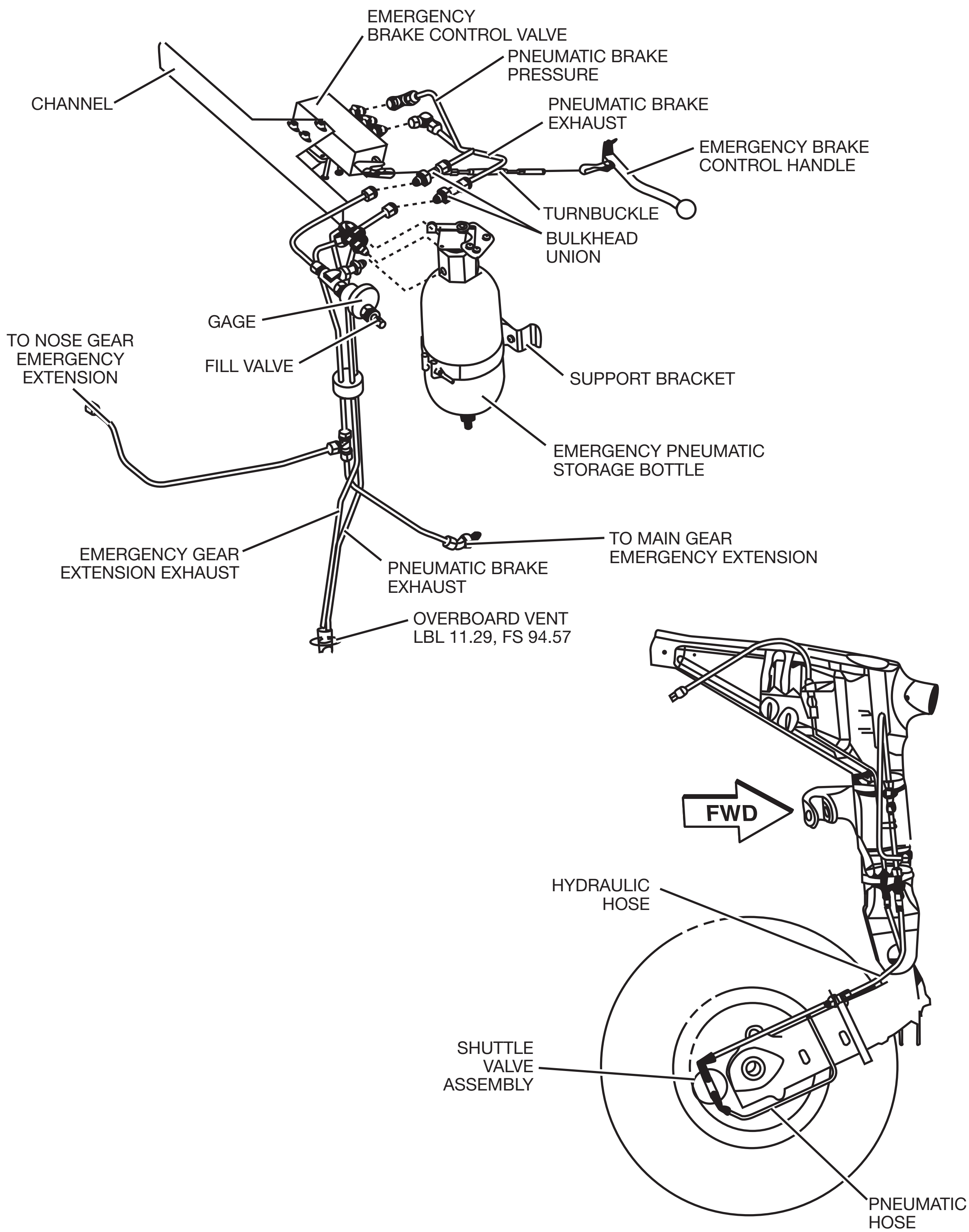


Figure 32-38. Emergency Brake

EMERGENCY BRAKE SYSTEM

The emergency braking action is accomplished by releasing high-pressure nitrogen to the main landing gear wheel brake assemblies (Figure 32-38). High-pressure nitrogen is released from the pneumatic storage bottle by pulling on the emergency brake valve handle. The high-pressure nitrogen is then directed to the main landing gear wheel brake assemblies, where it shifts a shuttle valve in the main landing gear wheel brake assemblies and nitrogen engages the brakes.

Moving the emergency brake valve handle toward the stowed position stops the flow of high-pressure nitrogen from the pneumatic storage bottle and vents the high-pressure nitrogen in the main gear wheel brake assembly overboard, releasing the brakes.

Manipulating the emergency brake valve handle provides effective emergency braking control.

The pneumatic storage bottle volume is sufficient for many individual braking applications.

Emergency Brake Control Valve

There is an emergency brake control valve immediately below the left instrument panel, left of the emergency gears release. Pneumatic pressure is obtained from the landing gear emergency extension pneumatic storage bottle. The main landing gear wheel brake assembly accepts the pneumatic pressure for emergency braking.

NOTE

When high-pressure air is released into the main gear wheel brake system, the hydraulic brake system must be bled before the next flight.

NOTES

QUESTIONS

1. On the ground, the LANDING GEAR handle is prevented from movement to the UP position by:

A. Mechanical detents
B. A spring-loaded locking solenoid
C. Hydraulic pressure
D. A manually applied handle locking device

2. The landing gear uplock mechanisms are:

A. Mechanically held engaged
B. Hydraulically disengaged normally; or pneumatically released in an emergency
C. Electrically engaged and disengaged
D. Both A and B

3. Landing gear down locks are disengaged:

A. When hydraulic pressure is applied to the retract side of the gear actuators
B. By action of the gear squat switches
C. By removing the external down-lock pins
D. By mechanical linkage as the gear begins to retract

4. Each main gear wheel incorporates a fusible plug that:

A. Blows out if the tire is overserviced with air
B. Melts, deflating the tire if an overheated brake creates excessive tire pressure
C. Is thrown out by centrifugal force if maximum wheel speed is exceeded.
D. None of the above

5. At retraction, if the nose gear does not lock in the UP position, the gear panel light indication is:

A. Red light on, green left and right lights
B. Red light off, green left and right lights on
C. Red light on, all three green lights off
D. All four lights off

6. The gear warning horn sounds when one or more gear are not down and locked and:

A. Flaps are extended beyond the 15° position–both throttles retarded below 70% N_2
B. Airspeed is less than 150KIAS
C. Either throttle is retarded below 70% N_2 rpm
D. Both throttles are retarded below 0% N_2 rpm and airspeed is >150 KIAS

7. When the LANDING GEAR handle is positioned either UP or DOWN:

A. The hydraulic system control valve is energized open
B. The hydraulic system control valve is energized closed
C. The hydraulic system control valve is not affected
D. The amber HYDRAULIC PRESSURE CAS message does not display

8. Emergency extension of the landing gear is accomplished by actuation of:

A. A switch for uplock release and application of air pressure
B. One manual control to release the uplocks and apply air pressure for extension
C. Two manual controls–one to mechanically release the uplocks, the other to apply air pressure for gear extension and down-locking
D. None of the above

9. The power brake valve is actuated (XLS/XLS+):

A. Mechanically by brake pedal pressure
B. Mechanically by the emergency airbrake control lever
C. Hydraulically by brake pedal pressure
D. Automatically at touchdown

10. Do not depress the brake pedals while simultaneously using the emergency brake system because:

A. Manual braking overrides the air brakes
B. The shuttle valve may allow air pressure into the brake reservoir, rupturing it or causing uncommanded differential braking
C. The shuttle valve moves to the neutral position after which no braking action occurs
D. The brakes become spongy

11. The DC motor-driven hydraulic pump in the brake system operates:

A. During the entire time the LANDING GEAR handing is in the DOWN position
B. As needed with the LANDING GEAR handle DOWN in order to maintain system pressure
C. Only when the amber LOW BRAKE PRESSURE CAS message displays
D. Even when the LANDING GEAR handle is UP to keep air out of the system as the aircraft climbs to altitude.

12. Appearance of the amber LOW BRAKE PRESSURE CAS message indicates:

A. Power brakes are still operational
B. The normal braking system may not work as well as normal, and the antiskid is still operational
C. Power brakes are inoperative, use the thrust reverser to stop
D. Use the emergency braking system and increase stopping distance

13. During preflight, to get an accurate reading on the brake reservoir and precharge indication, you must:

A. Turn the aircraft battery ON
B. Using the brake pressure bleed valve, bleed the pressure to the precharge level
C. Pull the POWER BRAKE CB
D. If the amber LOW BRAKE PRESSURE CAS message is not displayed, no other action is required

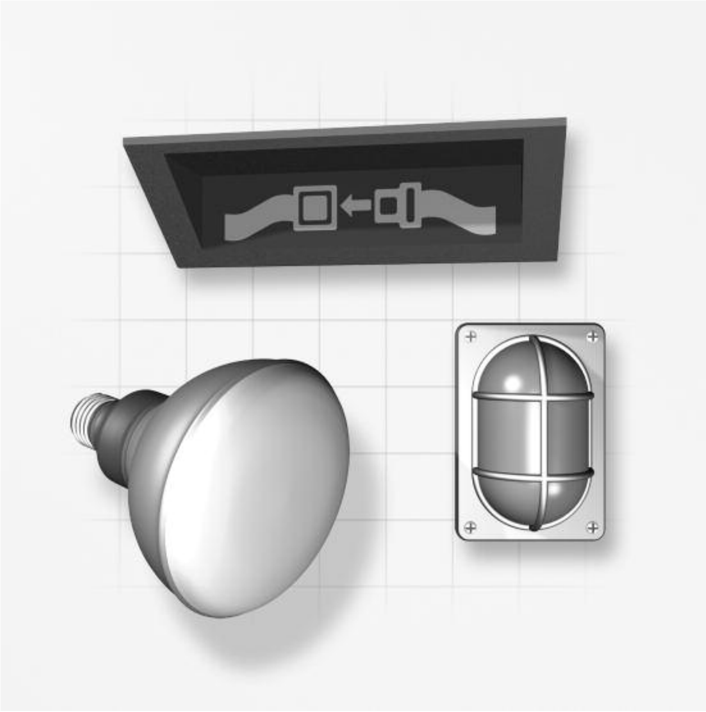
CHAPTER 33
LIGHTS

CONTENTS

ILLUSTRATIONS

TABLE

CHAPTER 33
LIGHTS



INTRODUCTION

This chapter describes those units and components which provide for external and internal illumination on the Citation XL/XLS/XLS+. This chapter does not include lights for individual systems. The information in this chapter must be used in conjunction with *Citation XL/XLS/XLS+ Wiring Diagram Manual* to correlate data required to maintain the lighting systems that illuminate the interior and exterior of the airplane.

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GENERAL

The Citation XL/XLS/XLS+ lighting consists of four major groups: interior, emergency, exterior, and tailcone lighting. Switches and rheostats control all lighting. Separate circuit breakers in the cockpit or tailcone J-box protect the system.

Interior lighting consists of direct, indirect,fluorescent, and incandescent lighting for the cockpit and cabin.

Emergency lighting is a separate independent system that provides automatic illumination in case of main DC electrical power failure, a +5G impact, or illumination of passenger safety lights.

Exterior lighting consists of lights for landing, taxi, recognition, anticollision, wing inspection, tail flood, and ground recognition beacon.

Tai1cone lighting consists of interior lighting in the tailcone and baggage compartment areas.

The Citation XLS/XLS+ incorporate light emitting diode (LED) lights to replace incandescent bulbs installed on the XL. LED lights operate at reduced temperatures that increase the life of the light assembly and reduce the possibility of heat damage to adjacent objects.

The following is a list of Citation XLS+ LED light assemblies:

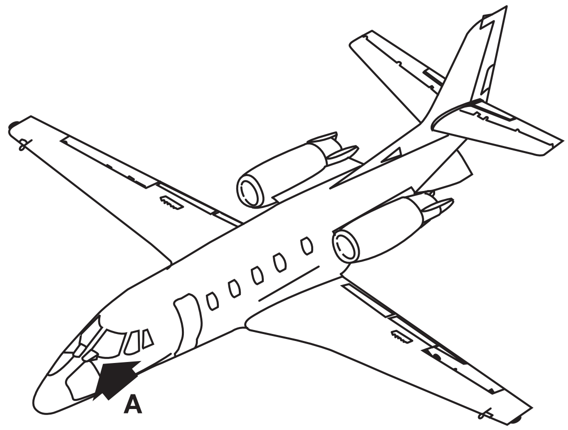
- Accent lights in refreshment storage area
- Aft vanity lights (indirect lighting and furniture lighting) (halogen over toilet)
- Cabin divider lamps
- Cabin drop aisle emergency lights
- Cabin indirect lighting
- Emergency egress lights (overwing) (units 5560 and subsequent)
- Emergency exit signs
- Navigation light tail
- All other panel lights, i.e., vanity switch panel control, entry door light panel control

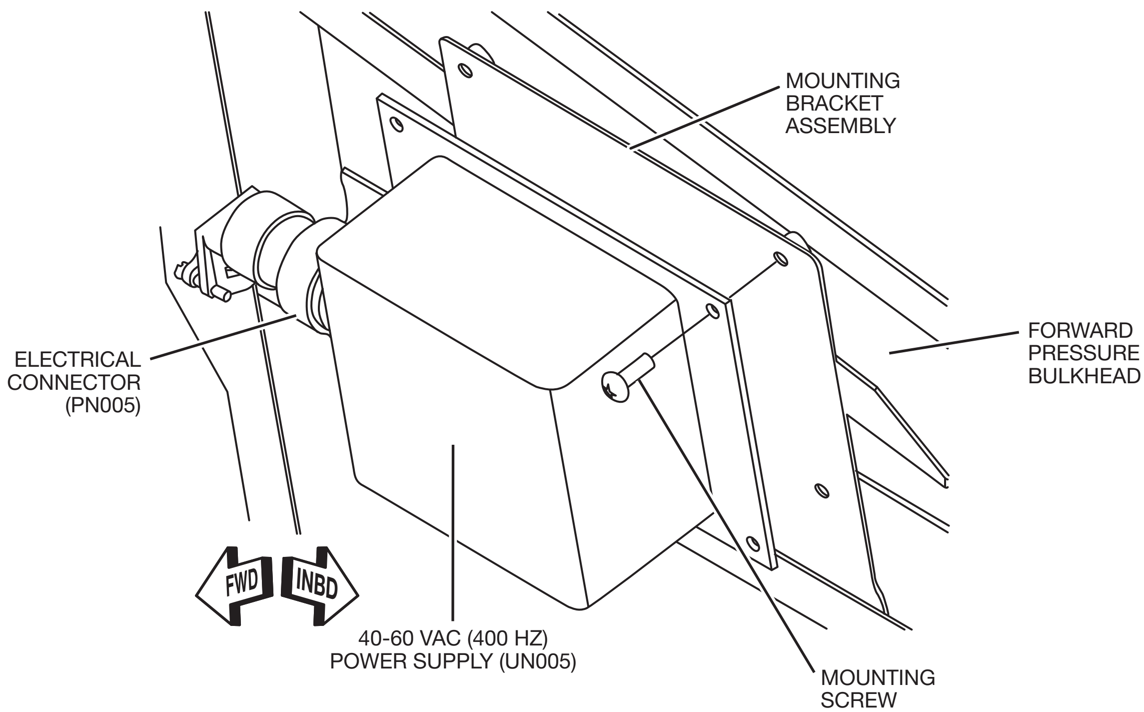
The XLS+ also incorporates LED anticollision lights, navigation lights, ground recognition lights, and AUX panel lights.

DESCRIPTION

The chapter is divided into sections to aid maintenance personnel in locating information. A brief description of each section follows:

- The section on flight compartment lighting provides system description and operation, troubleshooting procedures and maintenance practices for flight compartment primary lights, cockpit flood lights and map lights.
- The section on passenger compartment lighting provides system description and operation, troubleshooting and maintenance practices for the indirect fluorescent cabin lights, passenger reading and entrance lights and passenger sign lights.
- The section on cargo and service compartment lighting provides illumination for the nose baggage compartment and the tail cone compartment.
- The section on exterior lighting describes the lighting system used to provide illumination outside the airplane. This includes systems such as landing, navigation, recognition and the flood lights.
- The section on emergency lighting describes separate and independent systems used to provide illumination in case of primary electrical power failure or abnormal conditions.





DETAIL A

Figure 33-1. 40–60 VAC (400 Hz) Power Supply

FLIGHT COMPARTMENT LIGHTING

DESCRIPTION

The flight compartment lighting is subdivided into primary lighting, secondary lighting and warning lights. This lighting provides general and local lighting for the flight compartment area.

PRIMARY LIGHTING

Most instruments are internally lighted. Instruments and switches not illuminated internally or by an electroluminescent panel may be illuminated by a post light. The area of illumination includes:

- Instrument panel
- Side consoles
- Control pedestal

The internally lighted instruments utilize 5 volts direct current (VDC) power. Some post lights are 5 VDC while others may require 28 VDC. Consult the *Citation XL/XLS/XLS+ Wiring Diagram Manual* for information on instrument lighting.

The internally lighted instruments receive the 5-volt power from an inverter. An ON/OFF switch, supplies 28 VDC to all flight compartment lighting inverters. The internally lighted instruments utilize three inverters. Each inverter is controlled from dim to bright by a voltage control rheostat on the instrument panel.

For Maintenance on fire tray lights, refer to Chapter 26—"Engine Fire Tray Assembly" in the *Airplane Maintenance Manual* (*AMM)*.

Electroluminescent panels are used to illuminate position functions of switches and controls. The panels are on the:

- Instrument panel
- Side consoles
- Control pedestal

The electroluminescent panels utilize 40 to 60 volts alternating current (VAC) 400 Hertz power (Figure 33-1). Consult the *Citation XL/XLS/XLS+ Wiring Diagram Manual* for information on instrument lighting.

In all primary lighting, the lights have a dim to bright illumination pattern plus a PANEL LIGHT power ON/OFF switch (SI002).

SECONDARY LIGHTING

The secondary lighting includes the two floodlights and two overhead map lights. The floodlights are 28 VDC lights, overhead, used to illuminate the instrument panel. The two floodlights are controlled by a rheostat. The two overhead map lights are powered by 28 VDC and are individually controlled by a rheostat.

Refer to Chapter 26—"Engine Fire Tray Assembly" in the *AMM*, for maintenance on the fire tray lights.

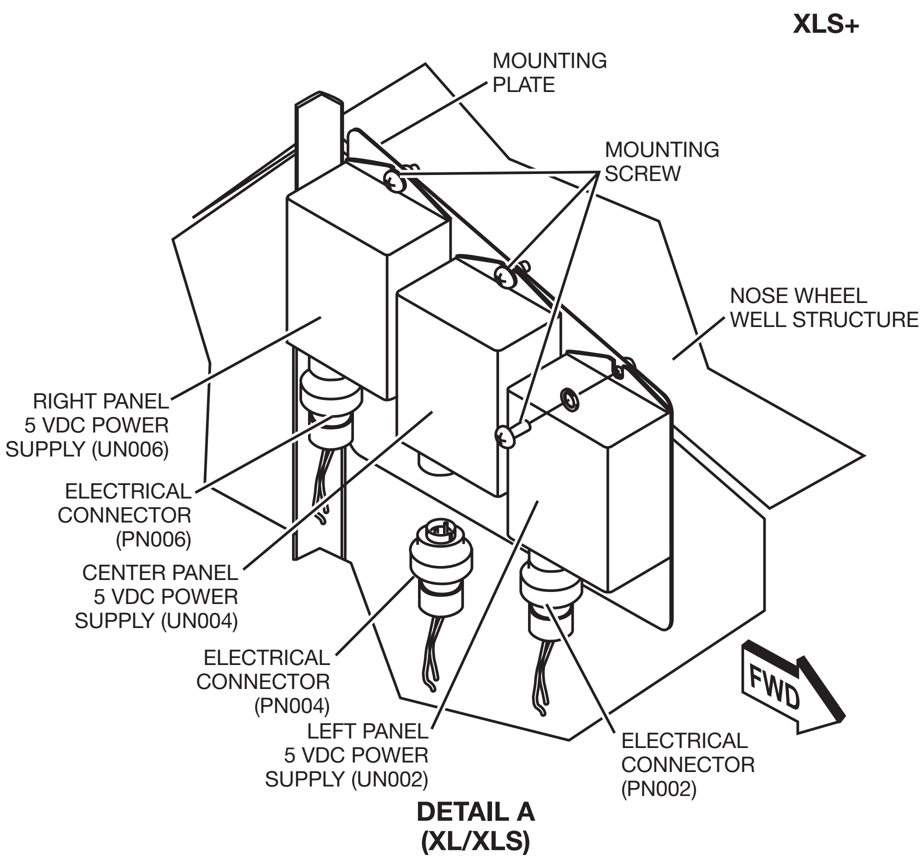


Figure 33-2. 5 VDC Lighting Power Supply

FLIGHT COMPARTMENT PRIMARY LIGHTING

Description

The control panel lighting is provided by the electroluminescent panels, consisting of a layer of phosphor sandwich between two electrodes encapsulated between layers of plastic. One end of the electrical wires (pigtail) is seated in the panel. White lettering on grey background is used on the panel faces.

The electroluminescent panels receive 40 to 60 VAC 400-hertz power from an inverter. The panel lights switch or interrupt the 28 VDC supply to the electroluminescent panel inverter. A single inverter voltage control rheostat controls dim to bright voltage to all the electroluminescent panels in the flight compartment.

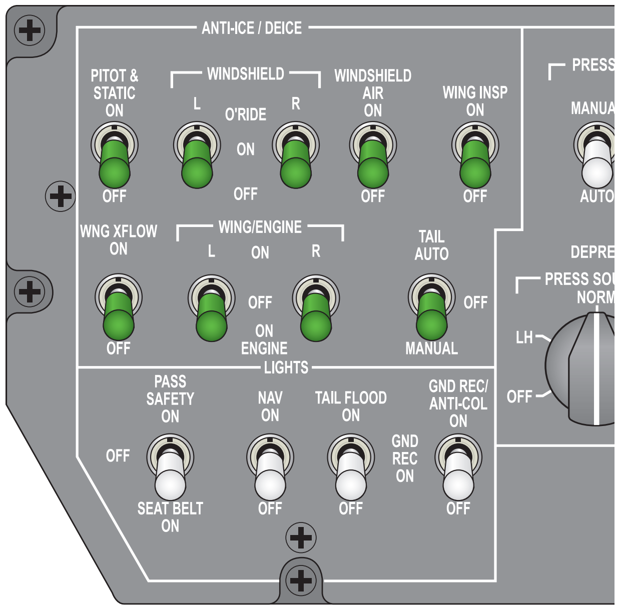
A damaged or malfunctioning electroluminescent indicator requires replacing the entire panel. Electroluminescent panel lighting is used on:

- Circuit breaker panels
- Switch panels
- Lighting control panel
- Environmental control panel
- Landing gear control panel
- Throttle quadrant panel
- Pedestal flight control panel
- Aux gear control panel
- MIC OXY MASK panel

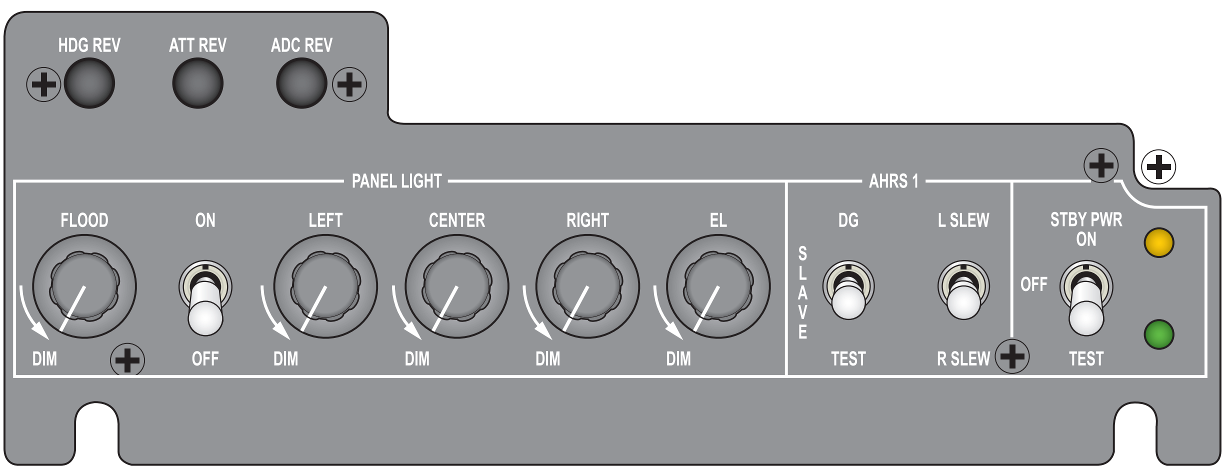
Electrical power to the light panels is supplied by a 40 to 60 VAC inverter (UN005), in the left side of the nose compartment, on the forward pressure bulkhead. The inverter incorporates a two-ampere overcurrent protection circuit. A shorted or pinched wire in the system shuts the system off. When the cause of the overcurrent is eliminated, the inverter functions normally. The electroluminescent panels are dimmed by rotating the dimming control knob counterclockwise on the instrument panel(s).

Instruments in the instrument panel have internal 5 volt DC lighting circuits. Electrical power is supplied to the instruments via 5 VDC lighting power supplies. The airplane utilizes three, 5 VDC lighting power supplies for instruments lighting. The three 5 VDC lighting power supplies are on right side of nose compartment (nose wheel well structure between FS 58.30 and FS 44.00) (Figure 33-2):

- XL/XLS:
  - One 5 VDC lighting power supply (UN002) supplies instrument lighting power to the left instrument panel
  - Another 5 VDC lighting power supply (UN004) supplies instrument lighting power to the center instrument panel
  - The third 5 VDC lighting power supply (UN006) supplies instrument lighting power to the right instrument panel
- XLS+:
  - 5 VDC lighting power supply (UN002) supplies power to the elevator quadrant EL panel and flap quadrant EL panel
  - 5 VDC lighting power supply (UN004) supplies power to the oxygen gauge and cabin differential pressure gauge lighting.



LIGHT SWITCH PANEL



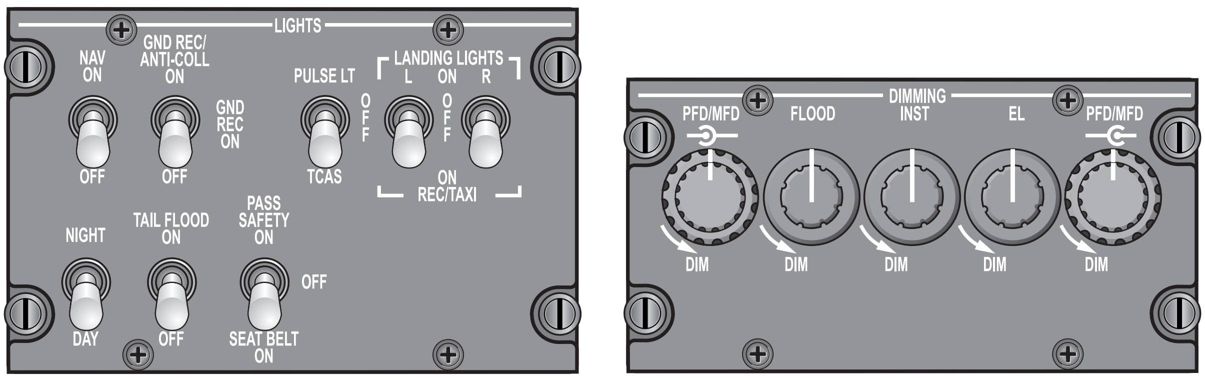
COCKPIT PANEL LIGHTS

Figure 33-3. XL/XLS Light Panels

Avionic control panels are included in the internal 5 VDC lighting circuits. Location of the instrument panels normally dictates which 5 VDC lighting power supply supplies power.

COCKPIT PANEL LIGHTS

Main DC electrical supplies power to the cockpit panel lights. The master panel light ON–OFF toggle switch on the pilot lower instrument panel for XL/XLS aircraft (Figure 33-3) or center pedestal for XLS+ aircraft (Figure 33-4) controls the lights.



LIGHTS SWITCH PANEL **COCKPIT LIGHTS PANEL**

Figure 33-4. XLS+ Light Panels

Table 33-1. DAY-NIGHT SWITCH—ON

XL	XLS	XLS+
ANNUNCIATOR PANEL DIMS	ANNUNCIATOR PANEL DIMS	STAND ALONE ANNUNCIATORS DIM
MASTER CAUTION LIGHTS DIM	MASTER CAUTION LIGHTS DIM	MASTER CAUTION LIGHTS DIM
T/R ANNUNCIATORS DIM	T/R ANNUNCIATORS DIM	T/R ANNUNCIATORS DIM
GREEN IGNITION LIGHTS DIM		
APU RELAY ENGAGED ANNUNCIATOR DIMS	APU RELAY ENGAGED ANNUNCIATOR DIMS	APU RELAY ENGAGED ANNUNCIATOR DIMS
LEFT, CENTER, RIGHT AND EL RHEOSTATS ACTIVATED	LEFT, CENTER, RIGHT AND EL RHEOSTATS ACTIVATED	LEFT, CENTER, RIGHT AND EL RHEOSTATS ACTIVATED
RED ICE DETECT LIGHTS ILLUMINATE	RED ICE DETECT LIGHTS ILLUMINATE	RED ICE DETECT LIGHTS ILLUMINATE
THROTTLE DETENT INDICATORS DIM		
ANNUNCIATOR LIGHTS ABOVE RMUS DIM	ANNUNCIATOR LIGHTS ABOVE RMUS DIM	
APU FIRE LIGHT DIMS	APU FIRE LIGHT DIMS	APU FIRE LIGHT DIMS
APU CONTROL PANEL DIGITS DIM	APU CONTROL PANEL DIGITS DIM	APU CONTROL PANEL DIGITS DIM
APU FAIL ANNUNCIATOR DIMS	APU FAIL ANNUNCIATOR DIMS	

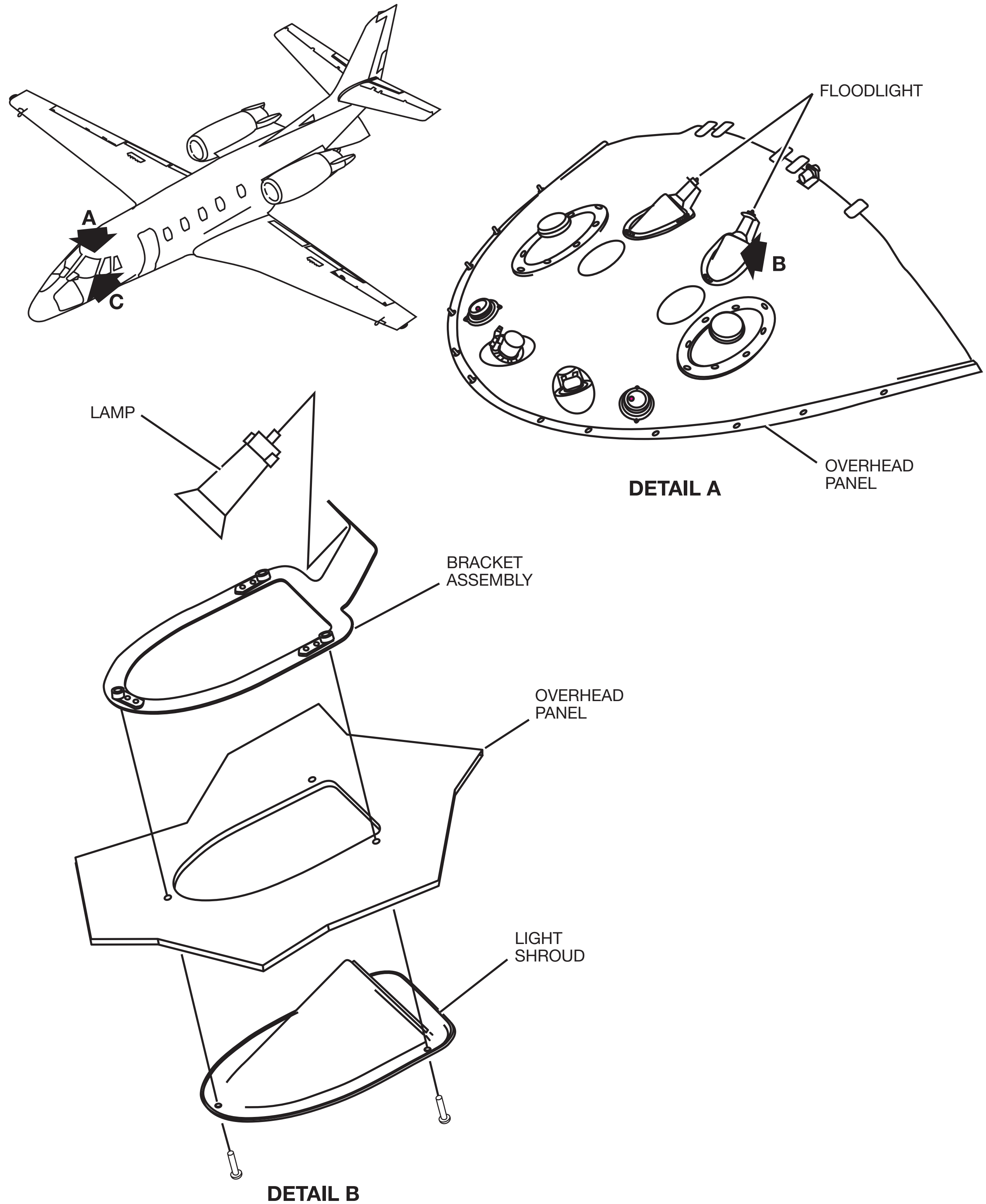


Figure 33-5. Cockpit Floodlight and Indicator Lights

COCKPIT FLOODLIGHTS AND INDICATOR LIGHTS

Description

The cockpit floodlights (FC013 left and FF012 right) and indicator lights provide cockpit and instrument lighting when the floodlight switch is on. The two cockpit floodlights are in the flight compartment overhead panel (at FS 157.00 near the airplane centerline) (Figure 33-5).

The floodlights and map lights are 28 VDC overhead lights. A single rheostat (RI005) controls both floodlights. The rheostat is equipped with an ON/OFF position. The floodlights normally operate during thunderstorms. The map lights are individually operated with an ON/OFF rheostat. A map light illuminates a particular area at the respective flight station.

An indicator light in the fire tray under the glareshield, provides illumination for the center portion of the instrument panel.

For maintenance on fire tray lights, refer to Chapter 26—"Engine Fire Tray Assembly."

For additional wiring information, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

CHART LIGHTS

Description

There are two chart lights in the flight compartment overhead panel (headliner). The light brightness for the chart lights (FC002 left and FF010 right) is controlled by rheostat switches (RC002 left and RF003 right) on the left and right CB panels (Figure 33-6).

Refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual* for additional information.

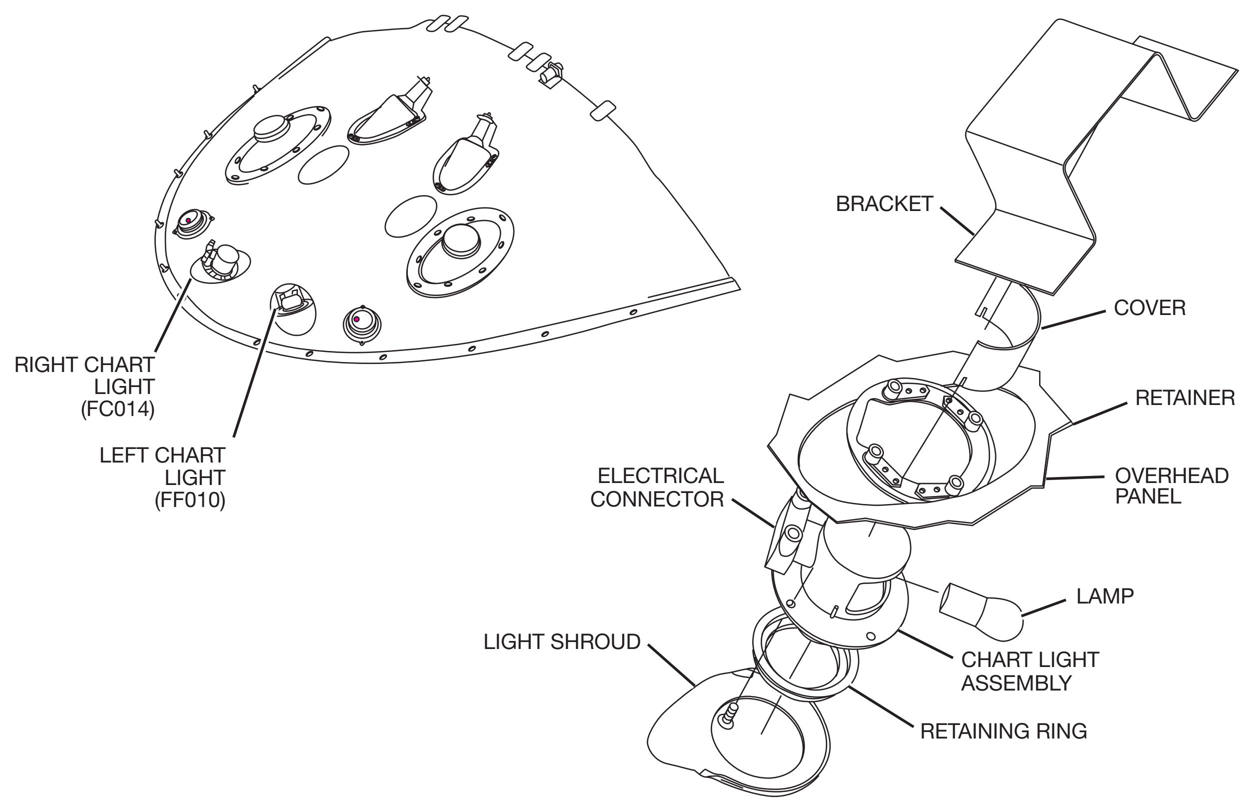


Figure 33-6. Chart Lights

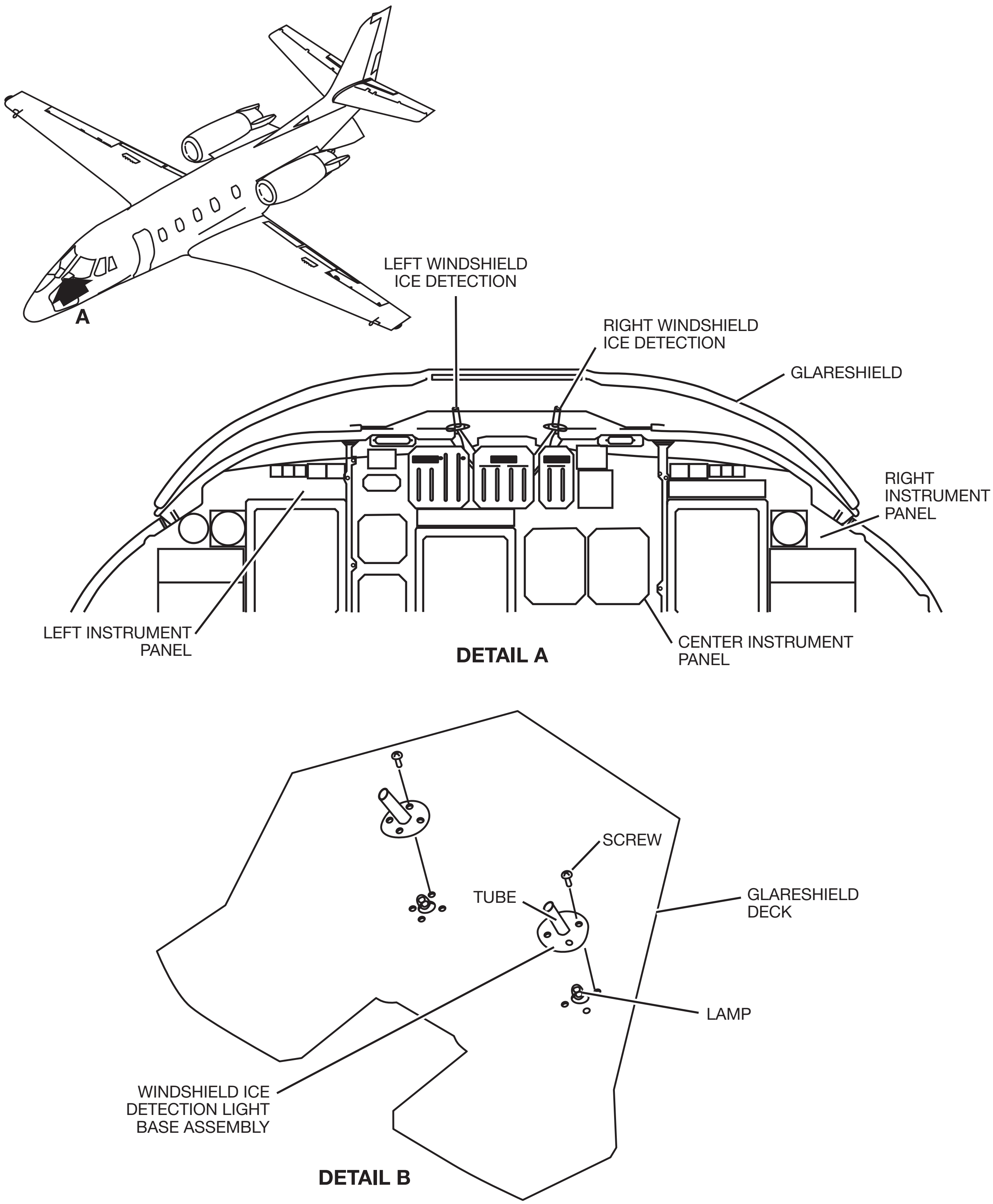


Figure 33-7. Windshield Ice Detection Lighting

WINDSHIELD ICE DETECTION LIGHT

Description

Two windshield ice detection light assemblies are on the deck forward of the glareshield aimed at the windshield (Figure 33-7).

The windshield ice detection lights (FF016 right and FC017 left) are powered-on any time the left instrument PANEL LIGHT switch (SI005) is in the ON position. The windshield ice detection lights are not visible to the pilot/copilot unless ice has formed on the left and/or right windshield.

NOTES

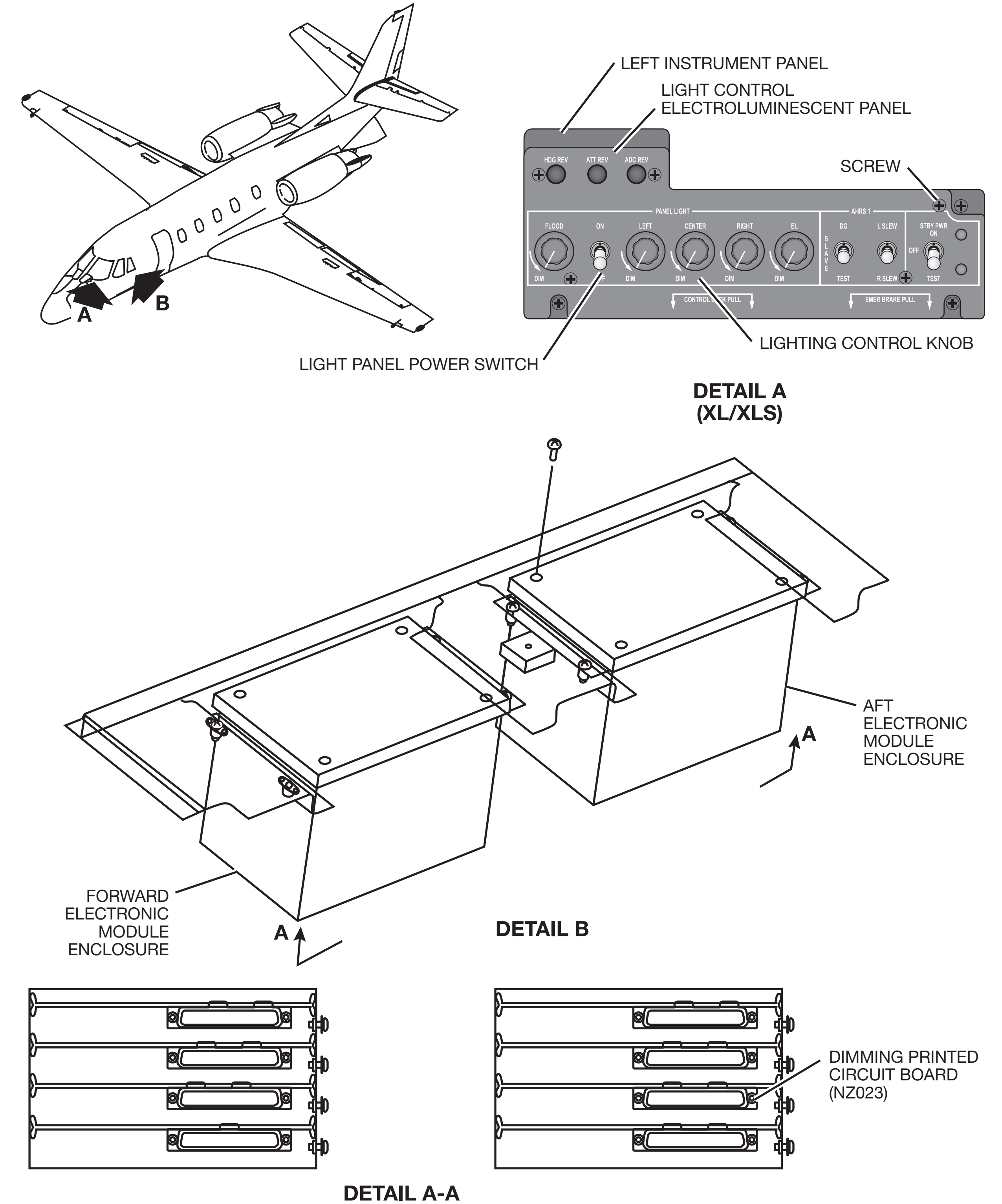


Figure 33-8. Dimming Control

DIMMING CONTROL

Description

A dimming control (printed circuit board) is in the forward electronic module enclosure unit on the left side console (Figure 33-8). The dimming control is utilized to provide dimming control to all the control lights.

A potentiometer (LEFT) on the light control electroluminescent panel at the bottom of the left instrument panel, consists of a large knob and small knob. Rotating the large knob controls dimming for the center instrument panel lights. Rotating the small knob clockwise out of detent, controls the manual display brightness of the controls.

Rotating the small knob full counterclockwise into detent, causes the display brightness of the controls to be automatically controlled by light sensors on the control panels.

NOTES

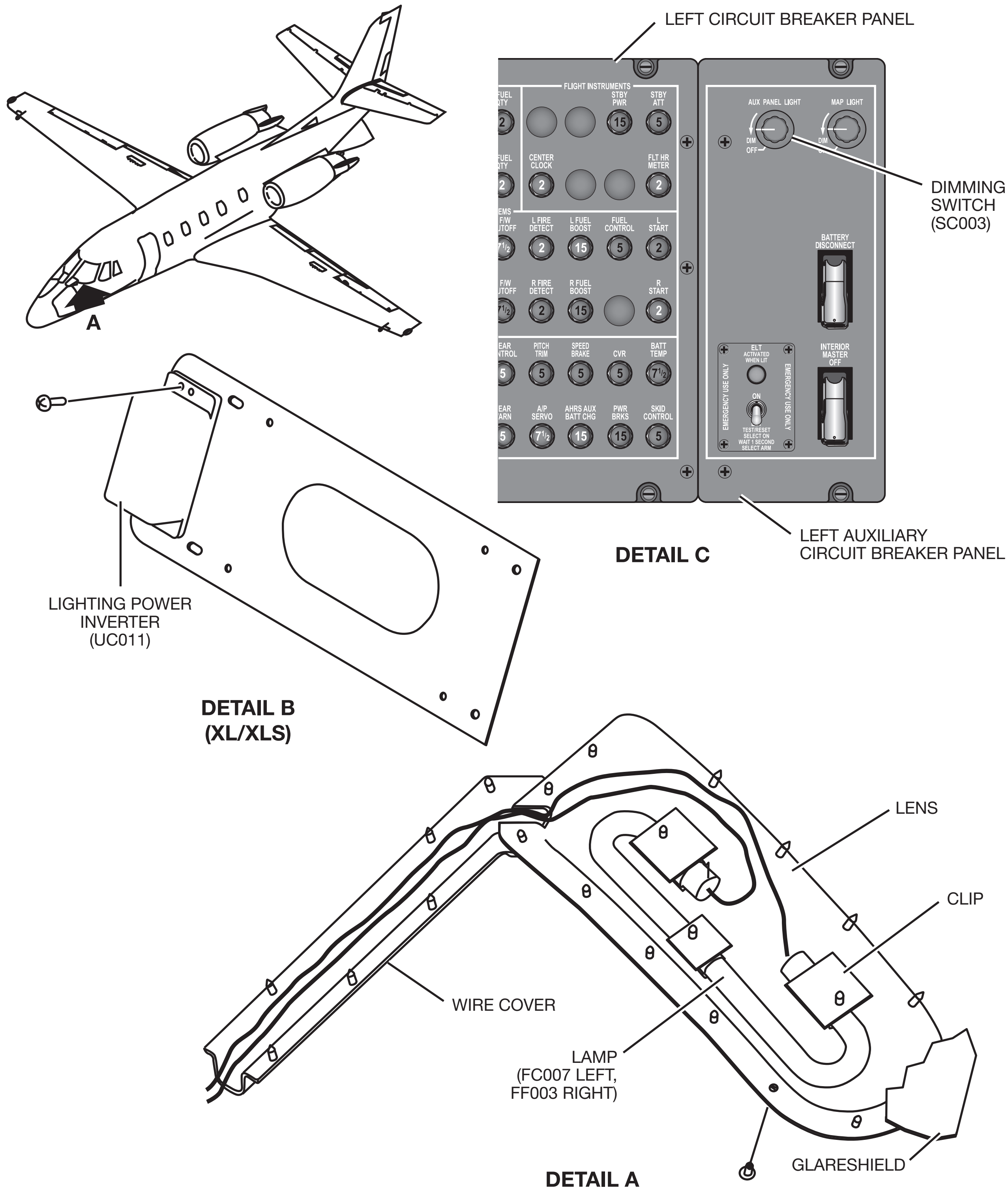


Figure 33-9. Supplemental Glareshield Lighting (XL/XLS)

SUPPLEMENTAL GLARESHIELD LIGHTING

Description

XL/XLS

Supplemental lighting is provided by two fluorescent lamps (FC007 left and FF003 right) under the glareshield. A high voltage power inverter in the pilot left side console to supplies the necessary DC voltage. The glareshield lighting dimming control switch AUX PANEL LIGHT (SC003) is on the left CB panel (Figure 33-9).

XLS+

Supplemental lighting is provided by two LED strips under the glareshield. 28 VDC is supplied through a rheostat on the subpanel forward of the pilot CB panel to control dimming.

NOTES

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PASSENGER COMPARTMENT LIGHTING

DESCRIPTION

The passenger compartment lighting includes:

- All cabin lights
- Utility lights
- Lighted signs

Indirect fluorescent (XL) or LED (XLS/XLS+) lights and passenger reading compartment lights are overhead.

There is indirect fluorescent (XL) or LED (XLS/XLS+) lighting on the left and right cabin sidewalls above the windows. The indirect lights can be dimmed by a rheostat switch in the forward frame of the main cabin door. The lights are on the top and bottom of the passenger service unit panels. For passenger service unit, refer to Chapter 25—"Passenger Service Unit" in the *AMM*.

XL

The indirect lighting lamps are a cool filament-type and are operable any time that 28 volts direct current (VDC) is available. Two rheostat switches control dimming through five power supplies that are overhead in the passenger service unit (PSU).

XLS/XLS+

Cabin indirect lighting consists of dual (upper and lower) LED lights. The overhead lights extend the full length of the cabin it two rows, one on each side of the PSU. The cabin indirect lights require main DC power and are turned ON or OFF by one of two switches. The first is on the entry light panel adjacent to the cabin entry door. The second switch is on the master switch panel on the refreshment center. Full dimming control of the indirect lighting is only available from the master switch panel.

There are cabin reading lights are in the passenger service units. Each light can be adjusted directionally and has a switch next to it for individual control.

An aft vanity light is provided and controlled by an independent electrical circuit. When a vanity and closet are installed, electrical power routes from the closet terminal board through a circuit breaker in the closet. The light switch is on the vanity.

There are lighted NO SMOKING/FASTEN SEAT BELT and EXIT signs throughout cabin area to aid in passenger safety and comfort. The lights are controlled by a switch on the pilot switch panel. The lighted signs let passengers know when they can/can not smoke and/or when to fasten seat belts. The seat belt/no smoking signs are controlled by a passenger safety switch on the pilot switch panel.

NOTE

An electronic chime is found on the right side of the passenger compartment at FS 261.85. A chime tone is produced by the dual remote amplifier to tell the passengers when smoking is not allowed and/or when to connect their seatbelts. Refer to Chapter 23—"Passenger Address Chimes" in the *AMM*.

There is dropped-aisle lighting along the left and right footwell sidewalls between FS 161.35 and FS 374.50. Cabin door threshold lights are in the main cabin door threshold and in the step risers. The dropped-aisle and threshold lights are controlled by an internal switch on the forward door frame above the indirect lighting rheostat switch.

For assistance in understanding the various systems, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

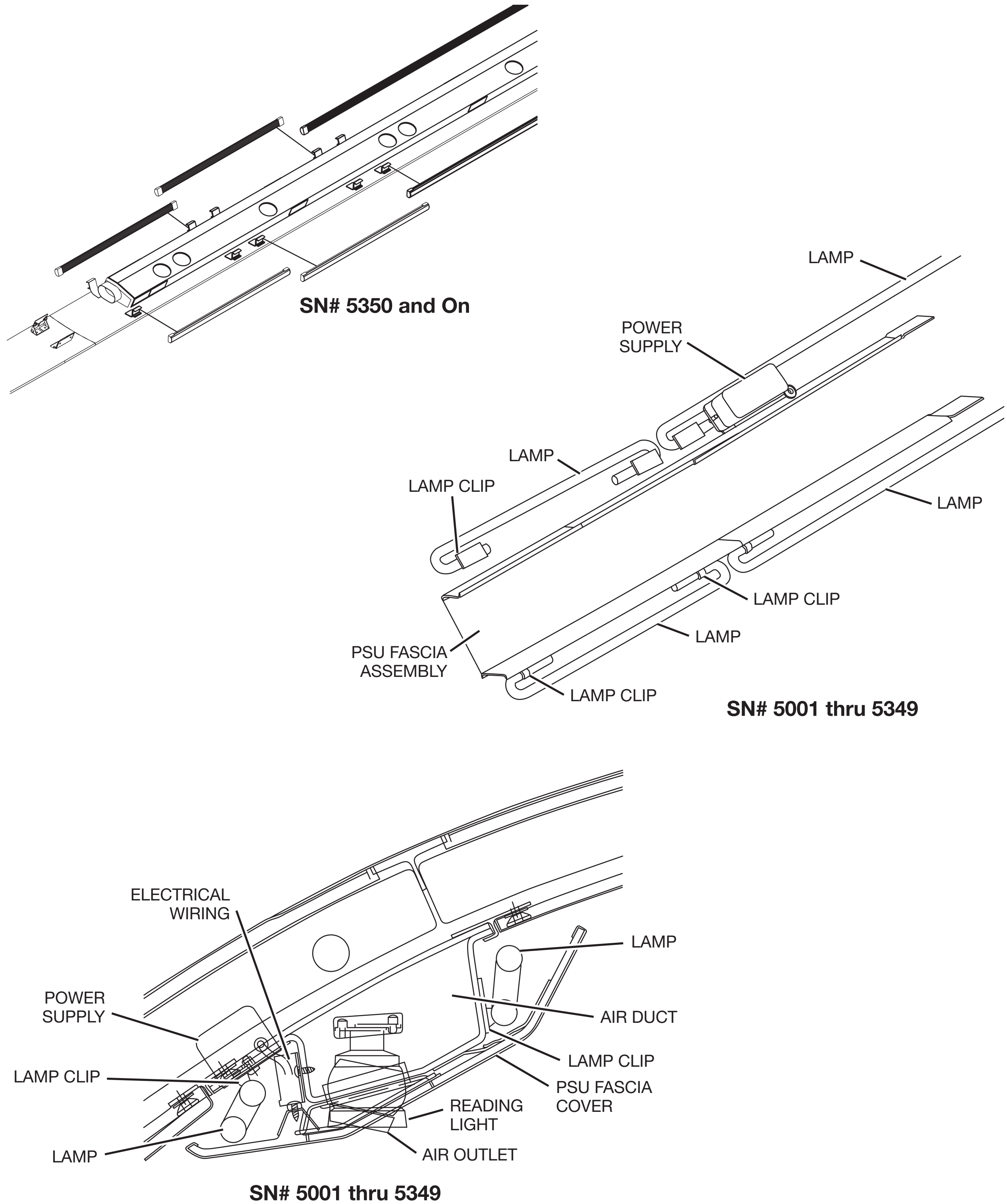


Figure 33-10. Indirect Lighting

CABIN INDIRECT LIGHTING

Description

XL

Cabin indirect lighting is provided by a system of dual (upper and lower) fluorescent lamps on the upper side(s) of the cabin above the windows (Figure 33-10).

The fluorescent tubes are a cold cathode-type lamp. This type of lamp is cooler operating, more shock resistant, and has a longer lamp life than a conventional fluorescent tube.

Groups of lamps are connected in series with individual power supply.

There is a door post control switch panel on the forward cabin entry door frame to allow dimming of the indirect lighting lamps.

Lamp failure causes a series of lamps to extinguish. If a lamp fails and a replacement is not immediately available, disconnect the electrical connector to the power supply, to prevent damage to the power supply.

Operation

The lamps on the top and bottom of the passenger service unit panels provide an upper/lower indirect lighting system. Some airplanes have a curved tube on the aft pressure bulkhead.

Dropped-aisle lighting power supplies are in the left and right footwell web (forward of FS 172.00 and aft of FS 368.68). Refer to "Passenger Compartment Lighting" in the *AMM*.

When various vanity and refreshment center options are installed, individual lamps are eliminated from installation. Components may be individually lighted with the same type of lamps. Dedicated power supplies for the lamps are inside the component.

The aft vanity/closet overhead indirect light assembly is along the top of the closet cabinet.

Electrical power for the light starts at the closet terminal board through a circuit breaker which protects the overhead indirect light electrical circuit.

The aft vanity/closet overhead light is controlled by a dimming control switch on the vanity switch panel through a power supply unit.

The aft closet indirect light is controlled by a circuit breaker and door actuated switch. Electrical power for the closet light originates at the closet terminal board through a circuit breaker (CB1F, CLOSET LIGHTS) which protects the closet light electrical circuit.

For an electrical diagram of the vanity/closet overhead indirect lighting light circuitry, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

NOTES

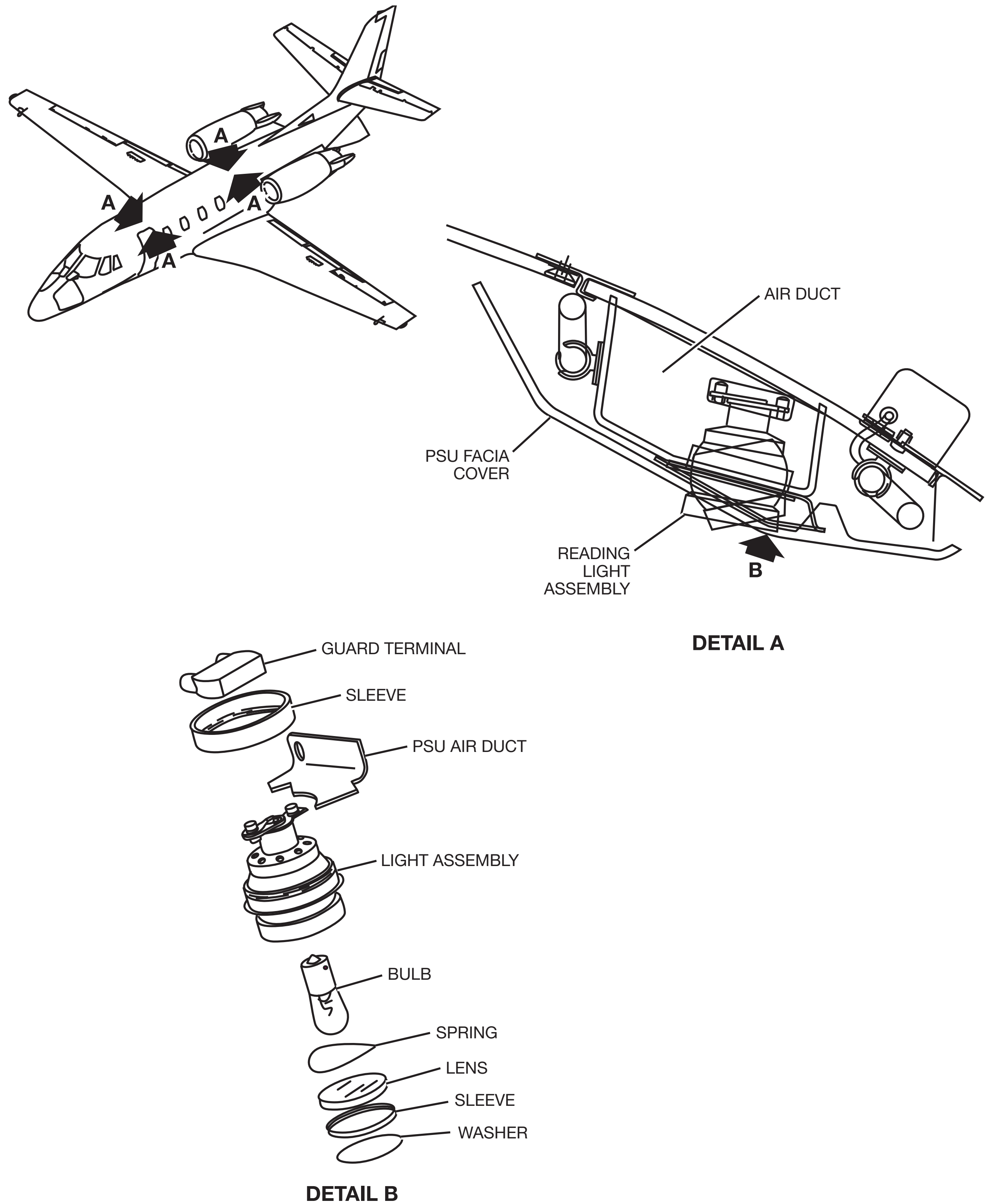


Figure 33-11. Cabin Reading Lights

CABIN READING LIGHTS

Description

The cabin reading lights are identical in construction and size. Each cabin reading light is individually controlled by a switch next to the light. The light beam can be controlled directionally (gimbal type mounting). Removal and installation procedures of these lights is typical (Figure 33-11).

The overhead reading lights are operated individually by switches any time 28 VDC is available.

Overhead reading lights in the passenger compartment, used as emergency lights, are controlled by a switch on the pilot switch panel or the 5 "G" switch in the emergency battery storage case. Refer to "Emergency Lighting" in the *AMM*.

NOTES

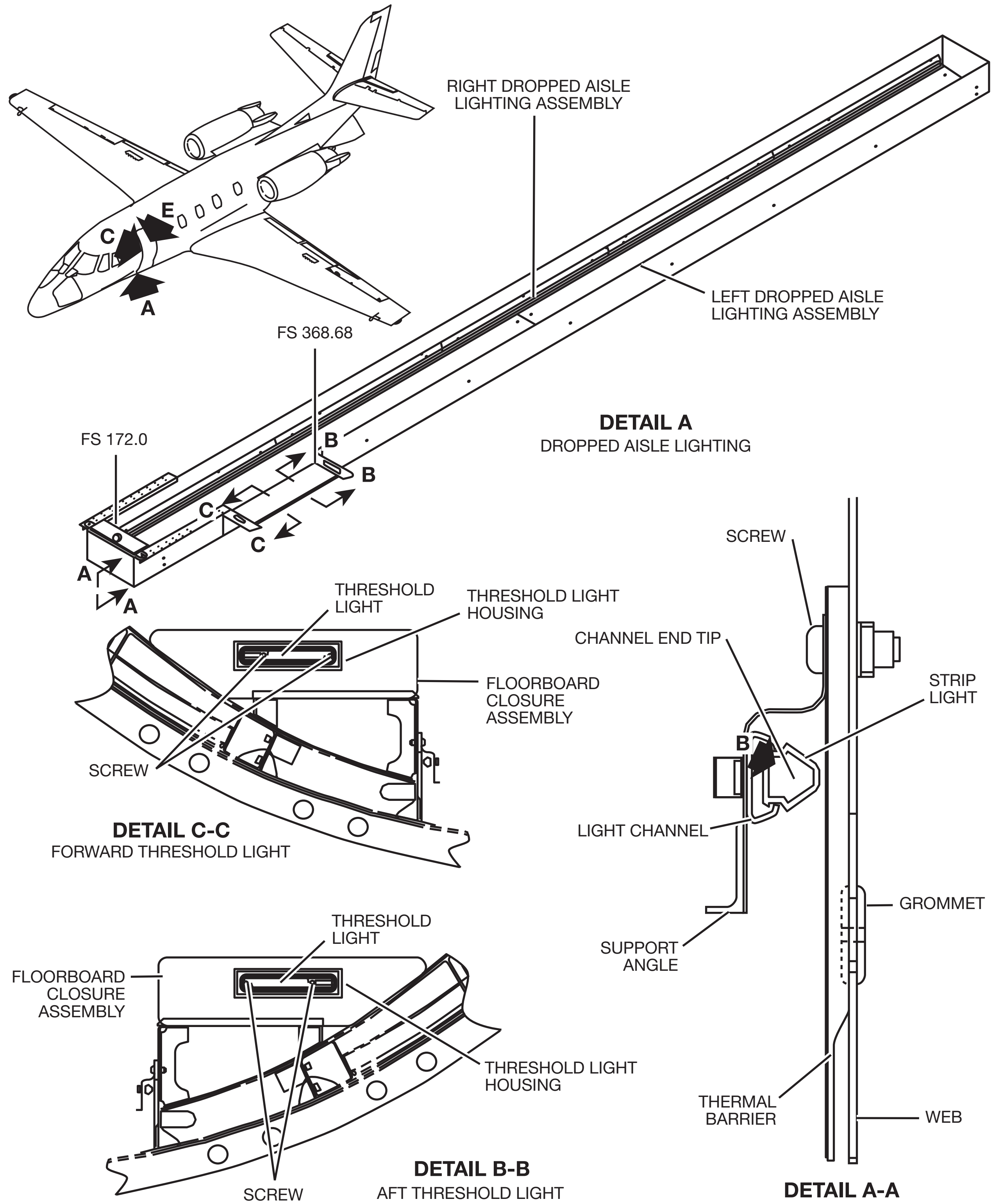


Figure 33-12. Cabin Dropped Aisle Light and Threshold Light

CABIN DROPPED AISLE AND THRESHOLD LIGHT

Description

Dropped aisle lights are strategically in the cabin footwell area (Figure 33-12). Five strip-light assemblies illuminate the dropped aisle (foot well) area; and five cabin door threshold lights illuminate the cabin door stair risers and cabin entry door threshold area. All are controlled by the footwell light switch. The FOOTWELL circuit breaker protects the system.

Cabin entry lighting consists of:

- Three threshold lights in the cabin entry door stair-risers
- Two threshold lights on the left and right sides of the cabin door threshold area

The cabin entry lighting is protected by the AFT/FWD COMP LTS circuit breaker in the aft power junction box.

The control switch for the dropped aisle lights is on the forward trim of the cabin entrance door frame. The cabin entry door post switch panel light is a keypad-type switch.

There are two strip lights on the left and right side of the dropped aisle between FS 172.00 and FS 368.68 aft vanity area.

When the cabin door is open, selected overhead, dropped-aisle and threshold entry door and step lights are automatically lighted when the entrance light switch is turned ON. An internal lamp in the switch automatically lights when the door opens to aid in locating the switch.

Operation

Unlocking the main entrance door actuates the cabin entry door lock signal. Actuation of the cabin entry door signal completes an electrical circuit to illuminate a light in the cabin entry door post switch panel when the PASS SAFETY switch is positioned to OFF only.

Positioning the PASS SAFETY switch to ON extinguishes the light in the cabin entry door post switch panel and illuminates the dropped aisle lights.

The cabin entry door post switch panel internal light is also lit whenever the PASS SAFETY switch on the tilt panel is ON.

CABIN ENTRY LIGHTS

The cabin entry lights are operated by the cabin entry door post switch, any time the AFT/FWD COMP LTS circuit breaker is engaged. The cabin entry door post switch light is internally lighted when the cabin door is open and the switch is OFF. The circuit, which lights the internal light, is completed through the upper forward cabin entry door post switch, when the door is open.

When performing maintenance on the threshold lights or dropped-aisle lights that are a part of the cabin indirect lighting system, ensure that all affected circuit breakers are disengaged. Threshold lights are powered by both FOOTWELL LIGHTS circuit breakers (CB135V and CB136V) in the interior power junction box; and by the AFT/FWD COMP LTS circuit breaker in the aft power junction box.

The cabin reading lights (part of the cabin indirect lighting system) are powered by the left or right READING LIGHTING circuit breakers (CB137V and CB138V) in the interior power junction box and by the emergency lighting battery packs. Ensure that left and right READING LIGHTING circuit breakers (CB137V and CB138V) in the interior power junction box are disengaged and that the PASS SAFETY switch (SI032) is OFF.

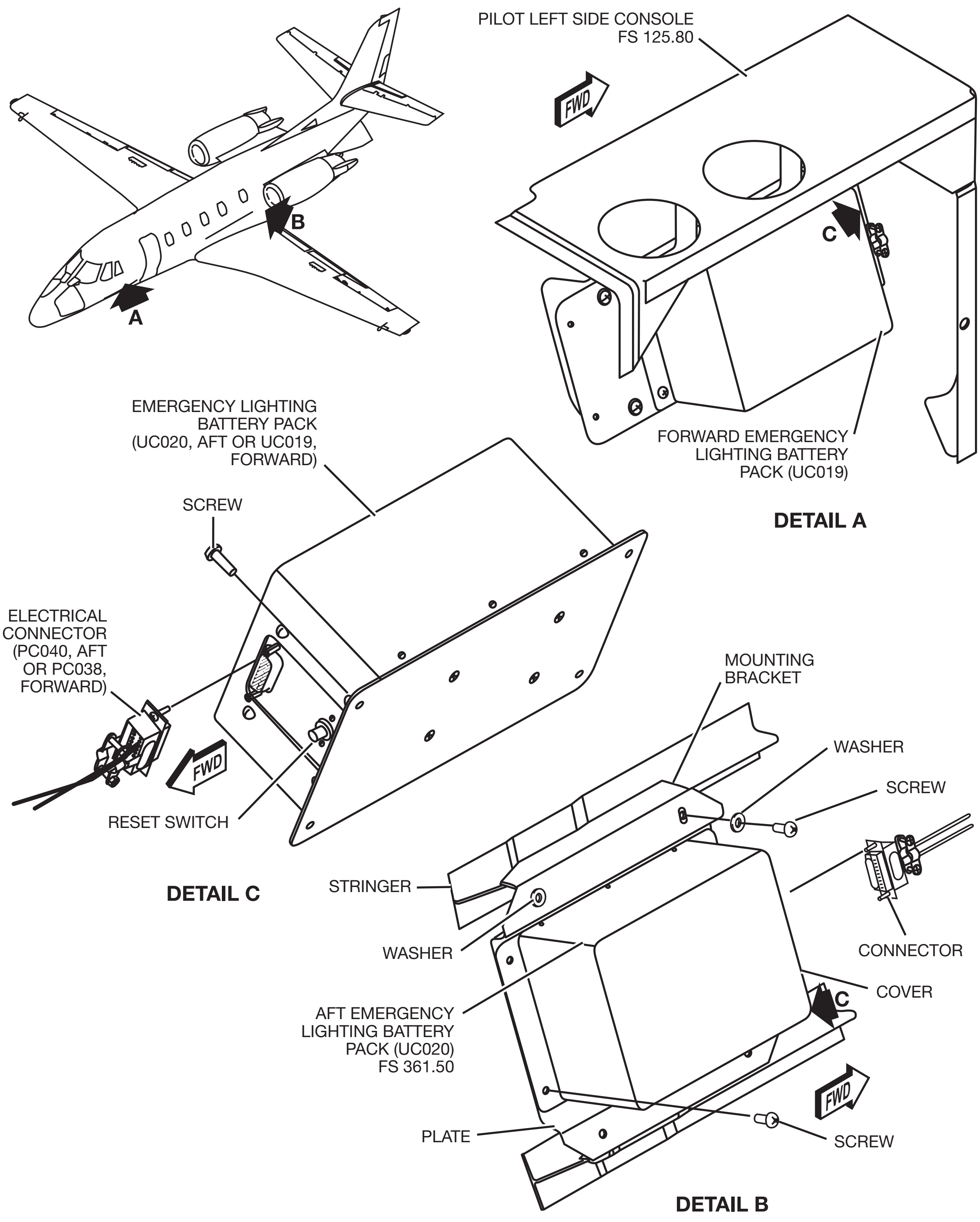


Figure 33-13. Emergency Lighting Battery Pack

EMERGENCY LIGHTING

DESCRIPTION

This section describes the components and subsystems, which provide lighting for emergency situations. These systems may be powered by the main battery, or in case of loss of main battery power, from the emergency lighting battery packs.

The emergency lighting system provides the following for an emergency evacuation during night operations or during reduced visibility conditions:

- General cabin illumination
- Emergency exit illumination and identification
- Evacuation path illumination
- Ground illumination

The emergency system consists of two nickel-cadmium battery packs with associated charging and control circuitry: one in the copilot left side console and one in the left aft cabin area.

Emergency interior lighting is provided at:

- Selected passenger reading lights
- Exit identification signs
- The engine instruments

The exit signs incorporate floodlights to illuminate the first step area of the main entrance and the escape door area.

Emergency exterior lighting consists of an emergency light in the right wing root fairing, and two additional over-the-wing emergency lights are in the right wing-to-fuselage fairing (to aid in exiting the airplane through the escape hatch) (Figure 33-13).

OPERATION

The emergency lighting system is operated by a three position EMER LTG switch (SI059) on the pilot lighting subpanel. The EMER LTG switch ON position provides power to all emergency lights from the airplane battery/generator system when available or from the emergency lighting battery packs (UC019 forward and UC020 aft).

In case the normal system fails, the EMER LTG switch ARM position provides automatic activation of all emergency lights. This occurs during loss of normal electrical power or when the airplane experiences a five gravity (G) force. The EMER LTG switch OFF position disables the emergency lighting system.

Adjacent to the EMER LTG switch (SI059) is an amber light which illuminates when airplane power is on and the emergency lighting system is not armed.

The emergency exit lighting battery packs are charged by a precision current limiter-charging circuit, any time that the main airplane power is on and the emergency lighting circuit breakers are engaged: EMER LIGHTS–FWD (HT043) and EMER LIGHTS–AFT(HT048).

The forward emergency battery pack (UC019) illuminates the following lights:

- Main cabin entrance door emergency exit sign
- Cabin reading lights
- Right exterior emergency exit light
- Left engine instrument light (during engine start only)
- Left dropped-aisle lighting

The aft emergency battery pack (UC020) illuminates the following lights:

- Emergency escape door emergency exit sign
- Exterior over-the-wing escape route light (both lights)
- Cabin right reading light
- Left exterior emergency light

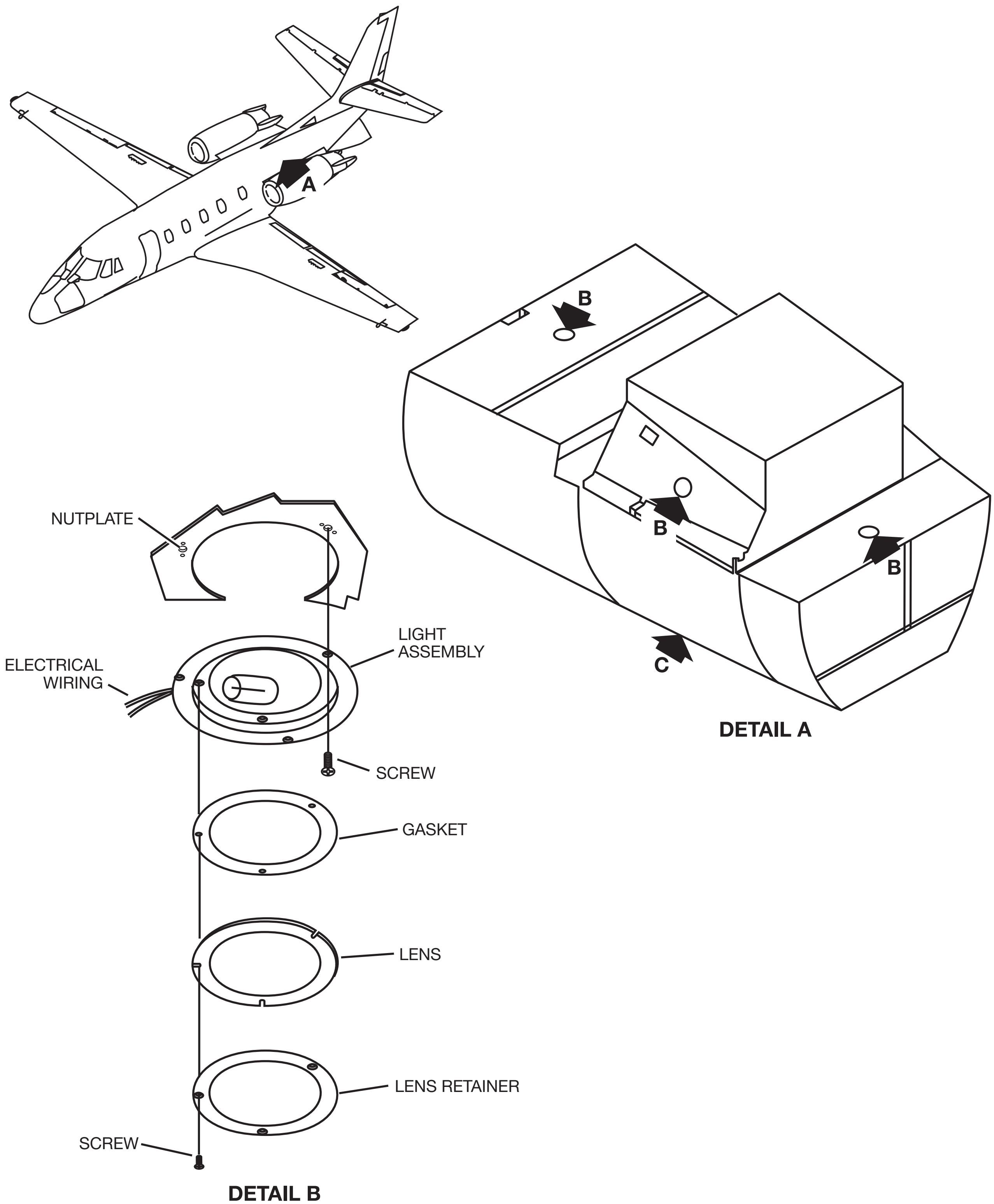


Figure 33-14. Tailcone Baggage Compartment Light Assembly

- Right dropped aisle lighting

BAGGAGE AND MAINTENANCE COMPARTMENT LIGHTING

Baggage and maintenance compartment lighting includes the tail cone baggage compartment and the tail cone maintenance lights. These units provide local lighting for their respective compartment. All lights are 28 volts direct current (VDC).

TAIL CONE MAINTENANCE COMPARTMENT LIGHT

The tail cone maintenance compartment light assembly is overhead and is controlled by two switches. The microswitch (ST008) is on the right side of the airplane in the tail cone access door frame at FS 512.80. The microswitch in the door bracket has to be activated (opening the door activates the switch) before the manual ON/OFF switch (ST007) at FS 479.50 illuminates the light (FT004). The combination of switches ensures that the tail cone maintenance light does not remain on after the door is closed even if the manual switch has been left in the ON position.

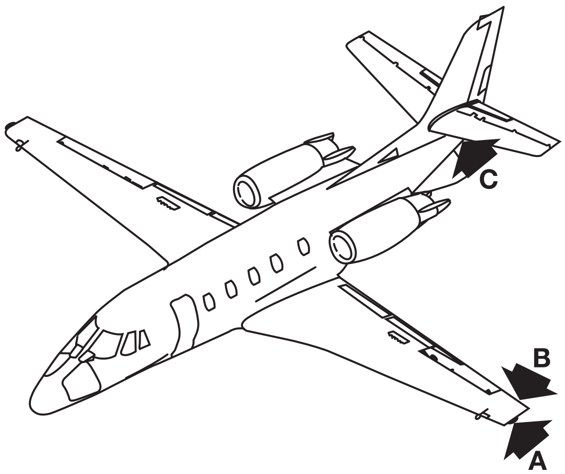
The system circuit is interrupted at each switch; therefore, the door must be open before the manual ON/OFF switch supplies power to the light. Turning off the light is accomplished when either or both switches open the circuit.

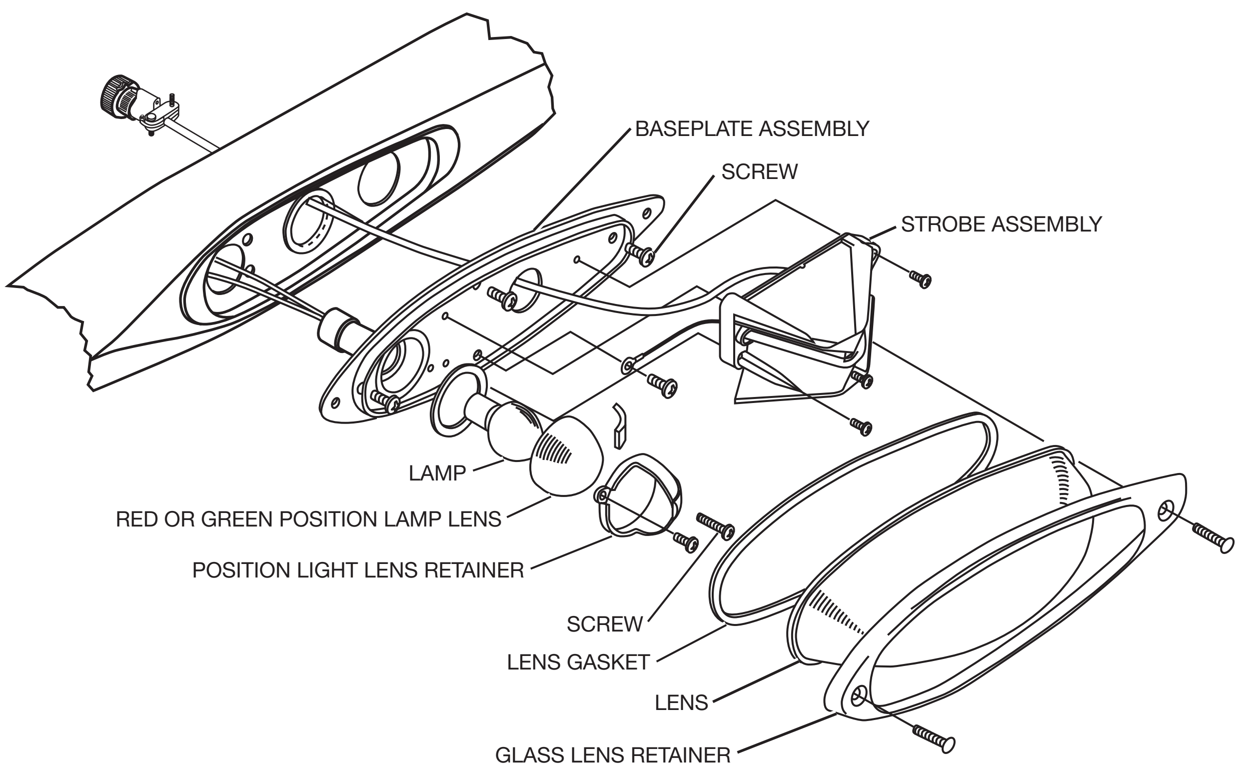
TAIL CONE BAGGAGE COMPARTMENT LIGHTING

The tail cone baggage compartment lighting consists of a door microswitch (ST011) and a manual ON/OFF switch that controls three compartment lights (FTO1O, FT011 and FT012). The microswitch (ST011) is on the left side of the airplane in the baggage compartment access door frame (between FS 421.50 and FS 424.50) (Figure 33-14). The manual ON/OFF switch (ST003) is inside the baggage compartment door closeout at FS 426.00. The microswitch is activated by opening the baggage compartment door before the manual ON/OFF switch can connect power to the light. The door microswitch closes the circuit when the door is open, but the circuit is not complete and the light does not come ON until the manual ON/OFF switch is in the ON position. If the manual switch is left ON and the door is closed, the microswitch opens the circuit extinguishing the light.

This arrangement ensures that the lights go out when the baggage compartment door is shut even if the manual ON/OFF switch is left in the ON position.

NOTES





DETAIL A

Figure 33-15. Navigation Lights (XL/XLS)

EXTERIOR LIGHTING

DESCRIPTION

Exterior lighting is subdivided into:

- Navigation lights
- Ground recognition lights
- Anticollision lights
- Wing inspection lights
- Landing lights
- Belly fairing lights
- Taxi lights
- Tail floodlights

For electrical schematics of the exterior lighting systems, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

COMPONENTS

Navigation Lights

XL/XLS

Navigation lights consist of a colored light at each wingtip and a clear light on the aft stinger. The left navigation light must be red, the right green, and the tail white. The navigation lights and anticollision lights are on the same wingtip base assembly (Figure 33-15).

Anticollision Lights

Anticollision lights are on the extreme outboard end of each wingtip. Ground recognition lights are on top of the rudder. The GND REC/ANTI-COLL, light switch (SI014) has three positions:

- OFF
- GND REC ON—Switch controls the beacon light on top of the rudder
- GND REC/ANTI-COLL ON—Switch controls both the wing anticollision lights and the beacon light

Landing Lights

The recognition lights consist of wingtip recognition lights aimed directly forward, and belly fairing lights, on the underside of the tail cone. Separate LANDING LIGHTS switches (SC054 right and SC055 left) control both the operation of the belly fairing and wingtip recognition lights.

Wing Inspection Lights

The wing inspection lights illuminate to enable the pilot/copilot to inspect for icing conditions on the wing leading edges. The wing inspection lights are controlled by the WING INSP light switch (SI016) on the ANTI-ICE/DEICE panel. The wing inspection lights are on the fuselage fairings, adjacent to the wing inboard leading edge.

Wing Landing Lights and Belly Multifunctional Lights

The airplane is equipped with wing landing lights and belly multifunctional lights. The belly multifunctional lights are in the belly fuselage fairing (between FS 240.35 and FS 253.20). The wing landing lights are in the left and right wings at WS 302.93. The landing lights are controlled by LANDING LIGHTS L/R switches (SC055 left and SC054 right) on the switch pedestal. Each set of lights is individually controlled to provide redundancy.

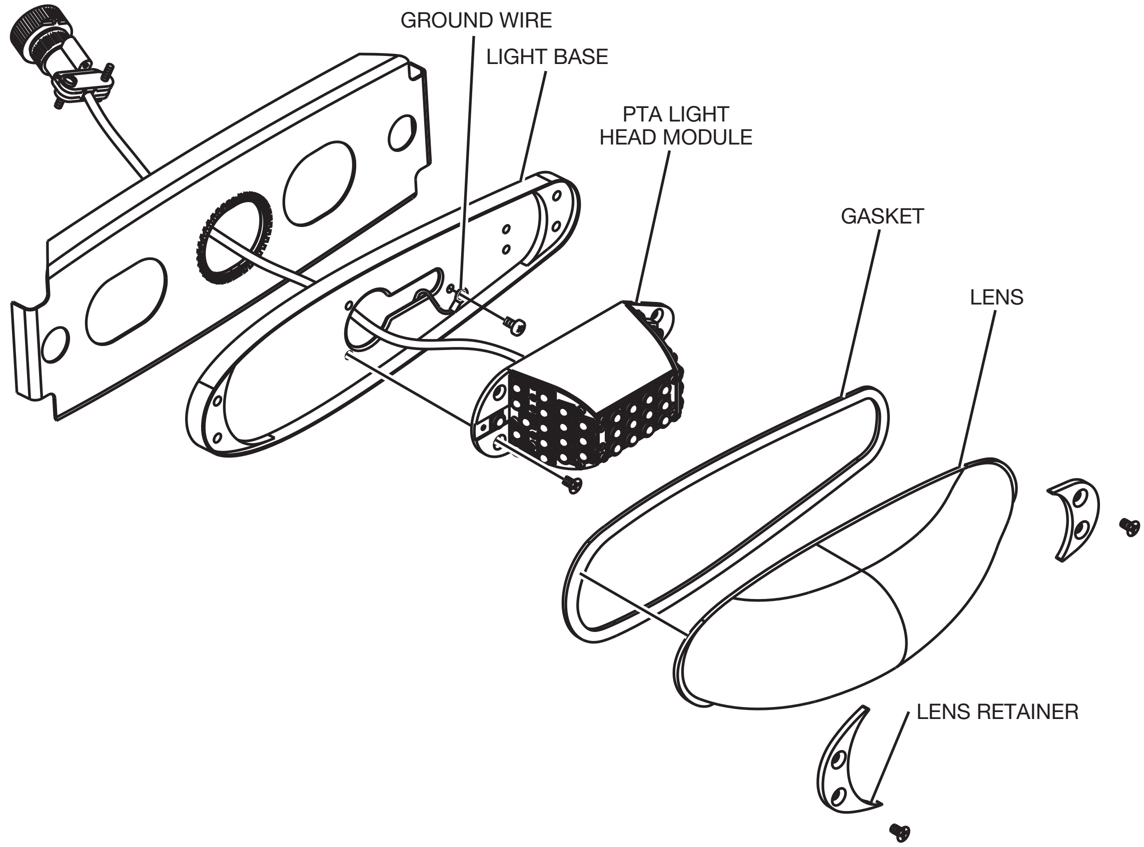
Taxi Lights

The taxi lights are the same as the recognition lights and are controlled by the same switches. The taxi lights are on the wingtips. They allow the pilot/copilot to visually detect the location of the wingtips during taxi operation.

Tail Floodlights

The tail floodlights are also known as:

- The identification lights
- Logo lights
- Tail lights



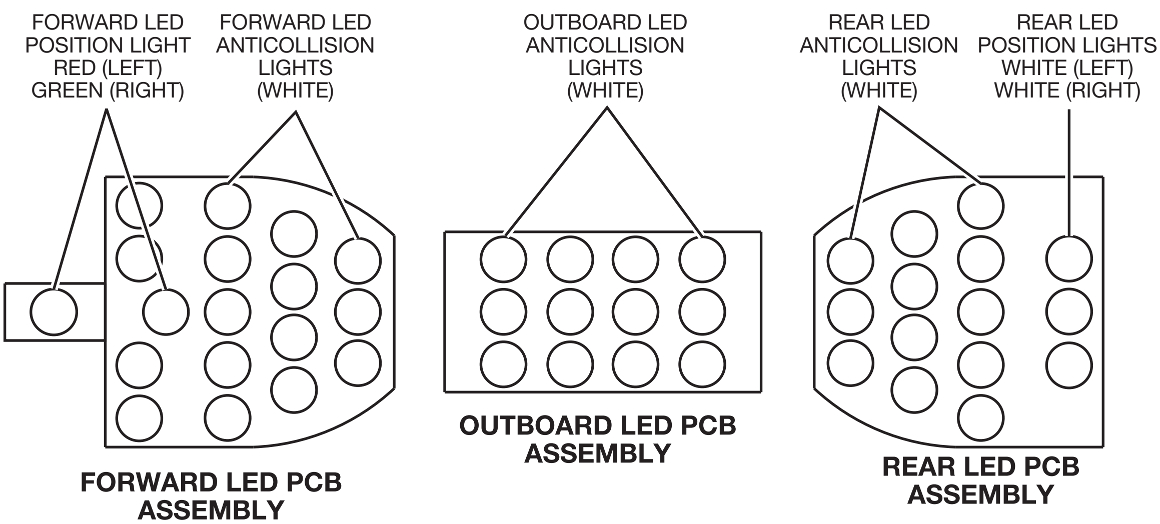


Figure 33-16. Navigation Lights (XLS+)

The floodlights are fixed position lights on the left and right horizontal stabilizers. The floodlights are controlled by the TAIL FLOOD ON/OFF switch (SI013) on the tilt switch panel.

NAVIGATION LIGHTS

Description

There is a forward navigation light on each wingtip. There is an additional navigation light in the tail cone stinger. The navigation lights in the left and right wingtips utilize the same light base assembly (see Figure 33-15 for XL/XLS and Figure 33-16 for XLS+).

For additional wiring information, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

NOTES

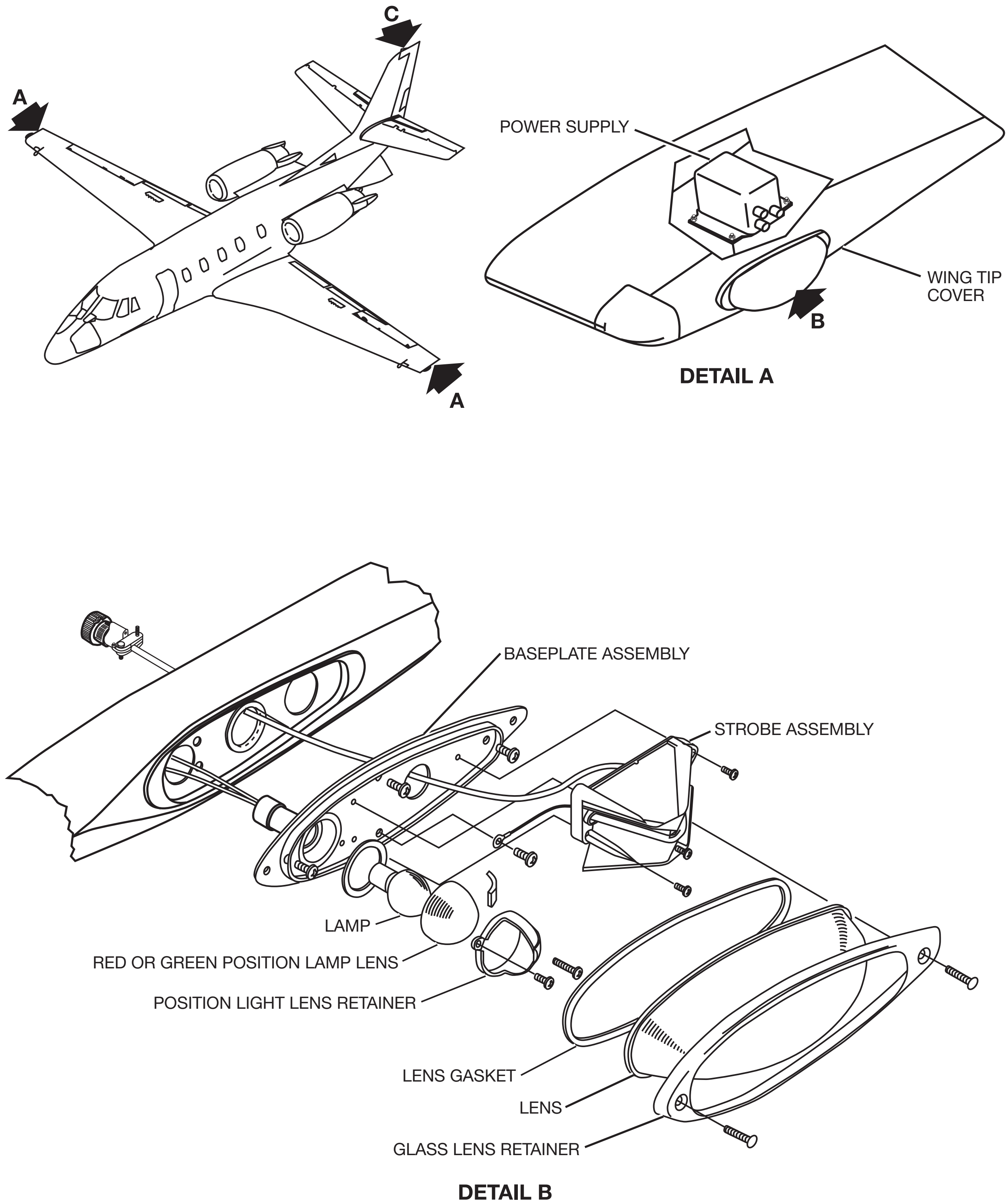


Figure 33-17. Anticollision/Ground Recognition Lights (XL/XLS)

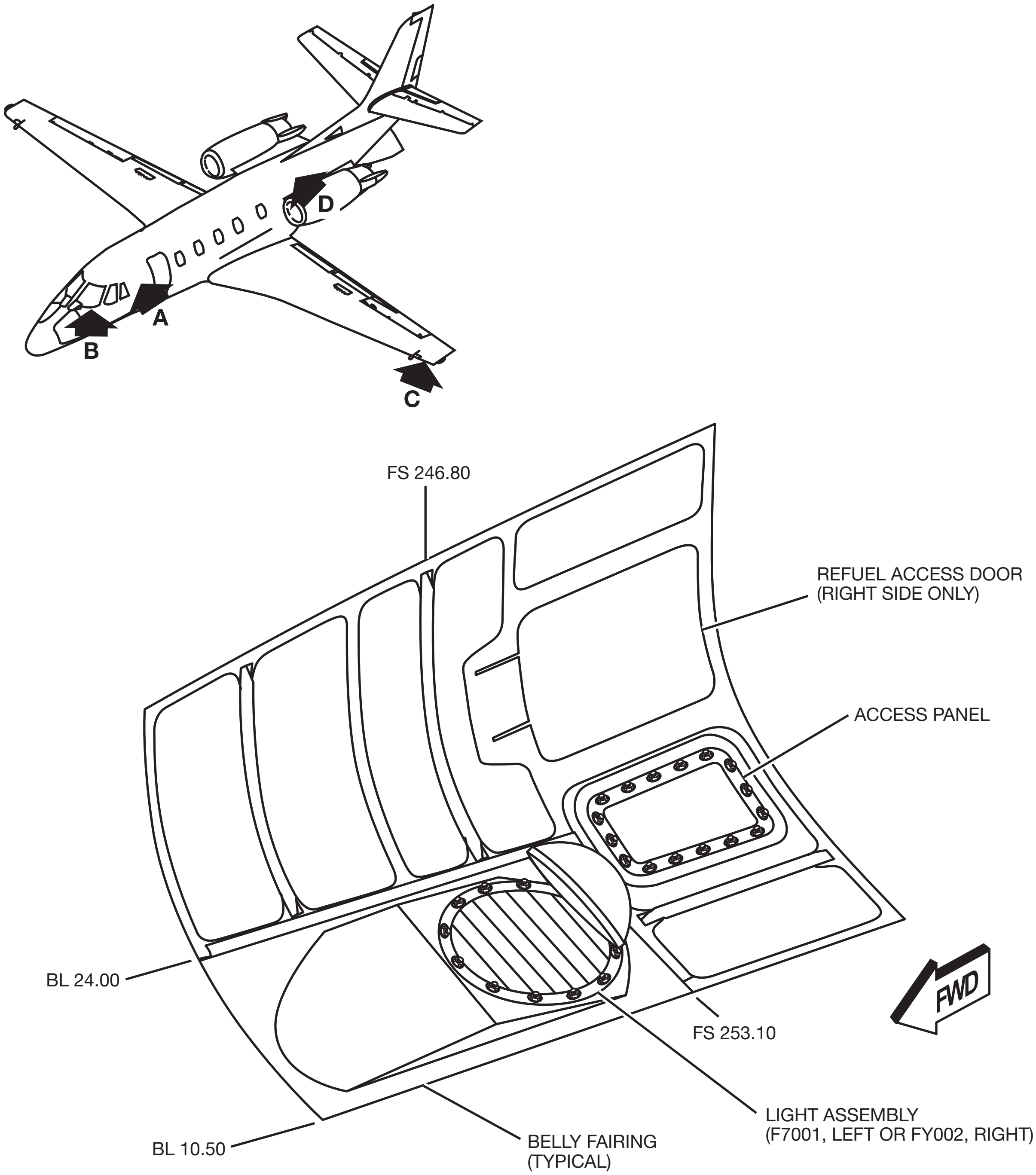
ANTICOLLISION/GROUND RECOGNITION LIGHTS

Description

The anticollision lights are on the outboard ends of the left and right wingtips (Figure 33-17). Anticollision lights are powered by an integral power supply in each wingtip assembly. The GND REC/ANTI-COLL switch (SI014) controls the anticollision and ground recognition lights. Positioning the switch to the GND REC ON position makes the ground recognition upper red beacon light (FV003) illuminate. When the GND REC/ANTI-COLL switch is in (SI014) the ON position, the ground recognition upper red beacon light (FV003) and the left and right anticollision come on.

For additional wiring information, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

NOTES



DETAIL A

Figure 33-18. Belly Fairing and Landing Lights

LANDING/REC/TAXI LIGHTS

Description

The airplane is equipped with two sets of multipurpose lights. The fuselage fairing lights are in the belly fuselage fairing (between FS 240.35 and FS 253.20).

The wingtip landing/taxi lights are in the left and right wingtips (between WS 303.20 and WS 315.97). The wingtip landing lights serve as landing lights during landing and taxi lights after landing.

The belly fuselage fairing lights function as landing/rec/taxi lights. The belly fuselage fairing lights are fixed-position, seal-beam lights controlled by two, three-position LANDING LIGHTS switches (SC055 left and SC054 right) on the switch panel (Figure 33-18). Removal and installation of the left and right belly fuselage fairing lights and left and right wingtip landing/taxi lights are typical.

For additional wiring information, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

NOTES

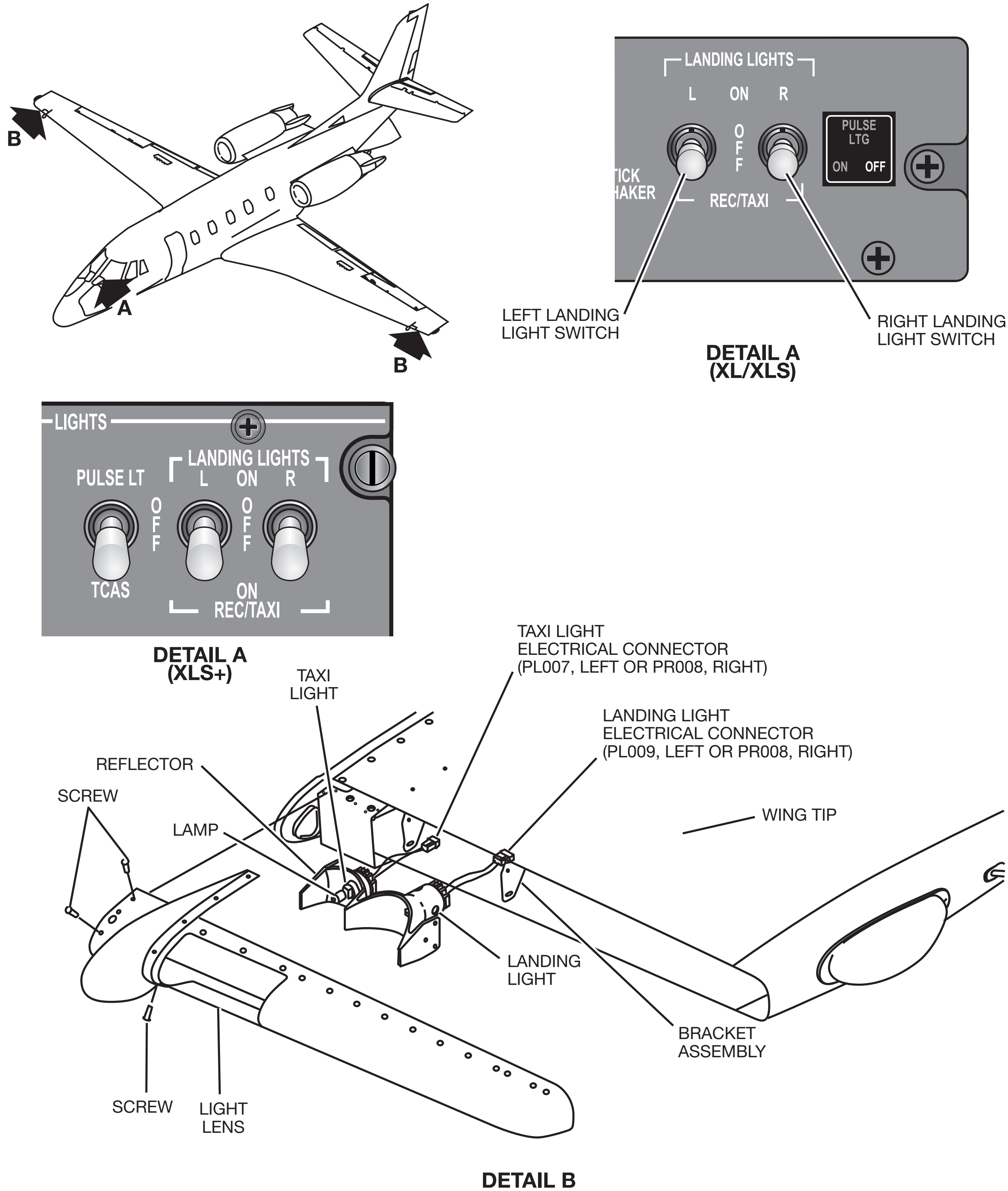


Figure 33-19. Recognition Lights

WING RECOGNITION LIGHTS

Description

There is a wing recognition light in each wingtip (between WS 303.02 and WS 315.97) (Figure 33-19). The wingtip recognition lights are controlled by the LANDING LIGHTS control switches (SC055 left and SC055 right) and four recognition light control relays (KZ039 left and KZ038 right).

For additional electrical wiring information of the systems, refer to *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

NOTES

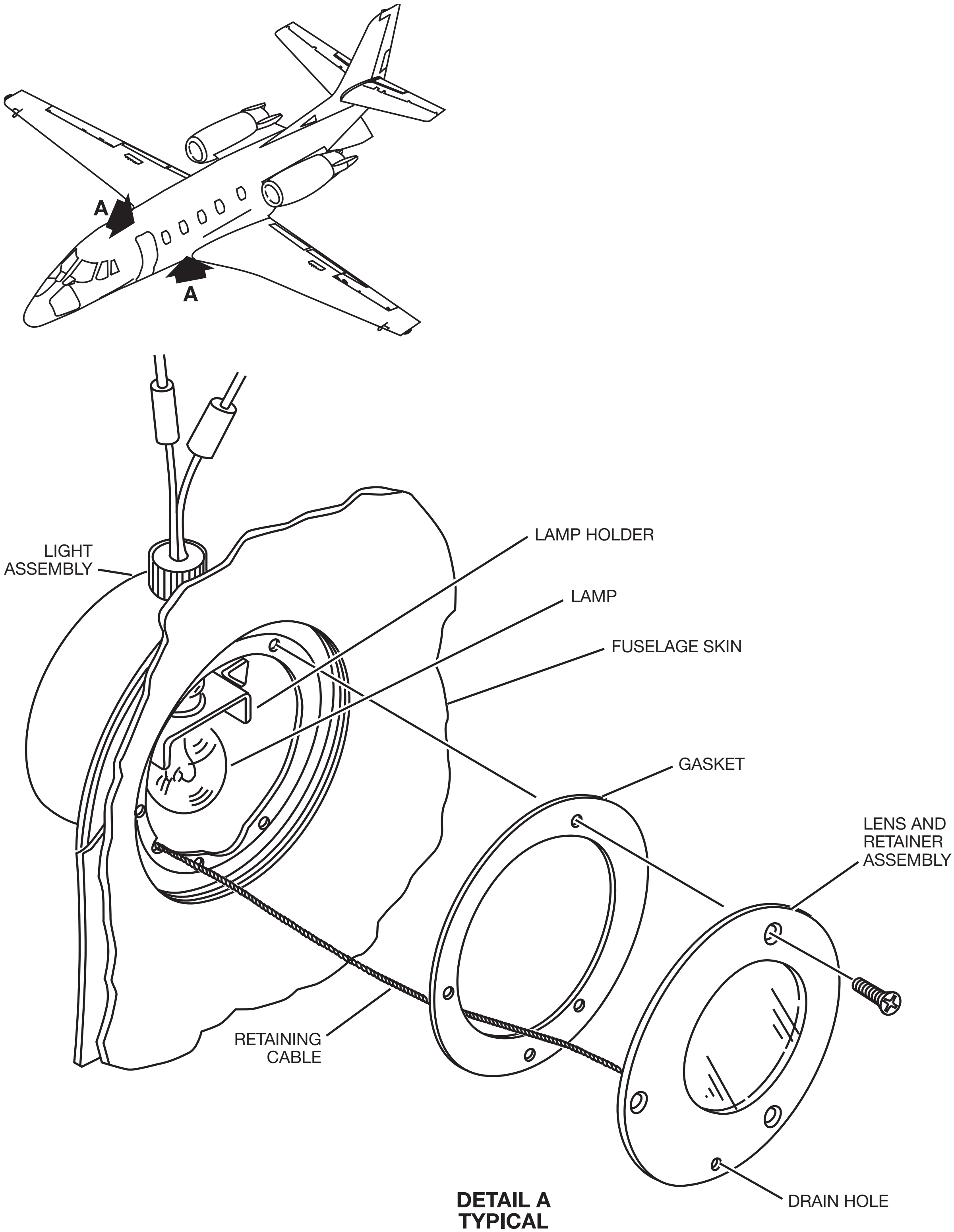


Figure 33-20. Wing Inspection Lights

WING INSPECTION LIGHTS

Description

The fixed position wing inspection lights are in the left and right sides of the fuselage. The wing inspection lights are forward of the left and right wing leading edges (between FS 253.20 and FS 258.00 and WL 119.95). The wing inspection lights are used to visually check the left and right wing leading edge for icing conditions. The inspection lights are controlled by the WING INSP control switch (SI016) on the pilot switch panel (Figure 33-20).

For additional electrical wiring information of the systems, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

The bulb in the wing inspection light can be replaced by removing the lens. A restraining cable prevents damage from accidental dropping of the lens and retainer assembly. Removal procedures do not include removal of the light housing. If a malfunction occurs in the lamp holder, replace the holder by removing two screws.

NOTES

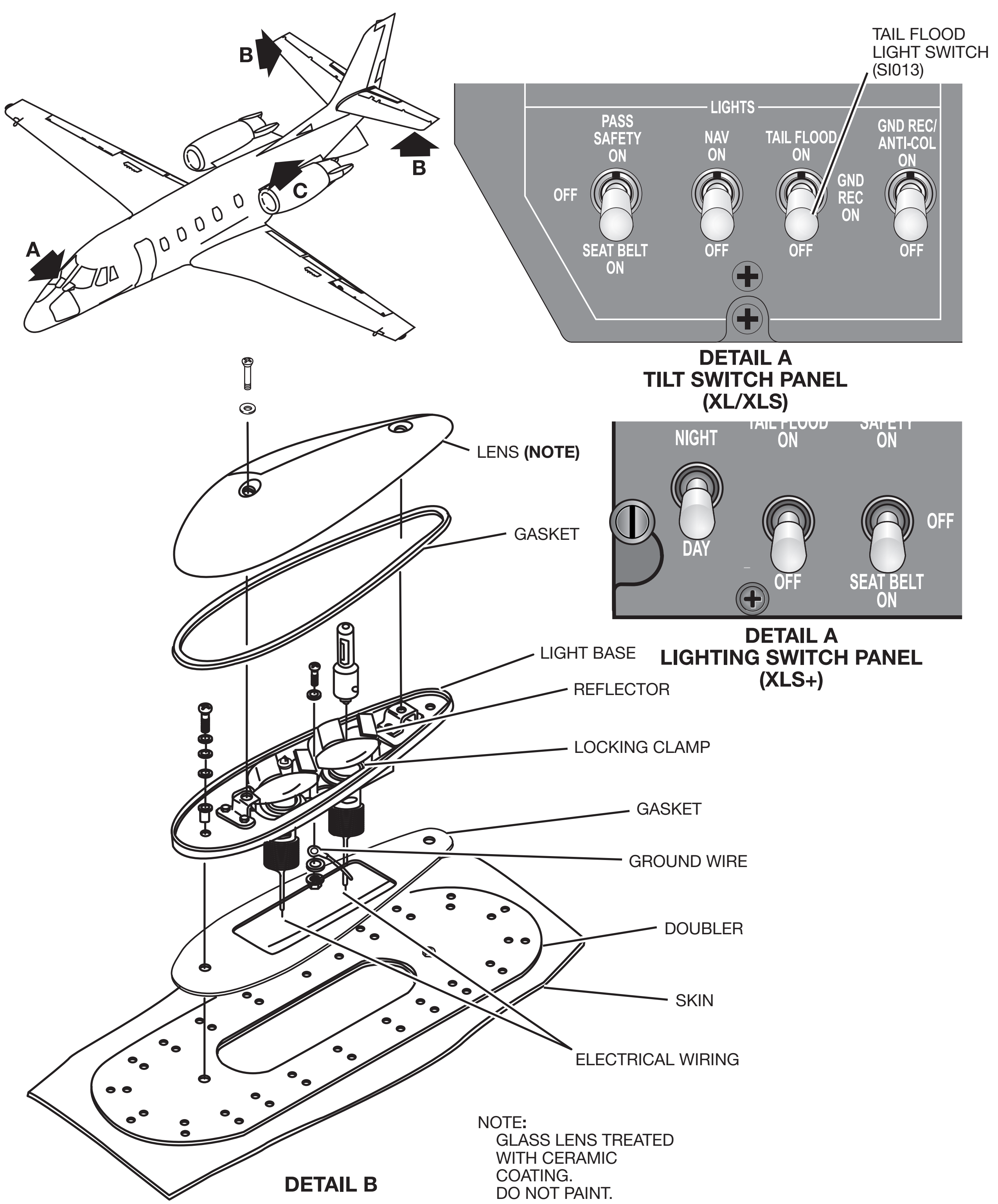


Figure 33-21. Tail Floodlight Assembly

TAIL FLOODLIGHTS

Description

The tail floodlights (Figure 33-18) are also known as:

- Identification lights
- Logo lights
- Tail lights

The floodlights are fixed-position lights on the left and right horizontal stabilizers.The floodlights are used primarily as recognition lights.

For additional electrical wiring information of the tail floodlights, refer to the *Citation XL/XLS/XLS+ Wiring Diagram Manual*.

Major components of the tail floodlight system are:

- A TAIL FLOOD ON/OFF control switch (SI013)
- A circuit breaker HT020 (TAIL FLOOD LTS)
- A tail flood light control relay (KZ012)
- Two floodlight assemblies (FH001 left and FH002 right)

The tail floodlights (FH001 left and FH002 right) are controlled by the TAIL FLOOD ON/OFF control switch (SI013) on tilt switch panel.

A control relay (KZ012) is utilized to prevent excessive use of heavy gage wire. The current required to illuminate the lamps is routed through the relay contacts. The control relay is in the right power junction box.

The tail floodlight assemblies (FH001 left and FH002 right) illuminate the company logo on the vertical stabilizer. There is an assembly on the top side of the horizontal stabilizer, on each side of the vertical stabilizer. Each tail floodlight assembly is comprised of:

- A reflector
- Base assembly
- Two lamps
- A housing

NOTES

QUESTIONS

1. Position the PANEL LIGHT control master switch to NIGHT:
   A. Activates the control rheostats
   B. Dims the annunciator panel lights
   C. Illuminates the STARTER DISENGAGE button
   D. All of the above

2. Emergency cabin lighting is powered from:
   A. Main aircraft battery
   B. Two emergency battery packs
   C. Emergency DC power
   D. Either A or B

3. Emergency lighting is activated by:
   A. Floodlight switch
   B. Emergency light switch ON
   C. Loss of main DC power or G force (switch ARM)
   D. Either B and C

4. Landing lights consist of:
   A. Belly lights only
   B. Belly lights and recognition light
   C. Both wingtip lights on each wingtip.
   D. Outboard wingtip lights

5. Which lights remain working when the battery switch is placed to EMER (generators offline)?
   A. Floods and auxiliary panel lights
   B. Flood and map lights
   C. EL and auxiliary panel lights
   D. Left, right, and center panel lights

6. Concerning the emergency light switch on the instrument panel:
   A. It must be placed in the ARM position prior to takeoff
   B. It must be placed in the ON position prior to takeoff
   C. Amber indicator light near the switch indicates maintenance must be performed on the emergency battery packs
   D. Crossfeed bus powers the system

7. While loading baggage after dark, you forget to turn off the baggage compartment lights. These lights:
   A. Stay on after you close the door
   B. Extinguish after one hour
   C. Extinguish when the main cabin door is locked
   D. Extinguish when you close the baggage door

CHAPTER 34
NAVIGATION

CONTENTS

ILLUSTRATIONS

CHAPTER 34
NAVIGATION

INTRODUCTION

The navigation section describes units and components which provide aircraft navigational information. This includes pitot static, flight director, VOR and other navigational systems and indicators. Each Citation XL/XLS/XLS+ aircraft is delivered with complete *Avionics Wiring Diagrams* specifically prepared for that serial number aircraft.

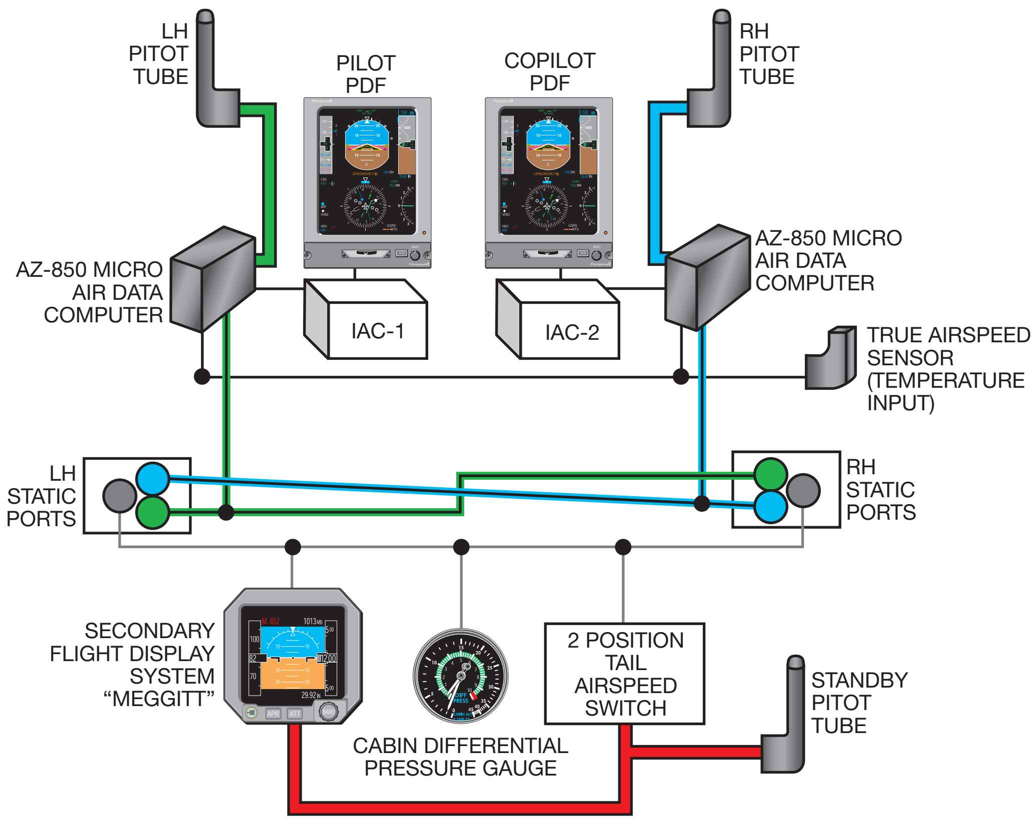


Figure 34-1. Citation XL Pitot-Static Block Diagram

GENERAL

Navigational information for the Citation XL/XLS is provided by pitot-static systems, Honeywell micro air data computer (MADC), radio altimeters, total air temperature (TOT) probes, angle of attack (AOA) system and magnetic compass. Electronic flight information system (EFIS), attitude heading reference system (AHRS), controllers, localizer, glideslope and marker beacon, radar, ground proximity warning system (GPWS), traffic alert and collision avoidance system (TCAS), global positioning system, and flight management systems (FMS) give the crew navigation information for flights. This information displays on either the primary or multifunction display screens and instruments in the cockpit.

COMPONENTS

FLIGHT ENVIRONMENT DATA SYSTEMS

Description

Flight environment data systems sense environment conditions and use this data to influence navigation. Air data computers, pitot/static, ram air temp, radio altimeter, and AOA are included in flight environment data system.

Pitot/static system supplies air and barometric pressures to operate air data instruments (Figure 34-1). Air data instruments/components utilizing pitot and/or static pressures include:

- Standby altimeter/air speed indicator system
- Delta “P” indicator (cabin differential pressure indicator)
- The two micro air data computers

Micro air data computers (MADCs)—Receive pitot/static pressures and total air temperature inputs for computing standard air data functions. These functions are made available in digital format, for equipment requiring this type of information.

TOT—Uses a sensor to detect outside air temperature.

Radio altimeter—Displays the aircraft absolute altitude below 2500 feet (762 meters) by utilizing a pulse recurrence frequency.

AOA system—Provides a constant readout which enables the pilot to prevent stalls during all flight maneuvers.

Ram air temperature (RAT)—Uses a sensor in the inlet of the right engine that sends raw data to the ARINC 429 Computer, which then transmits information to the RAT display.

NOTES

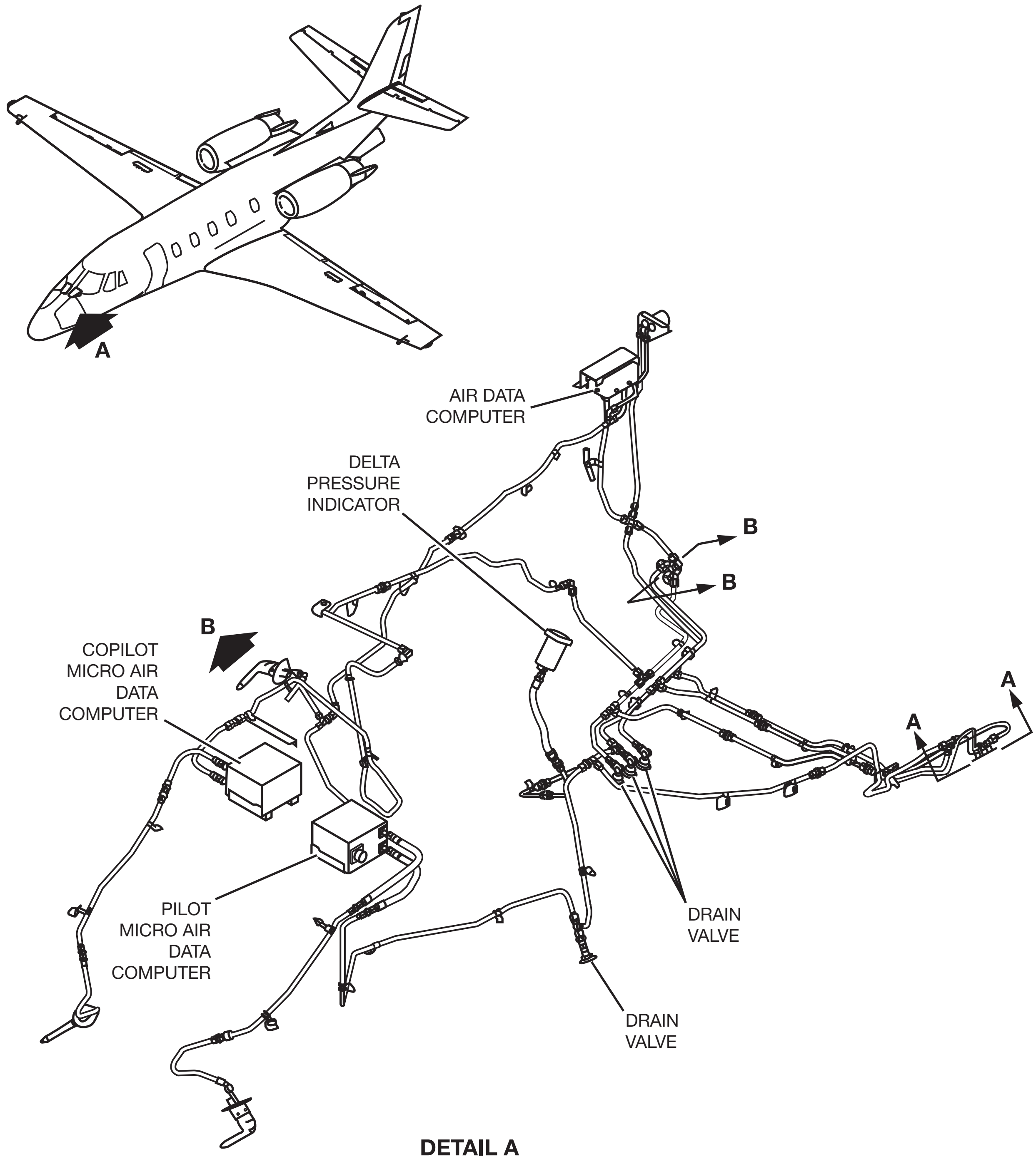


Figure 34-2. Pitot/Static System (XL/XLS)

Components

Pitot/Static Systems

Pitot/static systems have:

- Pitot tubes (heads)
- Static ports
- Static moisture drain valves
- Tubing
- Hoses
- Fittings

The pitot/static lines use clamps to attach the tubing to the structure. There are flex hoses between the tubing and the operating components/instruments.

Pitot System Description

The left (pilot) pitot system supplies pitot pressure to the left MADC (Figure 34-2). The right (copilot) pitot system supplies pitot pressure for the right MADC. The standby pitot system supplies pitot pressure for the standby altimeter/airspeed air-data unit.

Pilot and copilot pitot tubes are symmetrically placed on the nose of aircraft. The standby pitot tube is below the copilot side window. On aircraft 5060 and on, the standby pitot tube is in front of the forward pressure bulkhead on the nose of the aircraft.

Pitot tubes are separated and positioned to give total pressure input and to minimize the possibility of total pitot loss due to a bird strike.

Static System Description

The left (pilot) static system supplies static pressure to left MADC. The right (copilot) static system gives static pressure to the right ADC. The standby static system supplies static pressure to the standby altimeter/airspeed air data unit, and cabin delta P indicator (differential pressure indicator.)

Each static system has two static ports: one is on the left side and one on the right side of the aircraft.

Interconnecting lines connect the left static port with the right static port and to corresponding components/indicators. The pilot static system uses the top static port on the right side and the bottom static port on the left side. The copilot static system uses the bottom static port on the right side; and the top static port on the left side.

The standby static system uses the aft static port on both sides. Static ports are on the lower side of the fuselage below the pilot and copilot side windows.

Moisture accumulation in each pitot system collects at the lowest point in the tube length. The pitot tubes are the lowest point; and moisture automatically drains from the pitot tubes when the aircraft is not in flight. Moisture in the static system collects at five surface moisture drain valves. There are two drain valves on the left and right lower fuselage in front of the forward pressure bulkhead (FS 98.35) and three on the right lower fuselage aft of the forward pressure bulkhead (FS 130.50). All drain valves must be drained at a regular intervals.

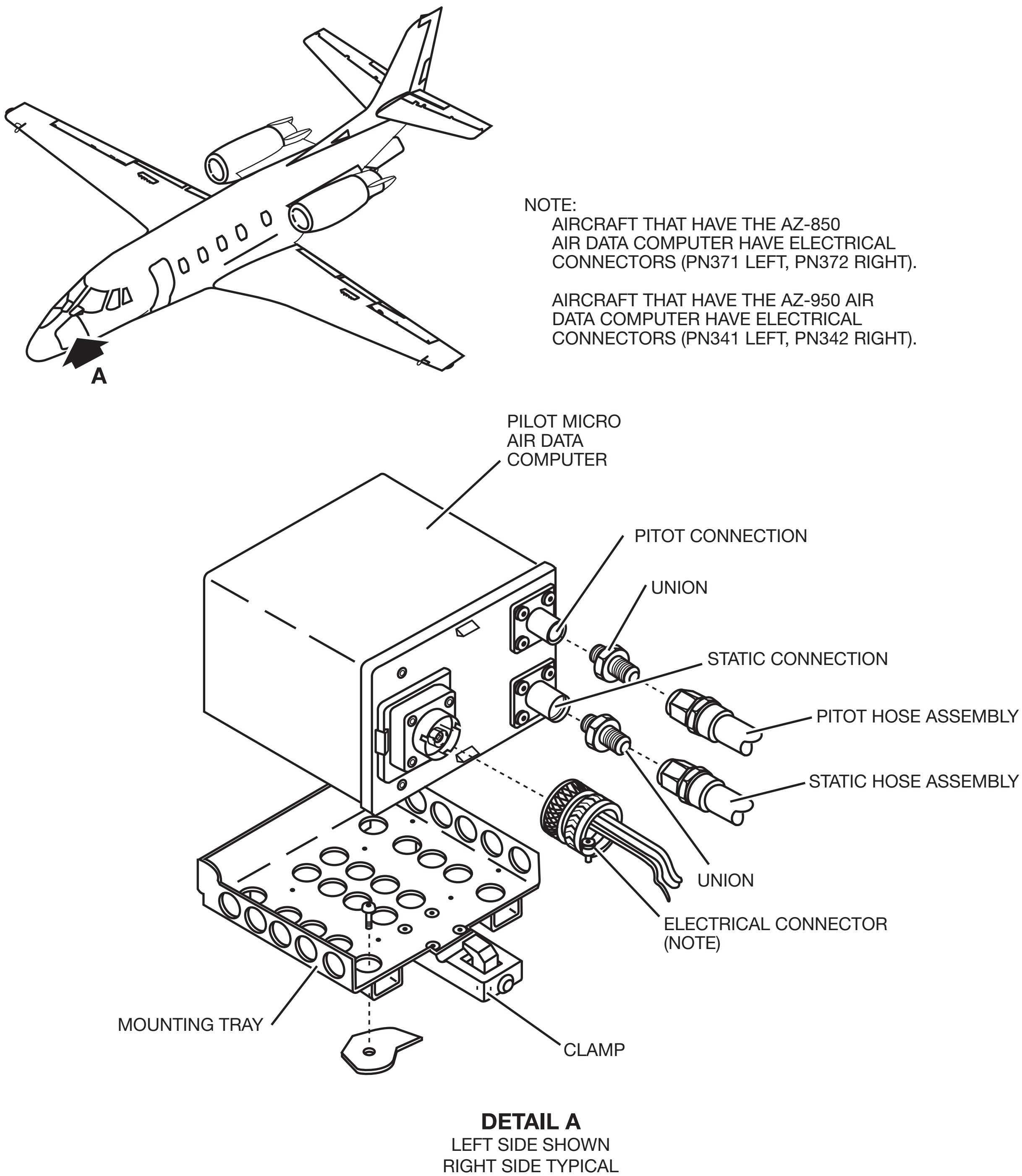


Figure 34-3. Honeywell AZ-850 Micro Air Data Computer

Operation

During flight, air pressure (impact air) collects in the pitot tubes from the forward movement of the aircraft. Pressure in the pitot tubes is transmitted through tubing and hoses to the correct components and instruments. The pitot tubes have heating elements to prevent inlet icing. The pitot heaters are controlled by the anti-ice pitot/static switch on the instrument panel, which receives power from the pitot heater electrical circuit.

Static systems send atmospheric pressure (from ambient air outside the aircraft) through static ports, tubing and hoses to the correct components/instruments. Static ports have heating elements that prevent icing. The pitot/static switch controls static port heaters.

NOTE

A leak test must be done after installation of components on systems that have had components removed.

Honeywell AZ-850 Micro Air Data Computer

The air data system is comprised of two AZ-850 MADCs: pilot (left) and copilot (right) (Figure 34-3). Each MADC provides digital information to components on its respective side through the avionic standard communication bus (ASCB) (Figure 34-3).

The AZ-850 MADC is a microprocessor-based digital computer, which:

- Accepts both digital and analog inputs
- Performs digital computations
- Supplies digital outputs

MADC receives pitot static pressures and total air temperature inputs for computing standard air data functions.

MADC provides outputs for:

- Driving barometric altitude and mach/airspeed displays
- Transponder (mode C and S)
- Flight recorder
- Flight director
- Autopilot
- Overspeed warning

NOTES

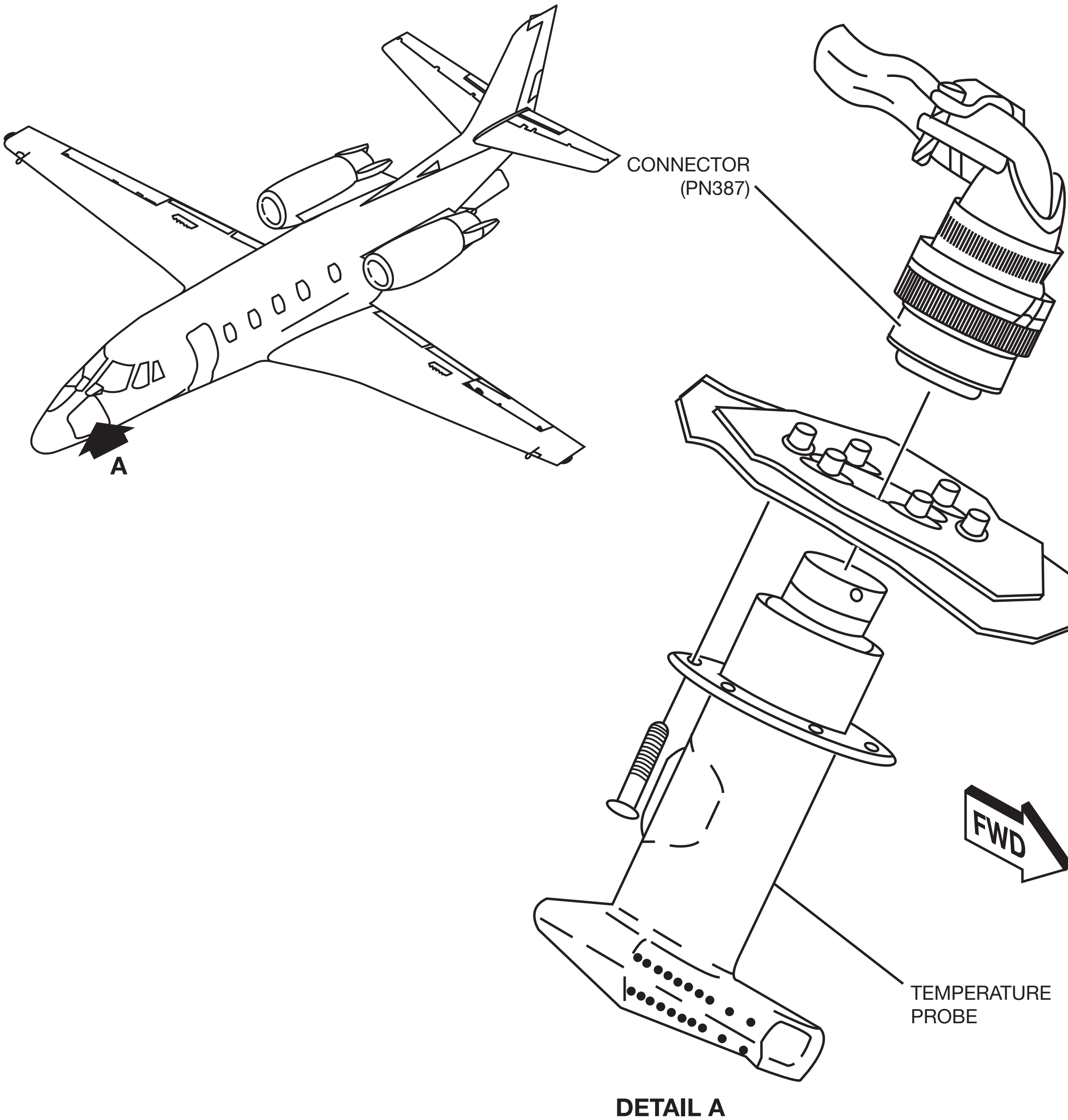


Figure 34-4. Total Air Temperature Probe

Total Air Temperature Probe

The TOT probe is aft and below the right nose bay door (Figure 34-4). The temperature probe provides information to the MADC, which uses this information to provide air data computations. These computations are sent to the integrated avionics computer, which utilizes and displays information on the electronic flight instrument system (EFIS).

NOTES

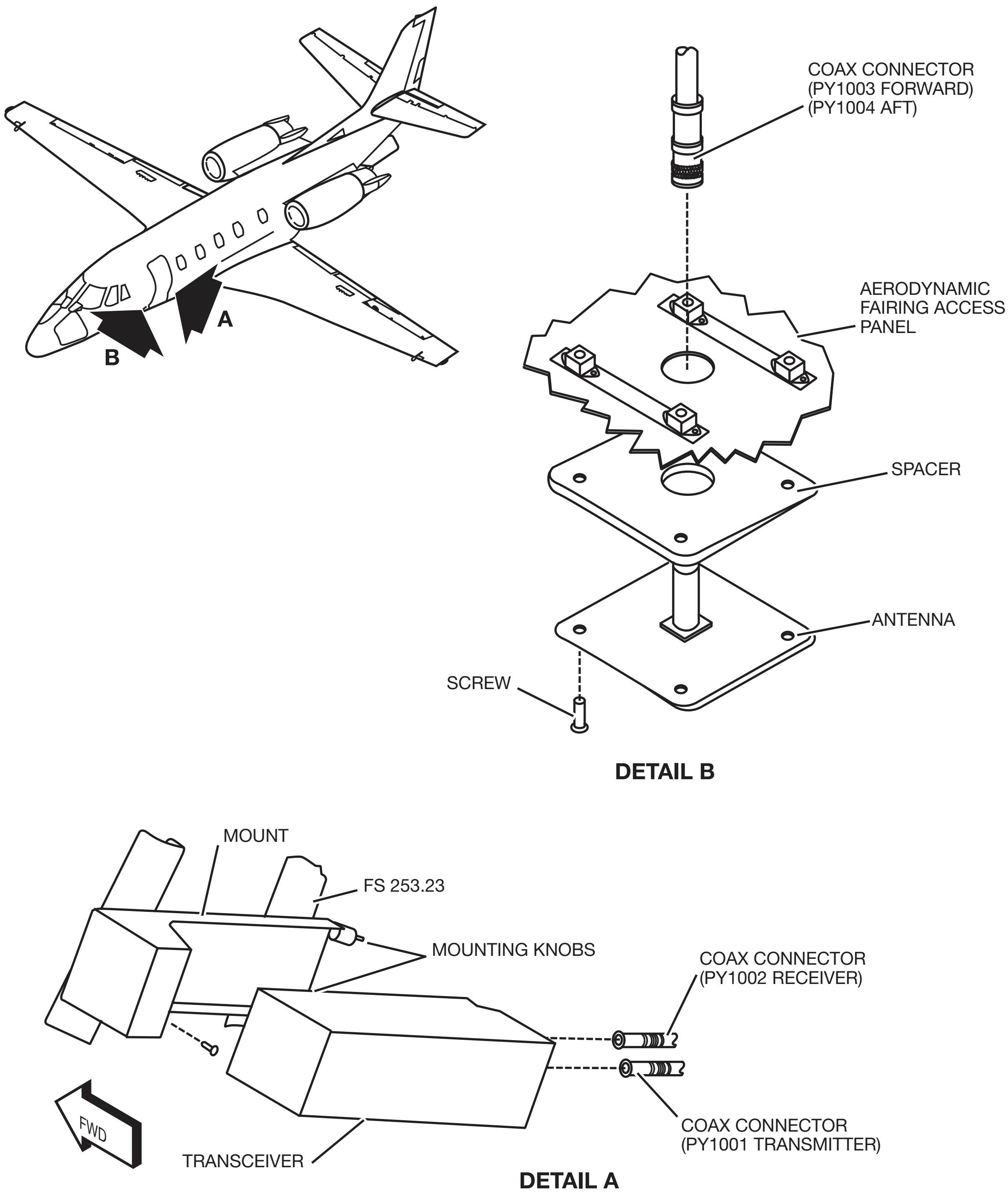


Figure 34-5. Collins (ALT 55B) Radio Altimeter

Collins ALT-55B Radio Altimeter

(Aircraft 5001–5312, 5314–5317)

Description

The ALT-55B radio altimeter provides the pilot with an indication of the aircraft vertical height above the terrain, when flying from ground level to 2500 feet (762 meters) above ground level (AGL). Radio altitude is displayed on the PFDs.

The radio altimeter system consists of a transceiver, a transmit antenna, and a receive antenna (Figure 34-5).

The transceiver is a solid-state receiver/transmitter. The transmitter has an output of 4300 MHz. A 100 Hz modulation signal causes the transmitter output to deviate from 4250 to 4350 MHz at a 100 Hz rate.

Radio altimeter information is provided on the pilot and copilot PFDs. There is a visual readout of the aircraft altitude above the terrain, on the lower portion of the attitude director indicator (ADI). The PF decision height (DH) is set by the DH TST knob on the pilot and copilot DC-550 display controllers. The DH display is a digital readout on the lower right side of the ADI display.

The DH can be set to any altitude between 0 and 999 feet (304.5 meters). When the aircraft descends to the selected altitude, the DH MIN indicator illuminates, providing a visual indication of the DH being attained.

When the DH TST knob is depressed, a functional test of the system is performed (excluding antennas).

A warning horn sounds when the aircraft reaches the DH set on the pilot PFD. A different DH value may be set on the copilot indicator. The DH MIN warning light on the copilot indicator illuminates when the altitude is passed; however, the DH warning horn does not sound. The DH warning horn is controlled only by the pilot DH value.

Operation

The transmitter output is a 4250–4350 MHz microwave signal, modulated at 100 Hz. The transmit antenna beams the radio frequency signal to the terrain, which reflects the signal from the ground to the receiver antenna.

During the time that the transmit signal travels to the terrain and returns, the transmitter changes frequency. The altimeter compares current transmitter frequency with the new signal frequency to determine the frequency difference.

The frequency difference is in proportion to the time required for the transmit signal's round trip to the terrain. Therefore, the frequency difference is in proportion to the aircraft altitude above the terrain.

The frequency difference is converted to a DC analog altitude signal with output voltage in proportion with the altitude.

Above 2500 feet (762 meters) or when the received signal is weak, the radio altimeter display is blank. If the altimeter system receives a weak signal (or otherwise senses invalid data) the digital window displays dashes.

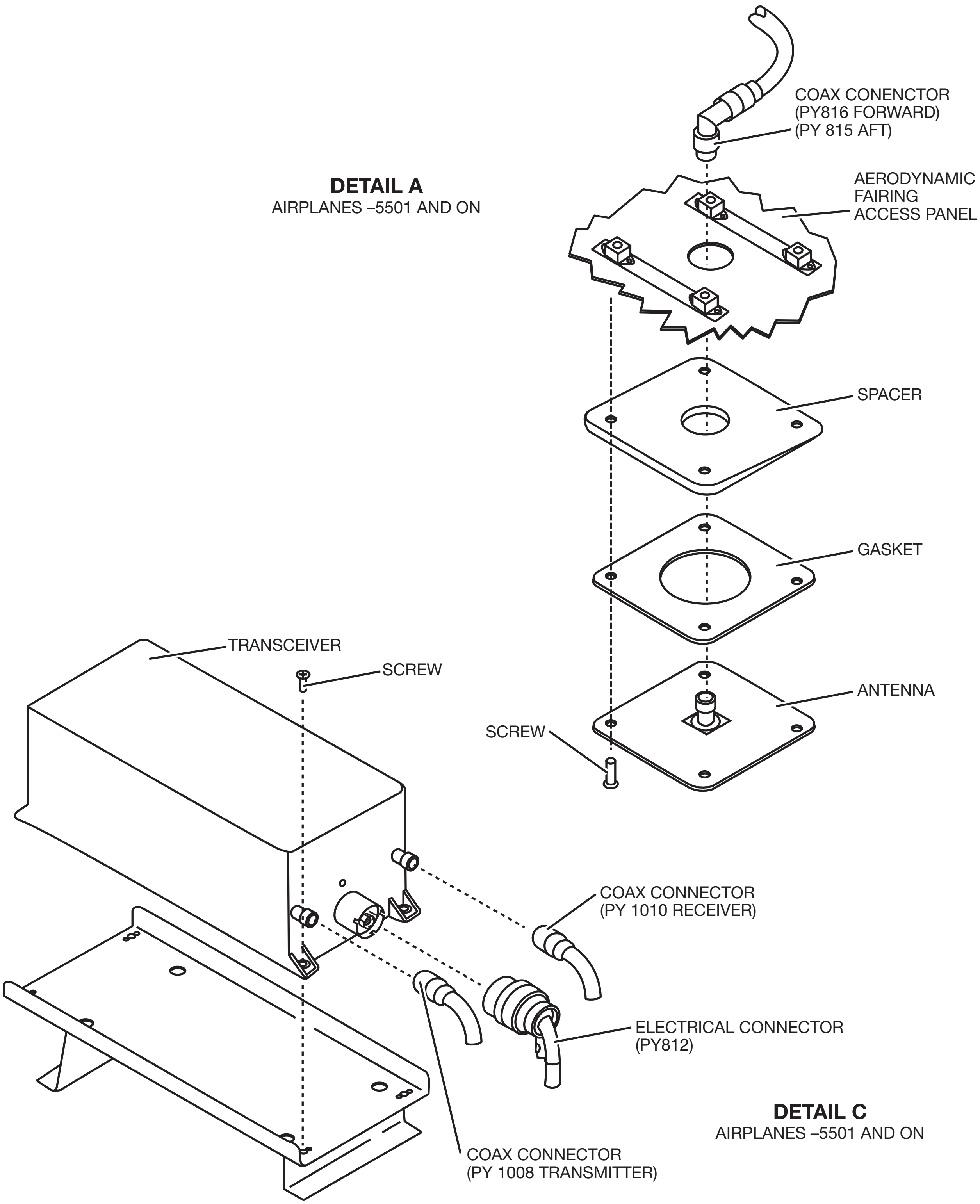


Figure 34-6. Honeywell AA-300 Radio Altimeter

Honeywell AA-300 Radio Altimeter

(Aircraft 5313, and 5318 and On)

Description

The AA-300 radio altimeter is a high-resolution, short-pulse radio system used for automatic continuous operation over a wide variety of conditions (Figure 34-6). The receiver/transmitter sends altitude outputs to the IC-600/IC-615 computer, which provides information for the PFD displays. Receiver/transmitter also provides information to the ground proximity warning system. The radio altimeter system has transmit and receive antennas, and a transceiver.

The AA-300 radio altimeter system utilizes an RT-300 transceiver. The RT-300 transceiver is a solid-state unit, operating at 4300 MHz with a pulse recurrence frequency of 10 KHz and a pulse width of 60 micro-seconds.

Information from the radio altimeter is displayed on the pilot and copilot PFD units.

The radio altimeter altitude digital read-out is displayed in lower part of the ADI on both PFDs. The digital altitude readout display is green, except when the aircraft is at or below DH; at which time the display becomes amber. At 2500 feet above ground level, no altitude is displayed. If the radio altimeter is invalid, a red box reading “RA” inside appears instead of the digital attitude display.

A solid brown raster band appears on the altitude tape (on the PFDs) when the radio altitude drops below 550 feet. The brown band covers the lower half of the altitude tape when aircraft is on the ground. There is a yellow line drawn where the brown raster and the gray band intersect (on the altitude tape). No written information is displayed in the brown band.

The DH may be set to a predetermined altitude by turning the inner TEST knob on the PFD controller (for aircraft with IC-600 computers). For aircraft with IC-615 computers, the DH is set with the MINIMUMS knob on the PFD bezel. The DH is displayed in a window on the lower right side of the attitude-director indicator display. When aircraft descends below that altitude, an amber DH (enclosed in a white box) appears on the upper left side of the attitude-director indicator display. The copilot and pilot DH indicators are independent of each other, even though only one radio altimeter is installed. The DH warning horn sounds only when the aircraft descends below the altitude selected in the pilot DH window (on the attitude director indicator display).

NOTE

There is a DH annunciator on both pilot and copilot PFDs. Each of the two DHs can be set independently, to control the DH annunciator on that indicator only (pilot or copilot annunciator). The different radio altitude indicators operate independently of each other, even though they are driven by the same transceiver.

If the radio altimeter receives/computes invalid data, dashes appear in the digital readout window.

The radio altitude test button (RA TEST) provides a functional self-test of the transceiver and indicator. Pressing the button causes the DH display window to display all dashes. The radio altimeter display indicates 100 feet and the DH annunciator is not displayed. After the button is released, actual altitude is displayed.

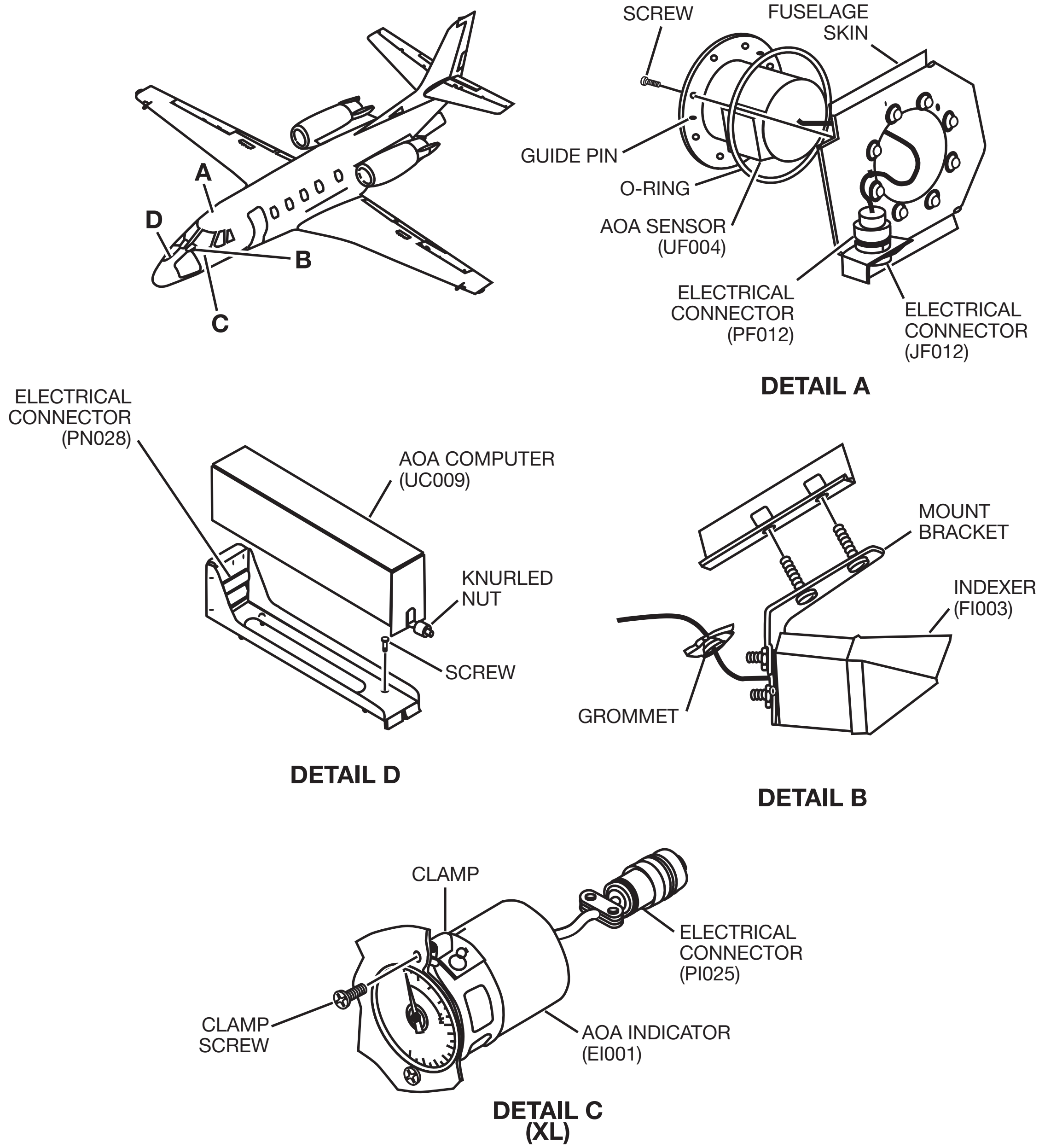


Figure 34-7. Safe Flight Angle-of-Attack (AOA) System

Safe Flight Angle-of-attack System

Description

The AOA system utilizes the local airflow direction relative to the aircraft pitch direction and flap position information, to calculate a normalized AOA (Figure 34-7).

The AOA computer is in the nose of the aircraft. The computer provides compensating circuitry to adjust for installation variables in the airflow sensor and logic lights, which assist in troubleshooting the system.

The AOA sensor on the right side of the fuselage (at FS 165, WL 119.81) provides an input to the signal-summing unit. The vane is electrically heated to provide anti-icing capability.

The AOA indicator is on the pilot instrument panel. The indicator is electrically operated. It has one pointer and a dial with a scale range from zero to 1.0. At 0.6. A white mark indicates a normal approach speed. Low speed is indicated by a red band. Overspeed is indicated by a yellow band.

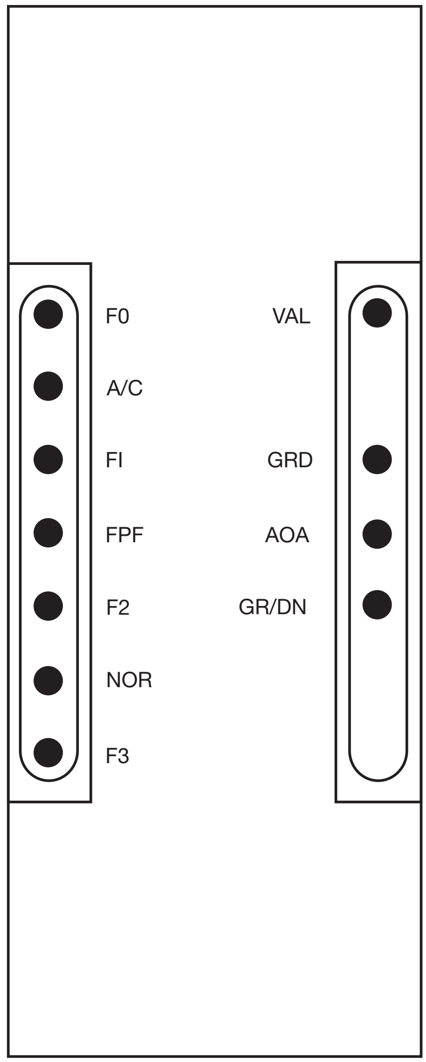
The indexer lights on the center windshield post provide a heads-up display of AOA whenever the aircraft is in the air with the gear down-and-locked. There is a 20-second delay after the gear is down-and-locked before the indexer illuminates to prevent operation at takeoff.

Operation

The AOA system determines the local air flow direction relative to the aircraft pitch axis. Using flap position information, it computes a normalized AOA for the AOA indicator, indexer, and low speed awareness. A reading of 1.0 on the indicator signifies stall attitude with 100% lift being used. A reading of 0.0 indicates a 1G condition with no lift being used. A reading of 0.6 is equivalent to 1.3 times stall speed. Full illumination of the green split-ring on the indexer signifies that the AOA setting is on the target speed, corresponding to a reading of 0.6 on the indicator. Illumination of the red light indicates a speed below target, while illumination of the amber light indicates a speed above target. The brightness of the indexer lights gradually increase or decrease in proportion to the speed error.

AOA information is also displayed with a low speed awareness bar on the airspeed tape on the primary flight displays (PFDs). A white bar is the equivalent of 1.3 to 1.2 times stall speed; an amber bar 1.2 to 1.1 times stall speed; and a red bar less than 1.1 times stall speed.

NOTES



RED LEDS ON - FAILURE INDICATION
A/C, FPF, NOR, AOA

GREEN LEDS ON - CONDITION INDICATOR
F0, F1, F2, F3, VAL, GRD, GR/DN

Figure 34-8. Safe Flight Angle-of-Attack Computer

Diagnostics

Troubleshooting of the Safe Flight AOA system utilizes the light emitting diodes (LED) within the AOA computer to indicate the possible source of a fault (Figure 34-8).

Troubleshooting Using LED Displays

F0—This LED indicates 0° flaps position when illuminated.

A/C (Aircraft Component)—When this LED is illuminated, it indicates that input to the AOA computer is outside the normal range; suggesting a failure in the aircraft wiring or in a related aircraft component.

F1—This LED indicates 7° flaps position when illuminated.

FPF (Flaps Pot Fail)—When this LED is illuminated, it indicates that the flaps potentiometer output voltage is outside normal range; suggesting either a short circuit or an open circuit.

F2—This LED indicates 15° flaps position when illuminated.

NOR (Normalization Circuits)—When this LED is illuminated, it indicates a failure in the normalization circuits within the AOA computer.

F3—This LED indicates 35° flaps position when illuminated.

VAL (Valid)—When this LED is illuminated, it indicates the AOA computer output is valid. When extinguished, it indicates that the flag has failed. A failed indication can be caused by a power failure, a failure in the AOA circuit (a primary or secondary of the AOA sensor or interconnect open), or a failure in the flaps potentiometer (open or shorted).

GRD (on ground)—This LED illuminates to indicate the left squat switch is in the ground position (true only when VAL LED is ON). Approximately 20 seconds after the squat switch goes to the in air position, this LED extinguishes.

AOA (computer)—When this LED is illuminated, it indicates a failure of the AOA computer.

GR DN (gear down)—When this LED is illuminated, it indicates that the landing gear is in the down position. When this LED is extinguished it indicates that the landing gear is in the UP position.

NOTES

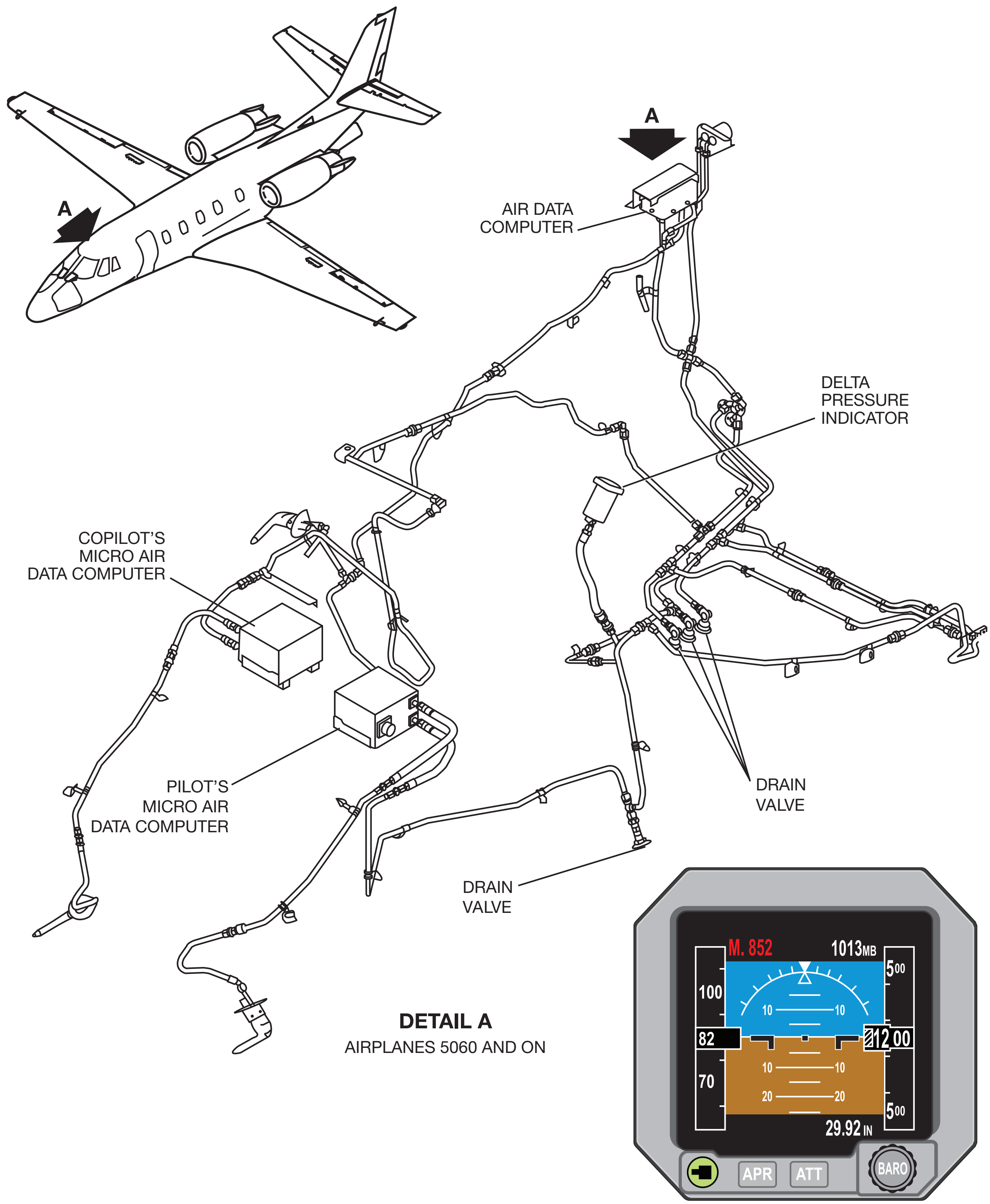


Figure 34-9. Meggitt Secondary Flight Display System

MEGGITT SECONDARY FLIGHT DISPLAY SYSTEM

Description

The secondary flight display (SFD) is an instrument on the panel, which operates in conjunction with a MADC to provide an indication of (Figure 34-9):

- Aircraft attitude
- Airspeed
- Mach number
- Altitude on a single display

Solid-state inertial sensors provide aircraft attitude. Airspeed and altitude are provided by the MADC.

The MADC is a pressure-sensing instrument, which provides highly accurate air data to the SFD. The MADC provides indications of the aircraft:

- Altitude
- Airspeed
- Mach number
- Barometric reference settings

Indication of aircraft data is displayed by a color active matrix liquid crystal display. Two control buttons and a baro set display control knob are on a front bezel. The bezel also houses the ambient light sensor, which automatically adjusts the luminance level of the unit.

The MADC is housed in a sealed case with integral pitot and static pressure ports at one end. The pressure ports are connected to internal sensors. To prevent ingress of dirt and dust particles, both pitot and static ports are fitted with mesh filters. Electrical connection to the aircraft power supply is through a connector situated above the pressure ports. The electrical connector is also used to link the air data unit with the SFD.

Operation

The SFD system is controlled by a switch marked STDBY PWR/OFF/TEST (SI001) on the pilot lower instrument panel. A separate battery pack (when fully charged) allows for 30 minutes of operation in the event of total loss of electrical power. The battery pack provides power for:

- SFD
- Air data unit
- 5-volt emergency lighting system for the N_1 ITT Indicator

The SFD uses air pressures from standby pitot and static system. Air pressure enters the MADC and is converted to digital output displayed on (SFD).

SFD has a built-in test feature, which automatically detects any failure of the display at power up and during continuous operation. If a failure is detected, a message flag appears.

When it is not possible to display an appropriate message, the display backlight is switched off.

The MADC uses pressures provided by the standby pitot and static systems. Pitot and static pressure is received and converted to digital output by two solid-state pressure sensors. The output from the sensors is displayed on the SFD indicator as altitude, calibrated airspeed and Mach number.

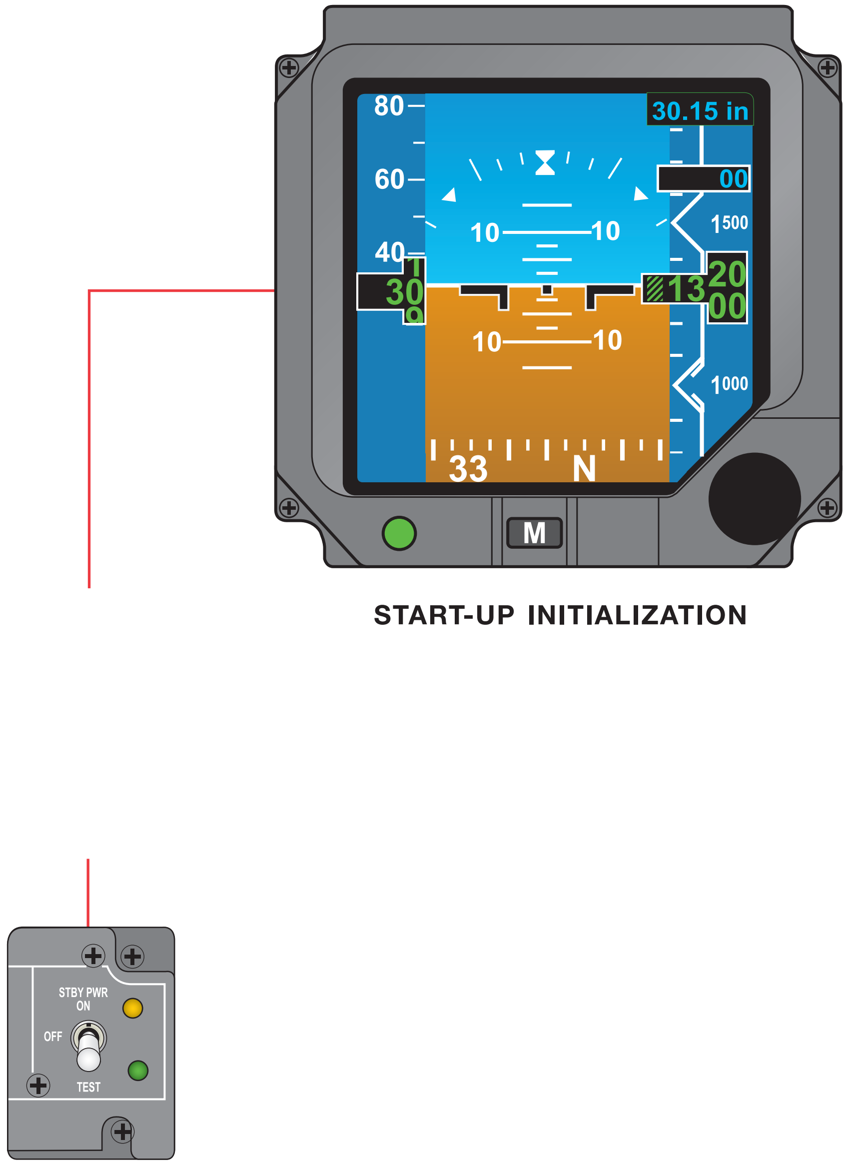


Figure 34-10. GH-3000 Electronic Standby Instrument System

GH-3000 ELECTRONIC STANDBY INSTRUMENT SYSTEM

Description

(Aircraft 5357 and On)

This section describes the GH-3000 electronic standby instrument system and its operation (Figure 34-10).

The electronic standby instrument system (ESIS) has an electronic standby attitude indicator (ESAI) that indicates pitch, roll, and skid/slip information with an internal three-axis inertial-sensor cluster.

The ESIS has an ESAI, a detachable configuration module (DCM), an air data computer, and a magnetometer.

The DCM has sufficient memory to record information specific to the hardware and software configurations for each installation such as:

- Panel angle
- Navigation interface
- Aircraft heading calibration
- Display format

When the ESAI is removed from the aircraft, the DCM stays attached to the aircraft wiring harness that mates with the ESA. When the DCM is attached to a new or replaced line replaceable unit (LRU) there is no need to reconfigure the unit.

Air data information is received from the ADC by a dedicated ARINC-429 data bus to the ESA. The magnetic heading is received from the magnetometer through a dedicated RS-422 data bus from internal inertial sensors. Navigational data is received from the aircraft VHF navigation system by an RS-422 data bus. All of the system components receive electrical power from the left feed bus in the left J-box. If normal power is unavailable, a standby battery in the left nose baggage compartment supplies the unit with electrical power.

Operation

The M button below the ESAI display shows the menus on the screen. To scroll through the menus, turn the adjustment knob on the bottom right corner of the ESAI.

Menu items that are followed by three dots have a submenu that can be viewed if that menu is selected. Push the adjustment knob to display the submenu. Once the submenu has been displayed, adjustments and selections are made by turning the adjustment knob. To toggle on and off, push the adjustment knob.

Menu access is closed when a setting is selected, or when the M button is pushed. Menu access also ends after 15 to 20 seconds of inactivity. Menus are available are as follows:

FAST ERECT—Push the knob to start.

SET BRIGHTNESS OFFSET—Push the adjustment knob for the submenu. Turn the knob to adjust brightness, then push the adjustment knob to end.

SET HEADING—Push the adjustment knob for the submenu. Turn the adjustment knob to set the heading, then push the knob to end.

NAV (ON or OFF)—Push the adjustment knob to toggle to on or off.

BARO TYPE—Push the knob for the submenu. Turn the adjustment knob to select the type, then push the adjustment knob to finish.

IAS TAPE DIRECTION (UP or DOWN)—Push the adjustment knob to toggle to the opposite condition.

To get the standard pressure barometric setting: Push the adjustment knob. Make sure that no menu function is displayed before pushing the adjustment knob.

Figure 34-11. Aeronetics HSI-315 Standby Horizontal Situation Indicator (HSI)

Figure 34-12. Magnetic (Standby) Compass

The GH-3000 ESAI has an EXT MANUV annunciation that displays when the system detects a long period of time in an attitude banked more than 7°. The EXT MANUV annunciation also displays if the system senses that it is not within 8° of the magnetic heading (provided by the magnetometer) for a long period of time.

Navigational data information inputs are made at the source navigation units, except for those navigational aid items that are configured for (and available through) the menu access.

The GH-3000 system has an attitude declutter feature that removes the navigational aid information from the display when the aircraft is in a bank more than ± 65°; or in a pitch that is +30° or –20°. The navigational aid information is displayed when the aircraft returns to approximately level flight.

AERONETICS HSI-315 STANDBY HORIZONTAL SITUATION INDICATOR

Description

The Aeronetics HSI-315 is a internally-lighted horizontal situation indicator (HSI) (Figure 34-11). This indicator provides the following displays:

- Heading
- Automatic direction finder (ADF) bearing
- Manually selected course
- Very high frequency omnidirectional radio range–Localizer (VOR–LOC)
- VOR to–from indication
- Glideslope information

Indicator maintenance consists of removal and replacement. Functional testing of the indicator is done in conjunction with functional testing the other systems.

MAGNETIC COMPASS

Description

A conventional magnetic (standby) compass is suspended from the center windshield divider for navigation in case of instrument or electrical system failure (Figure 34-12).

There is a compass card near the compass, containing compass deviation adjustment information.

Ensure that all metals within a one-foot radius of the compass are nonmagentic.

NOTES

Figure 34-13. Honeywell Primus 1000 Electronic Flight Instrument System (EFIS) (Aircraft 5001 thru 5500)

HONEYWELL PRIMUS 1000 ELECTRONIC FLIGHT INSTRUMENT SYSTEM

Description

(Aircraft 5001 Thru 5372)

The Honeywell Primus 1000 electronic flight instrument system (EFIS) is a comprehensive electronic display system that provides the pilot and copilot with displays of (Figure 34-13):

- Flight altitudes
- Airspeeds
- Vertical speed
- Attitude
- Heading
- Course orientation
- Flight path commands
- Weather
- Mapping presentations

The EFIS consists of the following components:

Integrated Avionics Computer (IAC)—Each IAC contains a symbol generator that functions as the data processor for the display systems. In normal configuration, the left IAC drives the pilot displays and the right IAC drives the copilot displays. Either IAC can drive the three-display units.

Display Controller—The DC-550 display controller interfaces with the IC-600 computer to provide for pilot selection of display modes and formats on the PFD.

Remote Instrument Controller—The RI-553 remote instrument controller allows the pilot to manually select heading and course for display on pilot and copilot HSIs.

Mode Selector—The MS-560 mode selector provides pushbutton controls and mode annunciators for the integrated flight guidance system.

Multifunction Controller—The MC-800 multifunction display controller (MFD) allows the pilot to select display modes and formats on the MFD. It interfaces with both IC-600 computers through a serial digital bus.

Electronic Display—The DU-870 display unit is the electronic display used to provide the PFDs and the MFD in the Primus 1000 system. Video and deflection signals from the symbol generator function in the display guidance computers drive the (tubes) electronic display.

NOTES

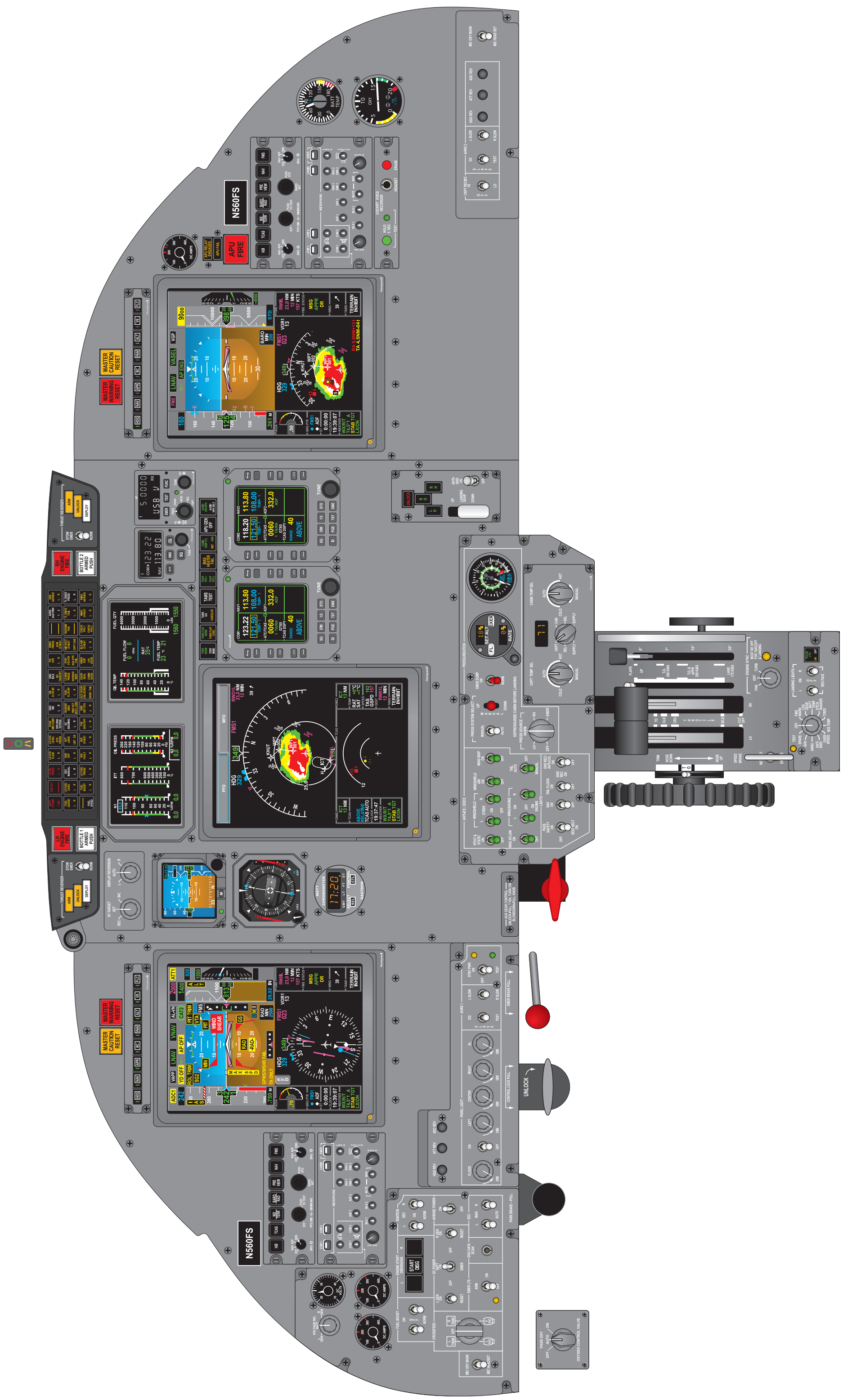


Figure 34-14. Honeywell Primus 1000 Electronic Flight Instrument System (EFIS) (Aircraft 5501 and On)

HONEYWELL PRIMUS 1000 ELECTRONIC FLIGHT INSTRUMENT SYSTEM

Description

(Aircraft 5501 and On)

The Honeywell Primus 1000 EFIS is an electronic display system that includes the following displays for the pilot/copilot (Figure 34-14):

- Flight altitudes
- Airspeeds
- Vertical speed
- Attitude
- Heading
- Course orientation
- Flight path commands
- Weather
- Mapping presentations

The EFIS consists of the following components:

Integrated Avionics Computer (IAC)—Each IAC has a symbol generator that functions as the data processor for the display systems. In normal configuration, the left IAC operates the pilot displays and the right IAC operates the copilot displays. Either IAC can operate the three-display units.

Display Controller—The DC-550 display controller interfaces with the IC-615 computer, allowing the pilot to select different display modes and formats on the PFD.

Remote Instrument Controller—The RI-553 remote instrument controller lets the pilot manually select heading and course for display on the pilot and copilot HSIs.

Mode Selector—The MS-560 mode selector gives pushbutton controls and mode annunciation for the integrated flight guidance system.

Multifunction Controller—The MC-800 multifunction display controller lets the pilot select display modes and formats on the MFD. It interfaces with both IC-615 computers over a serial digital bus.

Electronic Display—The DU-1080 display unit is the electronic display used for the PFDs and the MFD in the Primus 1000 system. Video and deflection signals from the symbol generator function (in the display guidance computers) drive the electronic display (tubes).

NOTES

Figure 34-15. Attitude Heading Reference System (AHRS)

ATTITUDE HEADING REFERENCE SYSTEM

Description

The attitude and heading reference system (AHRS) is an inertial sensor installation, which provides aircraft attitude, heading, and flight dynamics information to cockpit displays, flight controls, aircraft systems and instruments (Figure 34-15). The AHRS differs from conventional vertical and directional gyro systems in that the gyroscopic elements are fiber optic gyros that are "strapped down" to the principal aircraft axes.

The Citation XL/XLS has two AHRS. The No. 1 AHRS is powered by NORMAL DC power and normally supplies data for the pilot IAC and flight displays. The No. 2 AHRS is powered by the EMERGENCY BUS and normally provides flight information to the copilot IAC, flight displays, and the standby HSI. The AHRS system has a standby battery to provide DC power for temporary power loss (i.e. during engine start). If either AHRS flight data output becomes invalid, the other AHRS can be utilized in a reversionary mode to restore lost data.

Each AHRS is made up of the following components:

- The flux valve detects the relative bearing of the earth's magnetic field and is usually in the wing or tail section, away from disturbing magnetic fields.
- The compensator/controller provides correction for magnetic variation from the flux valve.
- The attitude heading reference unit (AHRU) is the major component of the system and is composed of four major subsystems.
- The inertial measurement unit (IMU) senses the aircraft movements, acceleration/deceleration, and rotation about the aircraft axis. It contains the rate gyros, accelerometers, and support electronics.
- The central processor unit (CPU) performs the computations necessary to extract the attitude and heading information. In addition to its computational activities, the CPU controls and monitors the operation of the entire system.
- The input/output (I/O) unit supervises the handling of data between components in the system.
- The power supply converts aircraft power to the regulated DC voltages required by the system.

Two modes are provided for routine operation: the normal mode for attitude, and the slaved mode for heading.

The normal mode uses true airspeed from the air data computer to compensate for acceleration-induced attitude errors. The slaved mode uses the flux valve to align the heading outputs.

Two reversionary modes are provided to maintain performance in the event of certain types of system failures. These are basic and DG. The AHRS system reverts from normal to basic mode if the MADC TAS output becomes invalid. This results in an attitude display similar to that of a conventional vertical gyro, subject to drift and acceleration errors.

The DG mode is selected by placing the DG SLAVE-TEST switch from the slave position to the DG position. This disables the automatic slaving of the AHRS heading output. Operation in this mode is similar to that of a conventional directional gyro. A two-speed manual slaving input switch is provided to manually slew the heading output while operating in the DG mode. Although the DG mode may be entered at any time, the mode is usually reserved for operation in the event of a slaving failure.

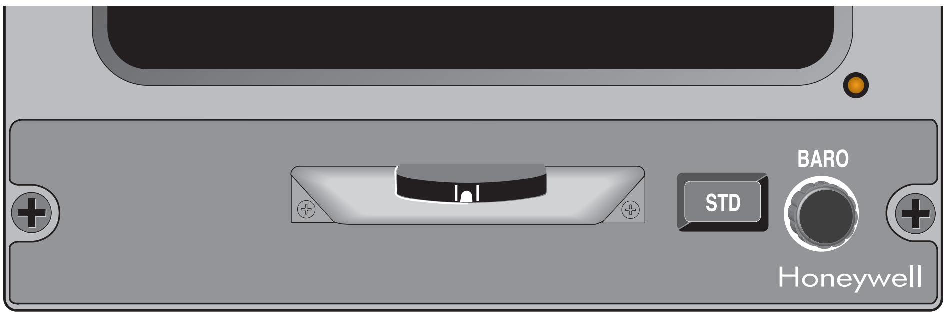


Figure 34-16. PFD Bezel Controller

Figure 34-17. MFD Bezel Controller

CONTROLLERS

BL-870 PFD Bezel Controller

The PFD bezel controller (Figure 34-16) is on the lower front of the PFD and provides the following functions:

STD—Pushbutton returns the barometric altimeter correction to standard value (29.92 in. Hg or 1013 HPa).

BARO—The rotary set knob allows selection of reported barometric altimeter correction in either inches Hg or HPa as determined by the IN/HPA pushbutton.

When the pilots are displaying cross-side MADC (amber) data on their PFDs, only the operating side MADC PFD bezel has control over both BARO settings.

The BARO set operates independently from the display controllers and does not require that the display controller be functional to set data.

BL-871 MFD Bezel Controller

The MFD bezel controller allows access for setting takeoff V-speeds, landing V-Speeds, and vertical navigation (VNAV) data through five menu-item pushbuttons and a rotary knob (left side) for setting data on various menus (Figure 34-17). The right rotary knob is used solely for altitude preselect inputs (displayed simultaneously on the MFD and both PFDs).

NOTES

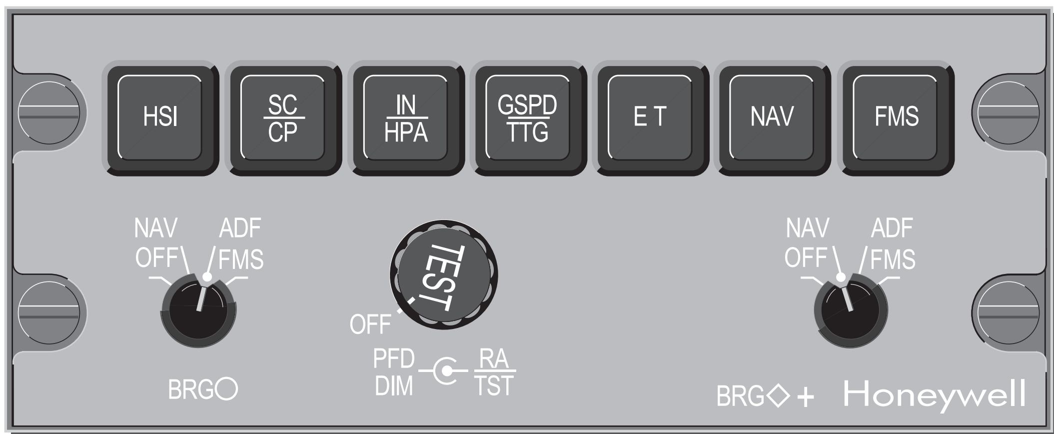


Figure 34-18. DC 550 Display Controller (Aircraft 0001—5372)

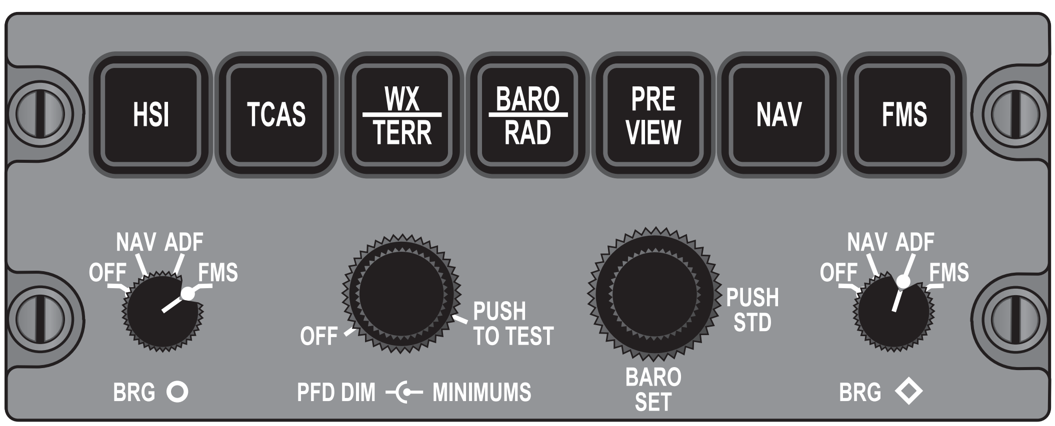


Figure 34-19. DC 550 Display Controller (Aircraft 5501 and On)

DC-550 Display Controller

The display controllers, directly to the left and right (respectively) of the pilot/copilot PFDs on the instrument panel, allow the pilots to select various formats on the PFDs (Figures 34-18 and 34-19).

These functions are described below:

HSI Button—Controls full or WX (partial compass display). Displays 360° in FULL mode and 90° in WX (ARC) mode. Successive pushes toggle between the two displays. WX returns can be displayed on the PFD when in WX mode and radar is transmitting.

SC/CP Button—Selects the flight director command bar display. Alternate action toggles between single cue and cross-pointer flight director display. Power-up state is single cue.

IN/HPA Button (inches of HG/Hecto-pascals)—Selects barometric display mode. Pressing the IN/HPA button toggles the display between inches of mercury and hecto-pascals

GS/TTG Button—Groundspeed (GS) or time-to-go (TTG) is displayed in the lower right center of the EHSI. Pressing the GS/TTG button provides alternating selection of GS or TTG to next station or waypoint.

ET Button—Controls a time-elapsed timer that appears in the EHSI location dedicated to GSPD/TTG. Initial actuation enters the mode at the previous position. If elapsed time is being displayed, it stops the display. Sequence of the ET button is: reset–elapsed time–stop-repeat.

NAV Button—Pressing the NAV button selects the VOR for display on the EHSI course deviation indicator (CDI). Pressing the button alternately selects NAV1 (green) and NAV2 (yellow) (displays VOR1 and VOR2 on the center right side of the EHSI; ILS 1 and ILS 2, if ILS frequency is tuned in NAV). The flight director interfaces with the NAV that is selected and displayed on the EHSI.

FMS Button—Selects FMS for display on the EHSI. The EHSI course needle represents FMS course information on the course deviation indicator. The FMS button first depicts onside data in magenta; and on second push displays cross-side data in yellow.

Bearing "O" Knob—This knob has four positions. The OFF position removes the No. 1 (blue) single line bearing pointer from the HSI display. In NAV position, VOR1 bearing information is displayed. In ADF position, the ADF1 bearing is displayed. Selecting FMS displays bearing to the next FMS waypoint in single FMS installations, or FMS1 data in Dual FMS installations.

NOTES

PFD DIM (Outer Concentric)—The DIM knob sets half the overall brightness of the PFD. When a reference level is set, photoelectric sensors maintain the relative brightness level in various lighting conditions. Turning the knob full counter-clockwise to the OFF position turns off the PFD, and reverts the display (through and EFIS backup mode) to the multifunction display. Sunlight increases the intensity of the display so that images are still visible.

EFIS backup is provided by the MFD as an addition to the existing EFIS reversionary modes. In case of failure of a PFD cathode ray tube, selection of an EFIS backup mode can be accomplished by turning OFF the PFD DIM button on the affected PFD. The MFD takes up the display selected on the controller. If both PFDs are OFF, the copilot PFD has priority on the MFD display. Dual reversion of both PFDs to the MFD is prohibited by limitation.

DH Knob (Inner Concentric)—Rotation of the DH knob adjusts the DH display on the EADI in 5-foot increments to 200 feet, and 20 foot increments above 200 feet to 990 feet. Rotating the knob fully counterclockwise removes DH information from the display.

Bearing Pointers

Bearing "♦" Knob—This knob has three positions. The OFF position removes the No. 2 double-line bearing pointer (white) from the HSI display. In the NAV position, the NAV2 bearing is displayed. In the ADF position, the ADF bearing is displayed in single ADF installations, (or ADF 2 bearing in Dual ADF installations). Selecting FMS displays bearing to the next FMS waypoint in single FMS installations (or FMS 2 data in Dual FMS installations).

NOTES

Diagnostics

Test Function

Test Function (TEST in Magenta) Pressing and holding the TEST button causes the displays to enter the test mode. Flags, cautions, and all flight director and mode annunciations are tested and presented on the display. Satisfactory or unsatisfactory test results are annunciated on the display. The test also results in a self-test of the radio altimeter system; 50 feet is indicated in green in the bottom of the EADI display, and the DH horn sounds.

The TEST button is wired through a squat switch and is completely active only when the aircraft is on the ground. The Primus 1000 test is not active in flight, but a self-test of the radio altimeter system may be made in flight if the GS capture mode is not active. The EFIS system also automatically self-tests when it is powered up, but this is normally not displayed due to the warm-up time of the EFIS tubes. If the test is not satisfactory it is so annunciated. Holding the test button for more than 5 seconds displays a maintenance test function of the PFD.

Integrated Maintenance Mode

To access the maintenance pages the DH must be set greater than 600, then press and hold the test button for 5 to 7 seconds, while holding the test button press push button #4 on the display controller, and then release both buttons. The maintenance pages can now be viewed. To access the different pages, rotate the DH/TEST knob and change the page numbers on the bottom of the PFD screen.

Honeywell has produced a SYSTEM TEST AND FAULT ISOLATION MANUAL (PUB # A15-1146-080) that describes all the different capabilities of the Integrated Maintenance Test.

NOTES

Figure 34-20. MFD Controller (Aircraft 0001—5372)

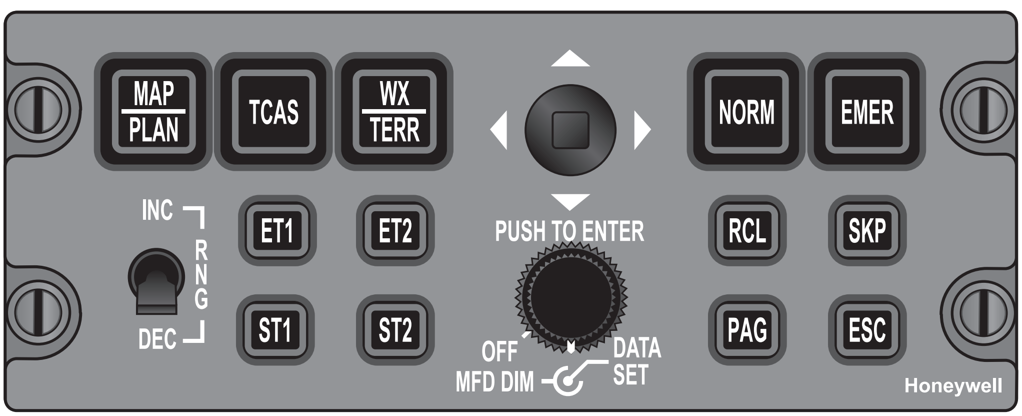


Figure 34-21. MFD Controller (Aircraft 5501 and On)

MULTIFUNCTION DISPLAY CONTROLLER

The MFD controller at the front of the pilot pedestal, allows mode selections, display control, and symbol generator reversion control of the pilot and copilot systems. TCAS control is also a function of the MFD controller (Figures 34-20 and 34-21).

Controls and Indications

Map Mode

The MAP function is a partial-arc, heading-up display, which is selected by the alternate-action MAP/PLAN pushbutton. The MFD display cycles from MAP to PLAN as the MAP/PLAN button is pressed. The MAP format allows totally independent use of the MFD display for navigation mapping and allows increasing of the maximum range, beyond normal radar range, on the display which normally serves as the radar indicator. Power-up mode is the MAP mode. To add weather to the display, press the WX button on the MFD controller.

The MAP format is always oriented to the aircraft heading, and the aircraft symbol is at the center of the display. When coupled to the FMS, the NAV route, with up to ten waypoints, can be displayed to the range limit. When weather returns are selected, range control defaults to the weather radar controller.

Plan Mode

In PLAN mode, the top of the display is oriented to True North; a three-inch range is displayed and centered horizontally on the displayed area. An aircraft symbol is plotted at present position (if present position is on the display) and is oriented with respect to heading. The PLAN mode display encompasses 360°. Weather radar returns cannot be presented in the PLAN mode.

TCAS mode—The TCAS button is optional and its button selects TCAS traffic display on the MFD display.

Weather Mode

The WX mode allows the MFD display to be used as a weather radar indicator. In WX mode, weather data is presented on the MFD and is superimposed upon the normal navigation display. Weather radar can be selected for display on the MFD only if MAP mode is selected. If the MFD is in PLAN mode, selection of WX mode forces the display into MAP mode. Range selection is controlled by the weather radar control on the pilot instrument panel. When the WX button is toggled, the progression of selection is: WX on, WX off. Annunciation of weather modes, warnings, and antenna angle are provided at the lower middle left of the MFD display. Annunciations are color-coded in magenta, green, and amber according to the importance of the display.

Timers

(Aircraft 5501 and On)

The ET 1 and ET 2 buttons cause an elapsed time counter to appear on the lower left corner of the MFD. The ST 1 and ST 2 buttons allow a time to be set and the elapsed time counter can then be used to count down to 0.

Checklist Controls

The Primus 1000 system can display both normal and abnormal checklists. The checklist must be provided by the operator, then loaded using a computer. The checklist is controlled through the MFD controller.

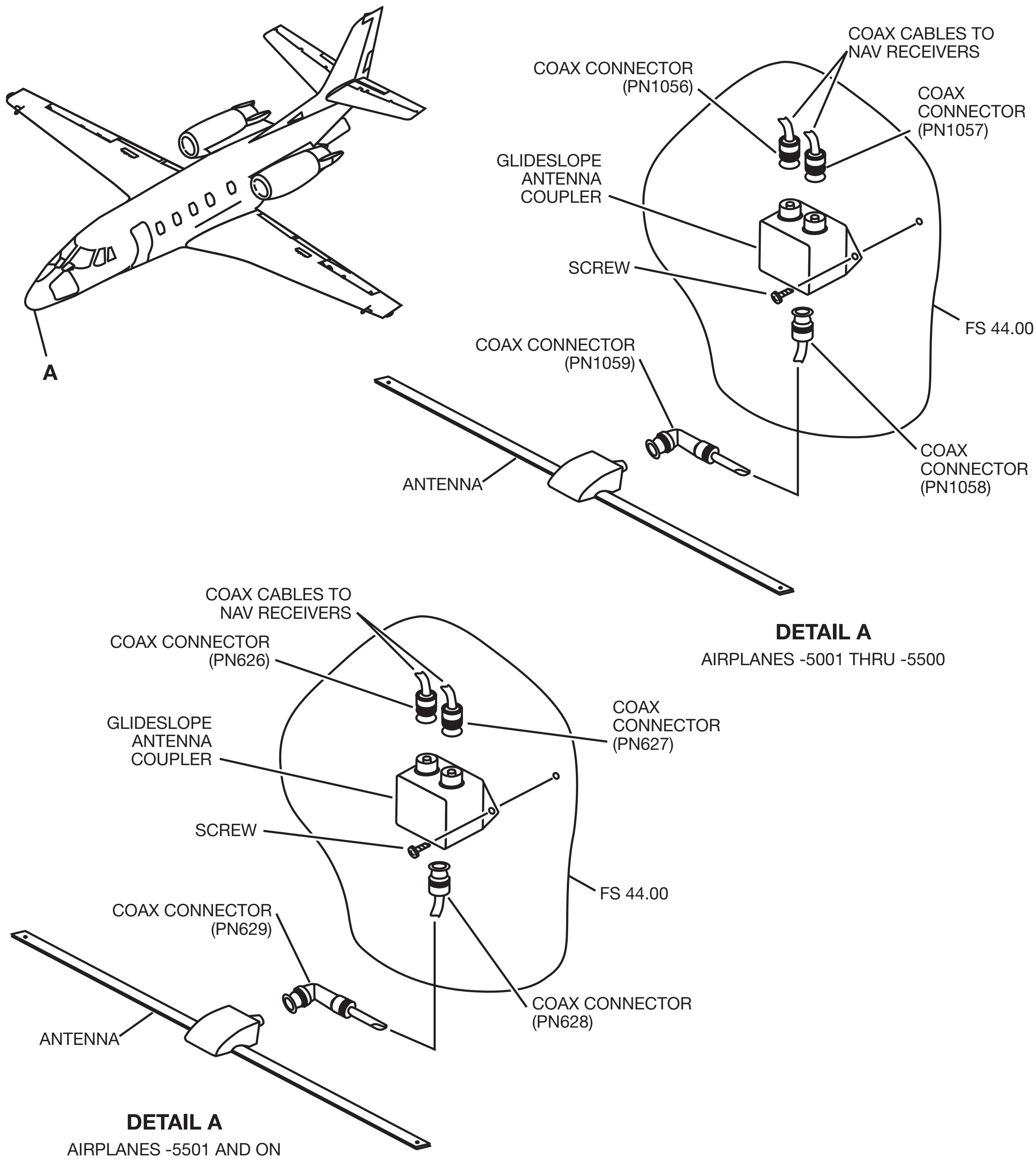


Figure 34-22. Landing and Taxiing Systems

LANDING AND TAXI NAVIGATION AIDS

Description

This section provides coverage of the landing and taxiing systems, which provide guidance during approach, landing, and taxiing. The localizer, glideslope and marker beacon systems are utilized for approaches and landings.

The navigation receivers contain the receivers for the localizer, glideslope and marker beacon. Selecting a localizer frequency automatically selects the paired glideslope channel.

The locations of the antennas for the localizer, glideslope and marker beacon are as follows:

- The localizer receives its signal through the navigation receiver antenna on the vertical fin.
- The glideslope has its own antenna inside of the nose radome The antenna is bonded to the bottom of the radome (Figure 34-22).
- The marker beacon has its own antenna on the bottom of the fuselage, on fuselage fairing panel 163BC (Figures 34-23 and 34-24).

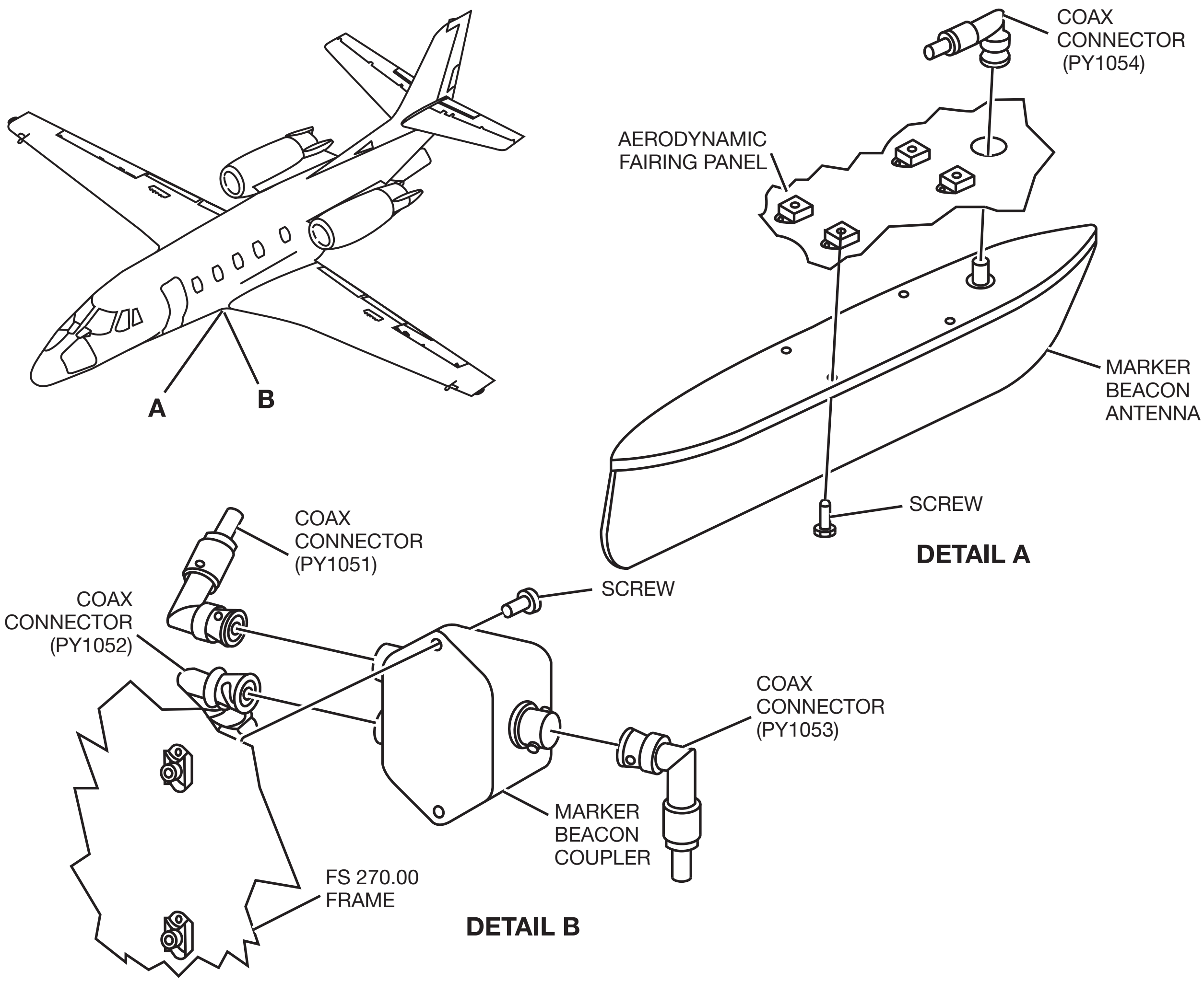


Figure 34-23. Marker Beacon

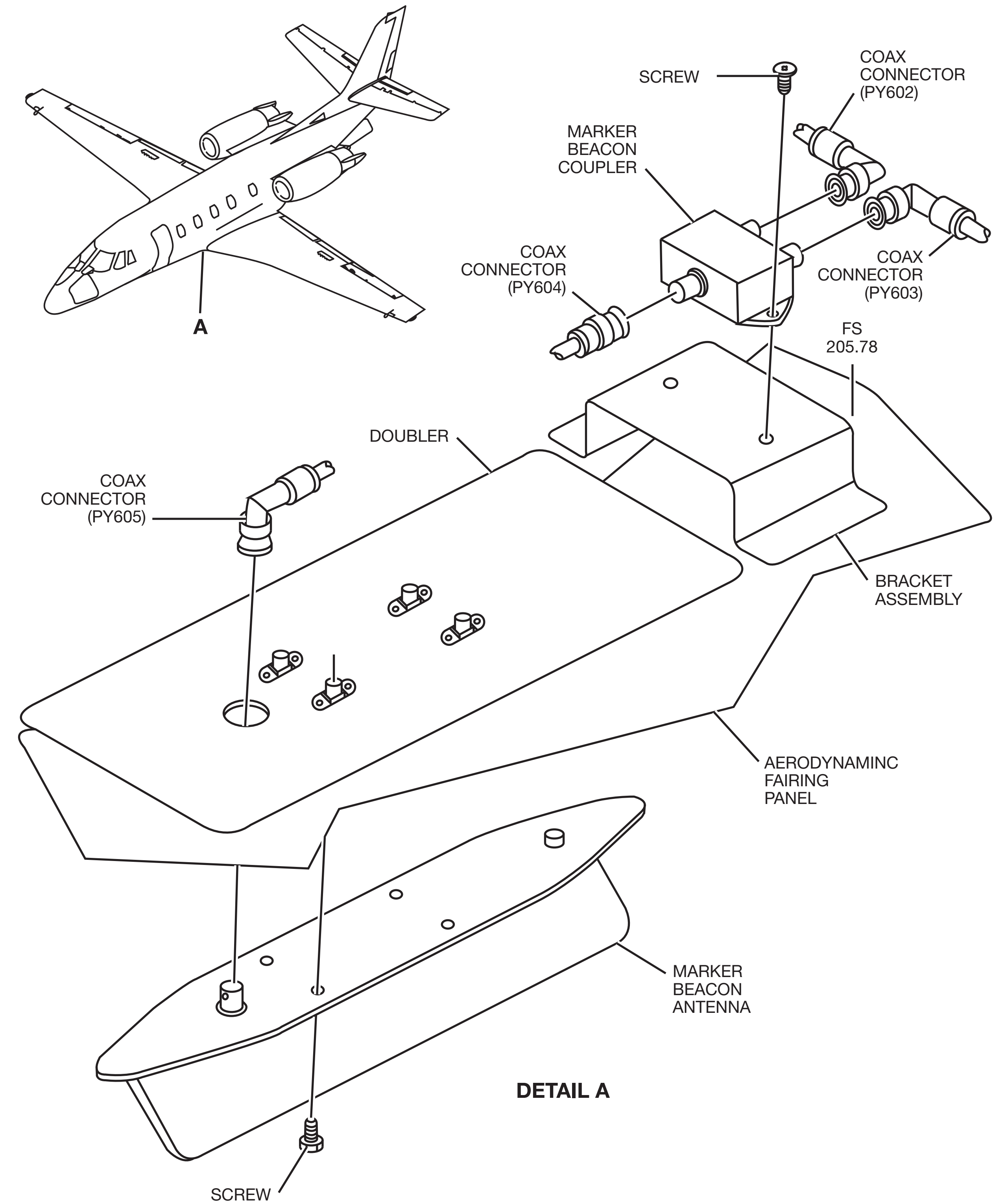


Figure 34-24. Marker Beacon Antenna

HONEYWELL PRIMUS II LOCALIZER, GLIDESLOPE AND MARKER BEACON

Description

The dual Primus II NAV radio system is a subsystem of the Primus II SRZ-850 integrated radio system. For description, operation, and maintenance, refer to Chapter 23—"Primus II Integrated Radio System".

The localizer, glideslope and marker beacon are the components of the instrument landing system (ILS). These components are part of the NAV radio system. The glideslope is automatically tuned when the NAV receiver is tuned to a localizer frequency.

The NAV receiver can be automatically tuned by the flight management system.

NOTES

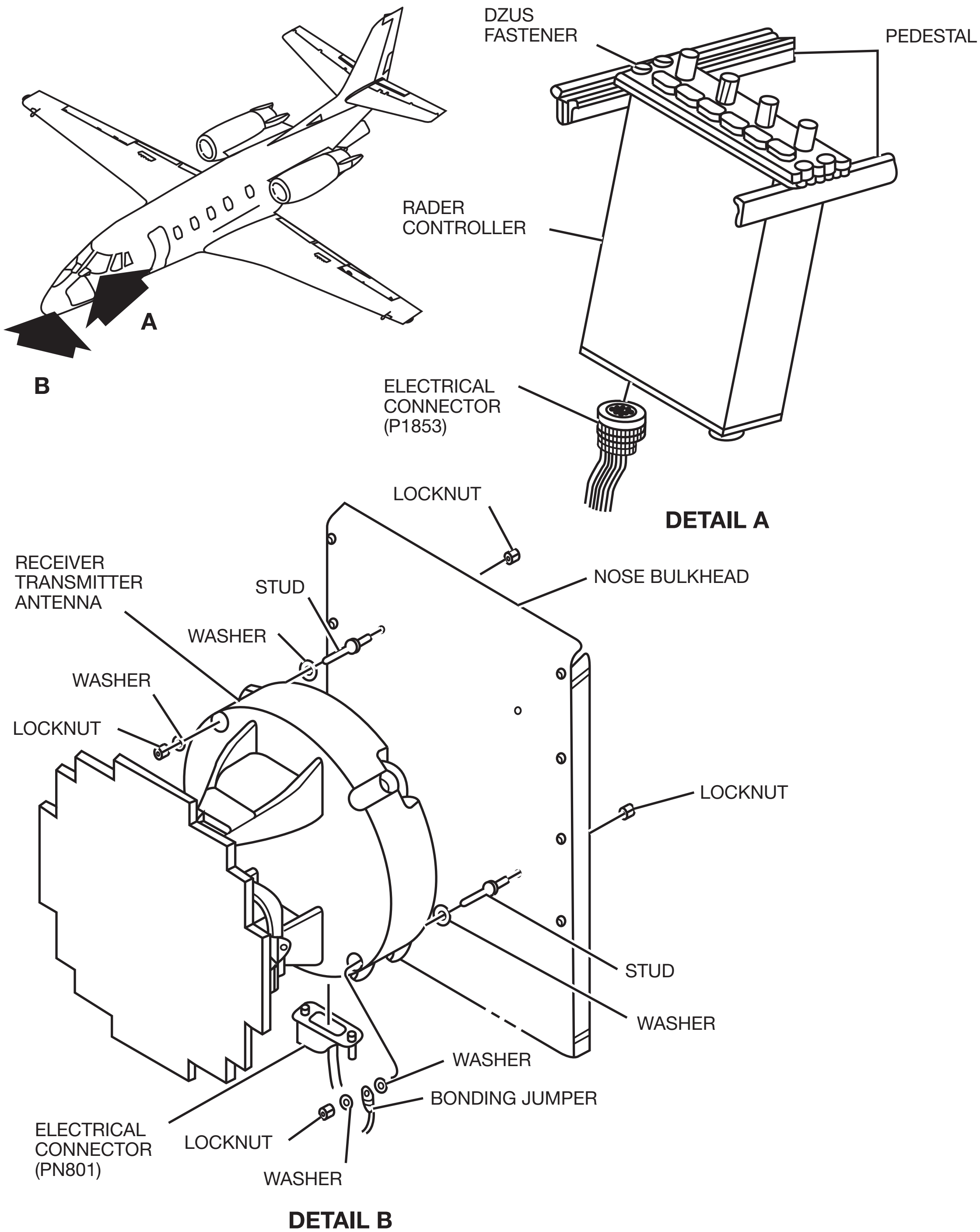


Figure 34-25. Honeywell Primus 880 Digital Weather Radar System

HONEYWELL PRIMUS 880 DIGITAL WEATHER RADAR SYSTEM

Description

The Primus 880 Digital Weather Radar system is a lightweight X-band, color digital radar for weather location and analysis, and for ground mapping (Figure 34-25).

The system detects precipitation in storms along the flight path of the aircraft and gives the pilot a color visual indication of storm intensity and turbulence. In weather detection mode, target returns are displayed in one of five video levels (0 thru 4). Zero is represented by a black screen, due to weak turns or no returns. Levels 1, 2, 3, and 4 (represented by green, yellow, red and magenta) show progressively stronger returns. Areas of potentially hazardous turbulence are shown gray/white in ground mapping mode. Video levels with increasing reflectivity are displayed as black/cyan (sky blue), yellow, and magenta.

The major components of the Primus 880 Digital Radar are the pedestal controller and the receiver-transmitter-antenna.

The system is operated in conjunction with the EFIS equipment to provide radar video on the EFIS displays.

Components

Receiver-Transmitter-Antenna

The receiver-transmitter-antenna is in the nose of the aircraft, protected by the radome. It is cantilever-mounted on the aircraft nose bulkhead. All components are integrated into a single assembly with no need for a waveguide.

NOTES

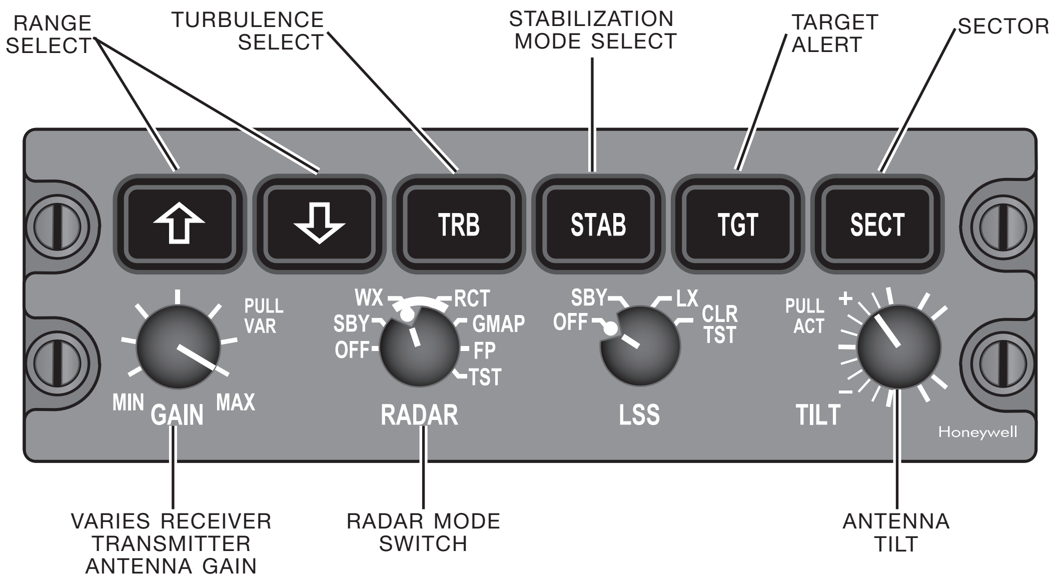


Figure 34-26. Radar Controller

Radar Controller

The Radar Controller (WC-880) is in the pedestal. The controller incorporates pushbutton and rotary switches (Figure 34-26).

All controls used to operate the Primus 880 system are on the pedestal controller. Functions of the controls/switches are as follows:

SECT (sector)—An alternate action pushbutton that selects either full azimuth scan angle (120°) or sector scan (60°).

TILT—A single-turn rotary control that varies antenna tilt between 15° up and 15° down. The range between ± 5° is expanded for ease in adjustment of antenna tilt. When pulled, AUTO TILT mode is engaged to adjust antenna tilt in relation to altitude and selected range.

GAIN—A single-turn rotary control that varies the receiver/transmitter/antenna receiver gain. Selection of RCT (react) on the MODE control overrides the variable gain setting, causing receiver gain to be fixed and calibrated.

RADAR—A rotary switch that selects primary radar modes.

OFF position—Removes power from system.

SBY (standby) position—Places system in non operational mode.

WX position—Selects the weather mode.

RCT position—Enables the cyan react field to indicate ranges at which the receiver calibration has been exceeded (controller only).

GMAP position—Places the system in ground map mode.

FP (flight plan) position—Selects the system flight plan (navigation) display mode.

TST (test) position—Activates the system self-test mode.

TGT (target)—An alternating action pushbutton, which enables the target alert function.

TRB (turbulence)—A momentary pushbutton, which selects the turbulence detection mode. In this mode, areas of potentially hazardous turbulence are displayed in gray–white in addition to the normal weather display. TRB may only be engaged in the WX mode at ranges of 50 nm or less.

STB (stabilization)—A momentary alternate action pushbutton, which selects stabilization ON/OFF.

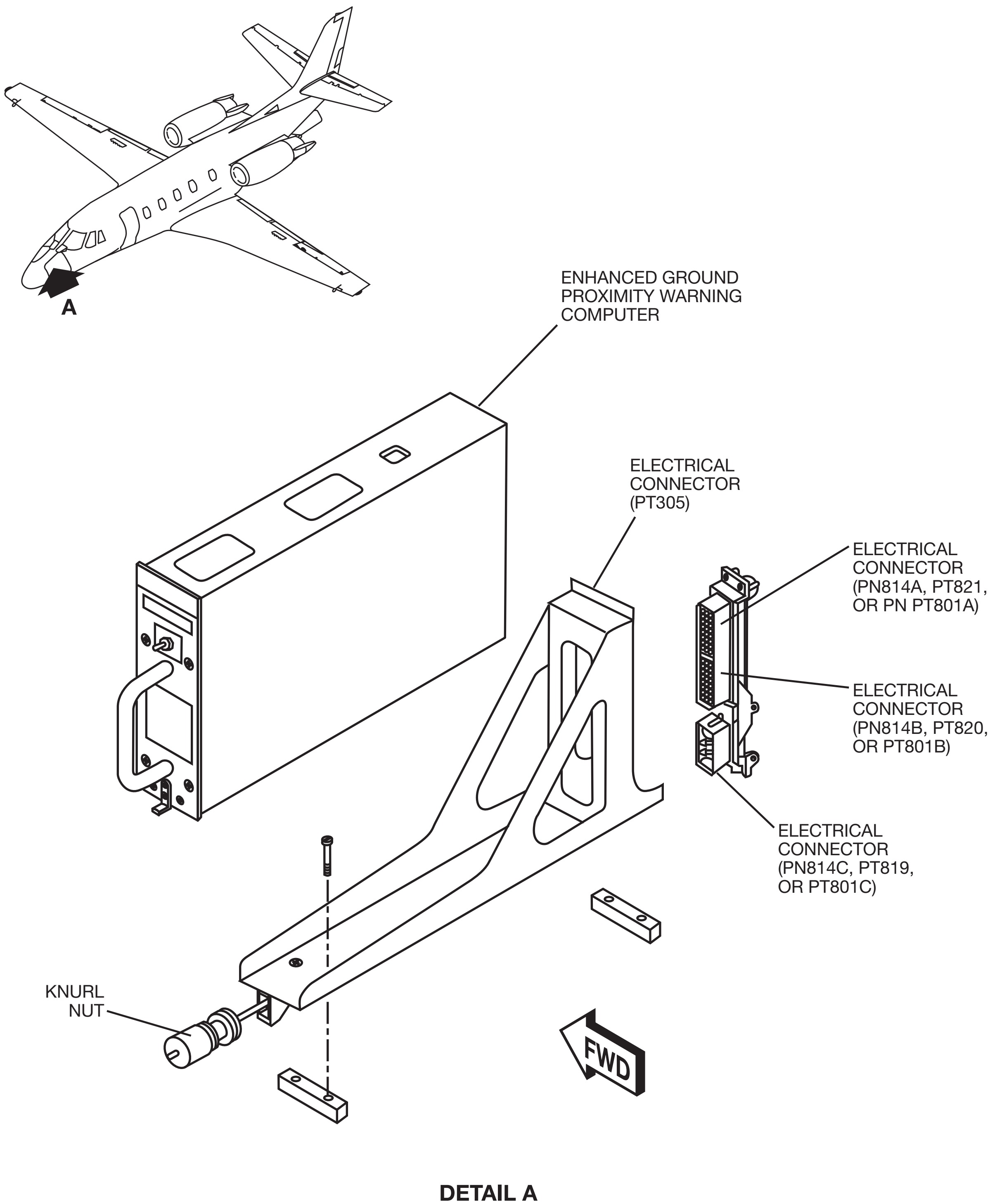
SLV (slaved)—A dead front annunciator used only in dual controller installations. One controller can be slaved to the other by selecting OFF with the radar mode switch. SLV illuminates when in this mode.

Operation

The system operates in one of two modes: WX display and ground-mapping (GMAP) display.

In the WX display mode, storm intensity levels are displayed in bright color contrasted against a deep black background. Areas of heaviest rainfall appear in lavender/magenta. The third level of rainfall appears in yellow. The areas of least rainfall appear in green.

In the GMAP mode, three different colors are used to display the various ground surfaces. The most reflective targets appear in shades of lavender/magenta. The next level of reflectivity appears in yellow. The least reflective targets appear in shades of cyan.



DETAIL A

Figure 34-27. Allied Signal Enhanced Ground Proximity Warning System

Federal regulations require that receiver-transmitters (like those used in the Primus 880) meet certain minimum requirements in order to be approved for operation in the United States. Approval is granted by the Federal Communications Commission (FCC). The regulation also requires that users of the Primus 880 have an operator license or permit. To qualify as an approved operator, an operator must hold a valid restricted radiotelephone operator permit or higher class license. To obtain this permit, apply at the nearest FCC field office.

The Primus 880 Digital Weather Radar system can be operated in a variety of configurations. At the heart of the system is a combined receiver/transmitter/antenna assembly, on the forward nose bulkhead beneath the radome. The radar display may be shown on the pilot and copilot EFIS, or on a MFD. The display is selected with the EFIS controller and MFD controller.

ALLIED SIGNAL ENHANCED GROUND PROXIMITY WARNING SYSTEM

Description

This section describes the enhanced ground proximity warning system (EGPWS), a terrain awareness and warning system (TAWS) (Figure 34-27).

The EGPWS has terrain alerting and display functions. These functions use the aircraft position, altitude, and a terrain database to predict possible dangers between the aircraft flight path and terrain, and to show hazardous terrain on a display. The EGPWS is interfaced with:

- The radio altimeter
- Air data computers
- AOA
- Attitude heading reference system (AHRS)
- Landing gear position sensors
- Flap position sensors
- Selected DH
- Glideslope navigation receivers

The EGPWS system utilizes information from these systems to calculate the flight path and the possibility of impending danger. Data from the EGPWS system goes to the Integrated Avionics Computer (IC-615). Visual warnings are shown on instrument panel annunciators and on the PFDs and MFD. Voice warning messages are announced on the aural warning system.

Operation of the EGPWS is automatic when the AVIONICS POWER switch is in the ON position and all related systems are valid. The system is operational from altitudes of 50 to 2450 feet (4.57 to 746.76 m) above ground level as sensed by the radio altimeter. The EGPWS has seven operational modes all of which are valid for ground proximity warning.

Mode 1: Excessive Sink Rate—This mode has two envelopes of flight operations: sink rate and pull up envelopes.

The sink rate envelope is measured barometrically and registers in a flight envelope beginning at approximately 5000 feet (1524 m) per minute at 2450 feet (746.76 m) above ground level. It decreases to approximately 1000 feet (304.80 m) per minute at 50 feet (15.24 m) above ground level. If this flight envelope is entered, the aural warning “sink rate” is heard.

The pull up envelope begins at a rate of descent of approximately 7125 feet (2171.70 m) per minute at 2450 feet (746.76 m) above ground level and goes down to approximately 1500 feet (457.20 m) per minute, at slightly below 200 feet (60.96 m) above ground level. When the aircraft enters this flight envelope, a warning, “whoop, whoop” followed by a voice warning, “pull up” is heard. The aural warning continues over the headsets and speaker until it is cleared by a positive pull up out of the dangerous area.

FAA MANDATES TERRAIN AWARENESS WARNING SYSTEM (TAWS)

MARCH 29, 2002 NEW AIRCRAFT

MARCH 29, 2005 EXISTING AIRCRAFT

ALL PART 135 AIRCRAFT TURBINE POWERED 6-9 PASSENGERS MUST BE EQUIPPED WITH A CLASS B TAWS

ALL PART 135 AIRCRAFT TURBINE POWERED 10 OR MORE PASSENGERS MUST BE EQUIPPED WITH A CLASS A TAWS

ALL PART 91 AIRCRAFT TURBINE POWERED 6 OR MORE PASSENGERS MUST BE EQUIPPED WITH A CLASS B TAWS

Mode 2: Excessive Terrain Closure Rate Terrain—Closure rate, during cruise operation (flaps up) is sensed by the radio altimeter. The upper limit is a terrain closure rate of 5100 feet (1554.48 m) per minute at a speed of approximately 0.45 Mach, at 2450 feet (746.76 m) above ground level. Speed and closure rate decrease linearly to near-approach speed and a descent rate of approximately 2000 feet (609.60 m) per minute.

The aural message is: "terrain, terrain", then "pull up" every 0.75 seconds if the aircraft stays in the mode envelope. With the flaps down, the envelope parameters are between approximately 8000 feet (2438.4 m) and 200 feet (60.96 m) above ground level. With the flaps up, the mode does not operate.

Mode 3: Descent After Takeoff—After takeoff, a negative rate-of-climb for a specific altitude loss causes an aural warning of "don't sink". The amount of altitude decrease changes from (minus) -15 feet (-4.57 m) at 100 foot (30.48 m) altitude to (minus) -70 feet (-21.34 m) at 700 feet (213.36 m) above ground level altitude.

Mode 4: Proximity to terrain and aircraft not in landing configuration.

There are three conditions and messages in this mode:

- If the aircraft falls below 500 feet (152.40 m) above ground level at a speed less than 0.35 Mach and the landing gear is not lowered, an aural warning of "too low–gear" is heard again, and again, every 0.75 seconds until the landing gear is lowered, or until the aircraft is flown out of the envelope.
- Between 50 feet (15.24 m) and 1000 feet (304.80 m) above ground level, at a speed between 0.35 and 0.45 Mach, increasing linearly, an aural warning of "too low–terrain" is heard.
- If the aircraft descends below 500 feet (152.40 m) above ground level at a speed of less than 0.35 Mach, when the landing gear is down and the flaps are not in landing position, an aural warning "too low–flaps" is heard.

Put the GPWS NORM/GPWS FLAP OVRD switch in the GPWS FLAP OVRD position to stop the PULL UP - FLAPS aural warning.

Mode 5: Inadvertent Descent Below Glideslope—Aural warnings of "glidescope" are heard and the amber BELOW G/S annunciator illuminates again, and again, if the aircraft descends slightly more than one dot below the instrument landing system glideslope. Go back to the glideslope to stop the soft warning. When the aircraft is more than two dots below the glideslope, between 300 feet (91.44 m) and 150 feet (45.72 m) above ground level, the aural warning becomes loud. The GLIDESLOPE aural warning illuminates is said again and again, louder and faster. The loud warning can be stopped only by a positive "pull up". Push the BELOW G/S annunciator while in the soft warning area to stop the Mode 5 operation. If the aircraft climbs to a radio altitude of above 1000 feet (304.80 m) or descends below 50 feet (15.24 m), Mode 5 resets if cancelled. If the pilot needs to go below the glideslope, Mode 5 operation can be stopped if the BELOW G/S annunciator is pushed at an altitude below 1000 feet (304.80 m): above ground level while still in the soft mode of operation.

Mode 6: Minimums—An audible message, "minimums" is heard twice when the aircraft falls below 1000 feet (304.80 m) above ground level, when the radio altimeter passes through the altitude set (in the radio altimeter DH window). No other system warning is provided. However, the DH annunciator on the pilot attitude director/indicator is activated due to inputs received from the radio altimeter.

Mode 7: Windshear Alerting and Warning—This occurs when a combination of aircraft body angle and vertical displacement occurs, indicating a wind shear situation. When a wind shear condition exists, a siren sounds followed by three aural messages of "wind shear".

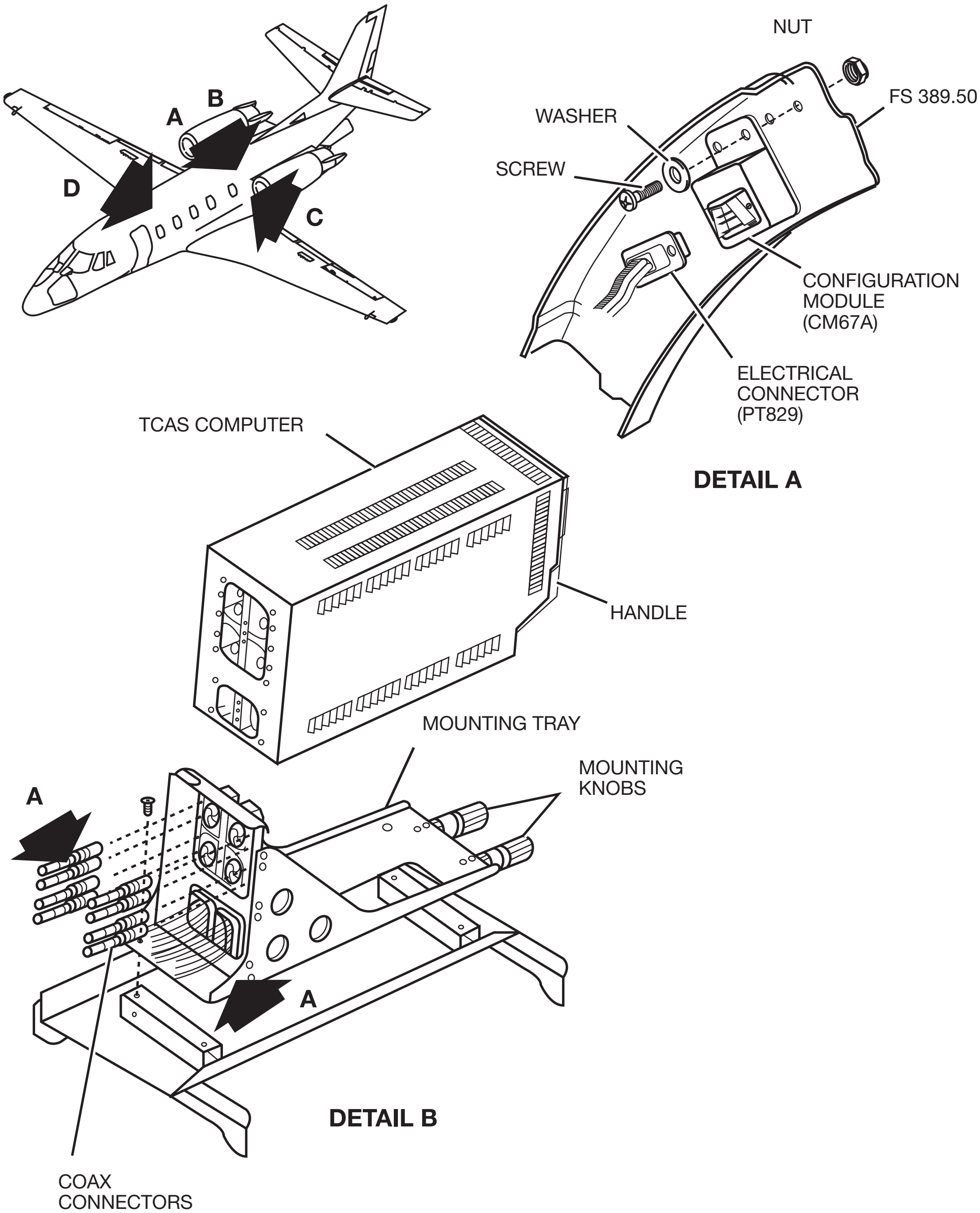


Figure 34-28. Bendix/King TPU-67A Traffic Alert and Collision Avoidance System (TCAS) Aircraft 5001 through 5500

BENDIX/KING TPU-67A TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEM

Description

The Bendix/King TCAS system presents visual traffic advisories to the flight crew on the MFD (Figures 34-28 and 34-29). Every transponder-equipped aircraft within range of the TCAS is asked for its bearing and altitude data. This data establishes a track for collision avoidance. Non-altitude reporting intruders are also detected and tracked by TCAS.

The Bendix/King TPU-67A TCAS II system performs traffic alert functions without support from ATC ground stations.

The system provides safe separation between the aircraft and other aircraft equipped with Mode S, Mode C or Mode A/C transponders.

The TCAS II system monitors the airspace surrounding an aircraft by interrogating the transponder of intruding aircraft.

The interrogation reply enables TCAS II to:

- Compute range between your aircraft and the intruder
- Compute relative bearing to the intruder
- Compute altitude and vertical speed of the intruder, if reporting altitude
- Compute closing rate between the intruder and your aircraft
- Issue a traffic advisory (TA) when closing traffic is in the vicinity
- Issue a resolution advisory (RA) to maintain safe vertical separation
- Track 45 aircraft, display up to 30, and coordinate a resolution advisory for up to three intruders at once

NOTES

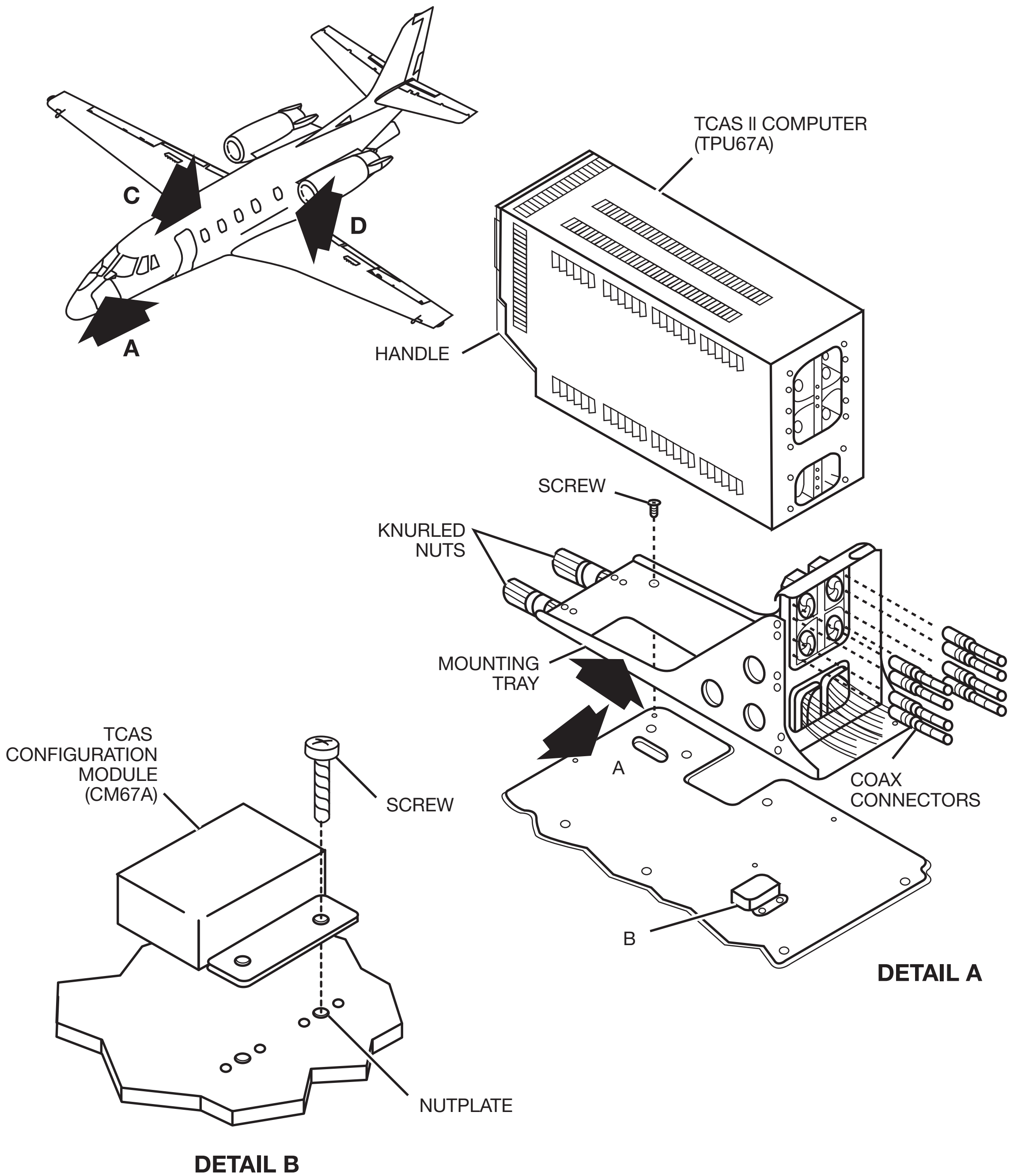


Figure 34-29. Bendix/King TPU-67A Traffic Alert and Collision Avoidance System (TCAS) (Aircraft 5501 and On)

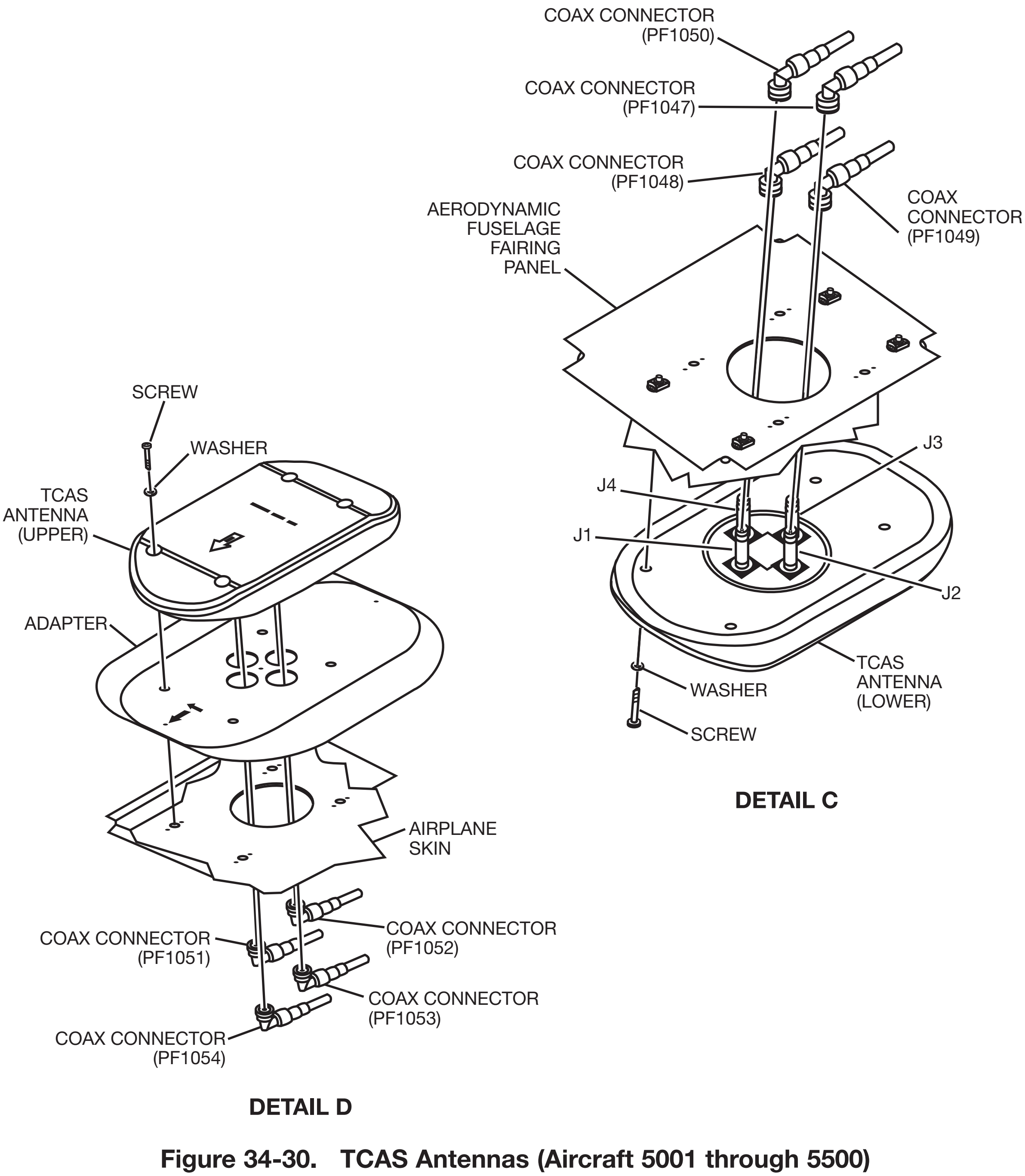


Figure 34-30. TCAS Antennas (Aircraft 5001 through 5500)

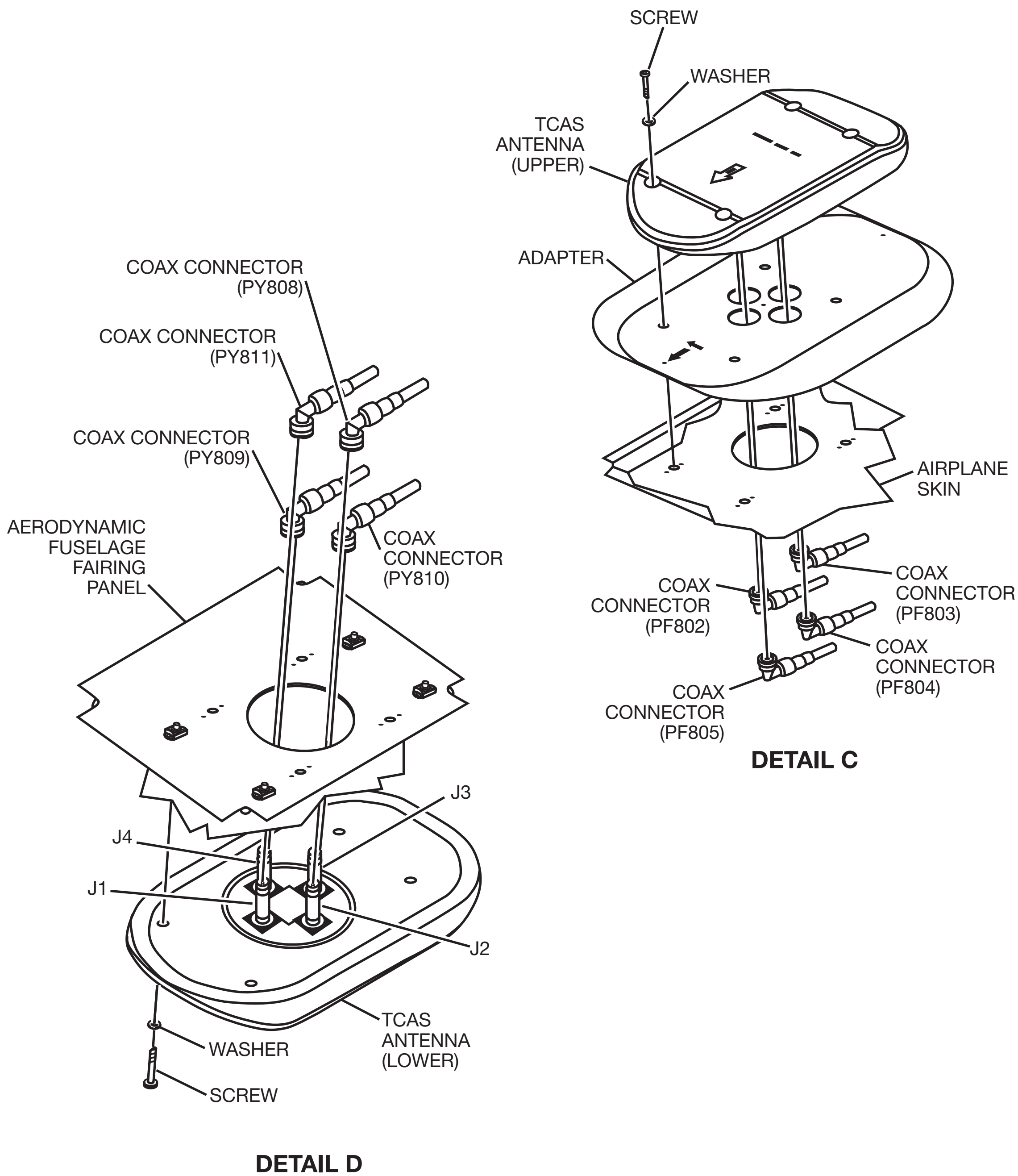


Figure 34-31. TCAS Antennas (Aircraft 5501 and On)

The Citation XL/XLS TCAS system consists of:

- TPU 67A TCAS II processor
- Top and bottom ANT 67A directional antenna (Figure 34-30 and 34-31)
- Honeywell Radio Management Unit (RMU)
- EFIS display system
- ATC Transponder

Operation

The TCAS computer:

- Selects directional antenna beams
- Generates and transfers pulsed 1030MHz surveillance interrogation data to the upper and lower antennas
- Receives 1090 MHz reply data from the TCAS antennas

The TCAS computer examines the reply data and determines the threat potential of intruder aircraft.

The TCAS computer routinely reads and stores the following own aircraft information:

- Aircraft heading (pitch, roll, and radio altimeter, if equipped) inputs. This information, in conjunction with pressure altitude data, allows TCAS to determine its own aircraft position and flight path, which is used during tracking advisory and traffic display computations.
- Pressure altitude is input to the transponder from the aircraft pressure altitude source.
- Pressure altitude is used to determine the aircraft flight level altitude relative to an intruder's reported altitude.
- Aircraft identification is input to the TCAS computer. The computer then broadcasts the information to other TCAS-equipped aircraft in the area for interference limiting.
- Aircraft maximum airspeed is input to the TCAS computer and is used for maximum closing rate projections.
- The TCAS computer performs functions that determine range, bearing, and altitude of intruder aircraft, based on information computed from or contained in the reply messages. The intruder's bearing can only be determined if their replys are received on the directional antenna. Altitude can be determined only if the intruder's transponder reply message is reporting altitude.

Based on information that can be extracted from and computed from the reply, the TCAS computer evaluates potential threat from an intruder by calculating the intruder's closing rate and position. Based on this evaluation, the TCAS computer categorizes the intruder as a nonthreat, proximity, or traffic advisory.

For traffic advisory category aircraft, the TCAS computer outputs a traffic advisory symbol position and alert data illuminates on the EFIS. Aural traffic advisory alert voice messages are also heard via the cockpit audio system.

For proximity and nonthreat aircraft, the TCAS computer outputs proximity or nonthreat traffic symbol position data on the EFIS. Voice alerts are not generated for proximity or nonthreat category aircraft.

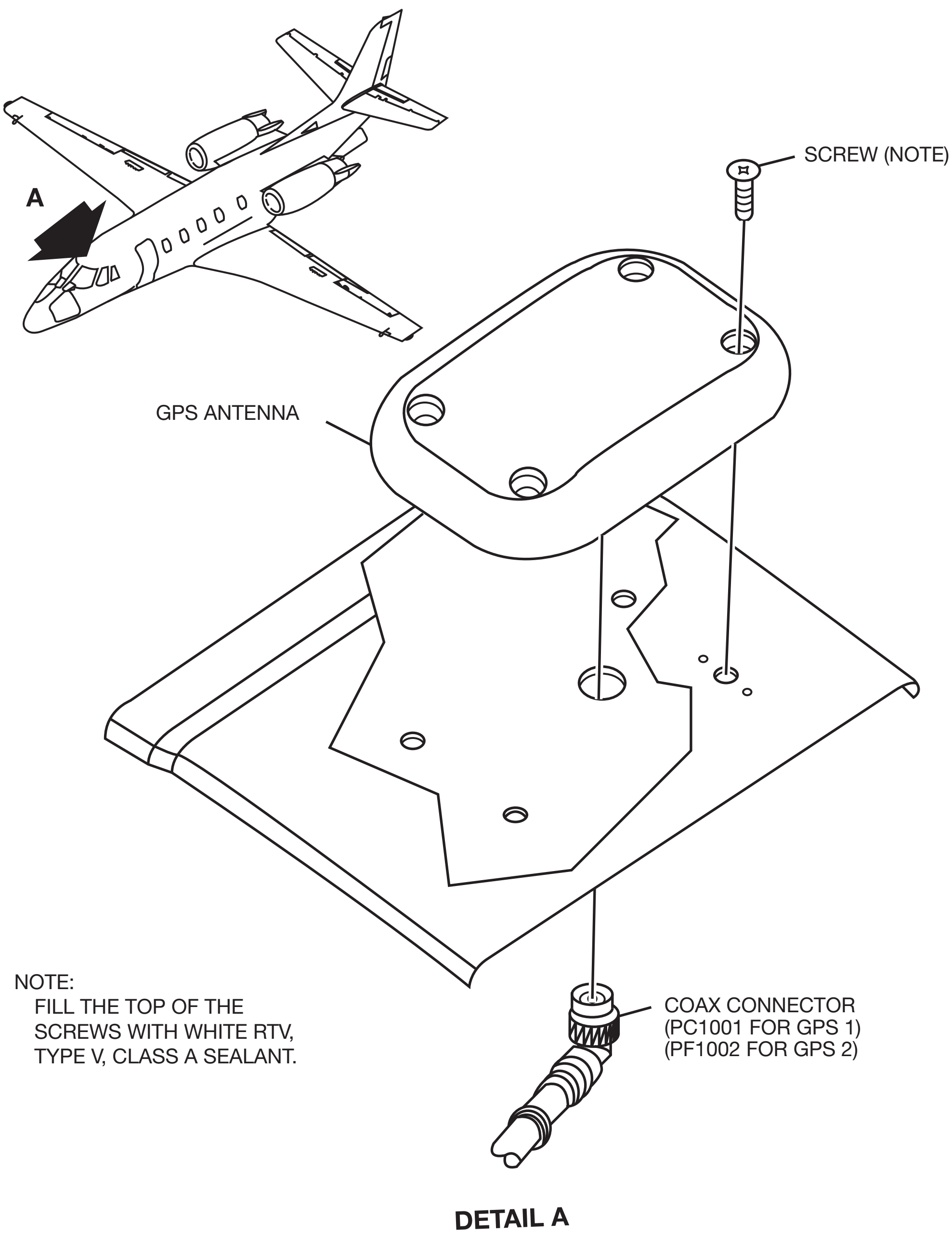


Figure 34-32. GPS Antenna Installation

LOW FREQUENCY NAVIGATION SYSTEMS

Description

The low frequency navigational systems are discussed include ADF navigation systems. The ADF system operating frequency is 100 KHz to 1799.5 KHz.

The automatic direction finder (ADF) is a radio receiver used for continuous automatic determination of bearing, in relationship to a radio station. It also monitors the reception of audio transmissions received from the station.

VERY HIGH FREQUENCY NAVIGATION SYSTEMS

Description

The very high frequency (VHF) navigation system includes the navigation receivers. The VHF system operating frequencies are as follows:

- VOR—108.00 to 117.95 MHz
- Localizer—108.10 to 111.95 MHz
- Glideslope—329.15 to 335.00 MHz
- Marker beacon—75 MHz

The navigation receivers (designated as NAV1 and NAV2) contain the VHF omni-directional range (VOR), localizer, glideslope and marker beacon receivers.

All navigation information is provided to the integrated avionics computer, which processes and formats the information for display on the EFIS.

Navigation receivers are controlled by the radio management unit (RMU). The audio control panels control navigation receiver audio.

ULTRA HIGH FREQUENCY NAVIGATION SYSTEMS

Description

The ultra high frequency (UHF) navigation systems includes the distance measuring equipment (DME) and transponder. These UHF system-operating frequencies are as follows DME: 962 to 1213 MHz, and the Transponder: 1030 and 1090 MHz.

The DME system measures distances electronically and displays them on the DME indicators, and left and right horizontal situation indicators (HSI).

The transponder system provides (response) and receives (interrogation) electronic pulses from the ground station or TCAS equipped aircraft.

GLOBAL POSITIONING SYSTEM

Description

The global positioning system (GPS) uses orbital satellites to determine aircraft position. The GPS receiver is integrated into the UNS-1C flight management system. GPS is controlled and displayed through the flight management system. Antennas for the GPS are on top of the aircraft fuselage (Figure 34-32). The GPS1 antenna is at FS 176.49, LBL 4.30. The GPS2 antenna is at FS 167.17, RBL 4.34, if installed.

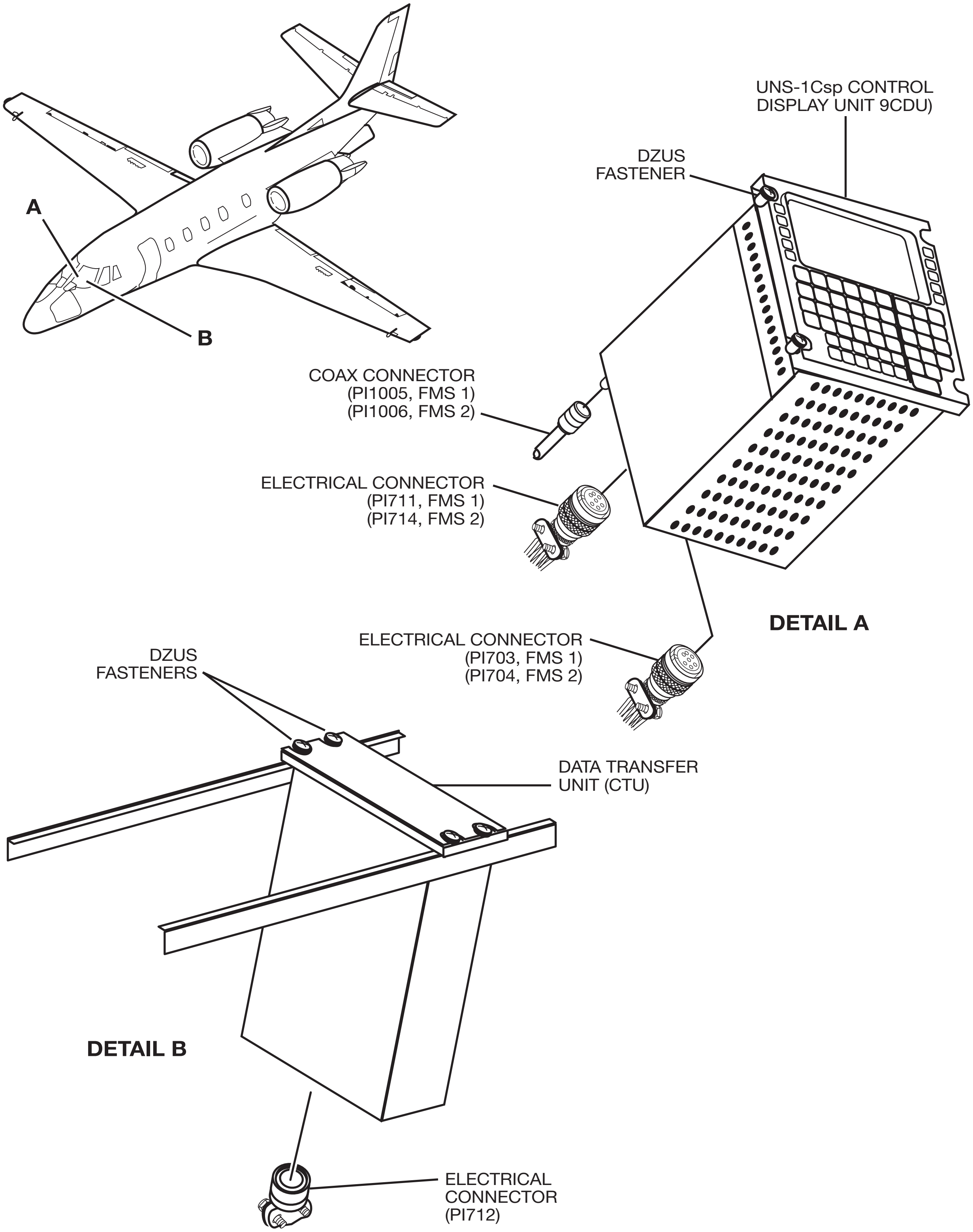


Figure 34-33. Universal UNS-1CSP Flight Management System

UNIVERSAL UNS-1CSP FLIGHT MANAGEMENT SYSTEM

Description

The FMS receives data from various systems, and utilizes this data to determine position, flight path, and aircraft performance. The UNS-1CSP FMS is a fully integrated navigation and flight management system providing the operator with centralized control of the aircraft navigation sensors, computer-based flight planning and fuel management (Figure 34-33).

HONEYWELL FLIGHT MANAGEMENT SYSTEM

Description

The Honeywell NZ-2000 FMS is a comprehensive flight management system, which performs the tasks of lateral and vertical navigation plus performance calculation.

The FMS determines its position from a variety of sensors such as VOR/DME and GPS. The FMS accounts for the characteristics of each sensor in the position determination, using the strongest features of each sensor to compute the aircraft position.

The control display units (CDUs) display systems information to the flight crew and are used by the crew to enter information into the systems.

The integrated avionics computer (IAC) contains several avionics systems while the FMS navigation computer is one system. The FMS computer uses sensor inputs, that come directly into the IAC from other aircraft system sensors, to determine the aircraft position.

The database is also contained within the IAC. The database consists of navigation information data and a custom database used to store pilot defined waypoints and flight plans. Combined with the information from the database and aircraft position from the sensors, the navigation computer provides guidance along a specified flight plan.

The navigation database must be updated every 28 days. The data is entered through a data port on the center pedestal, by means of a data loader.

The custom database never needs updating.

NOTES

XLS+

ELECTROSTATIC DISCHARGE PROTECTION

With the increase of integrated avionics systems, protection against static electricity must be provided to prevent damage to the electronic systems (Figure 34-34). This assures that the information displayed to the crew is not corrupted or inaccurate.

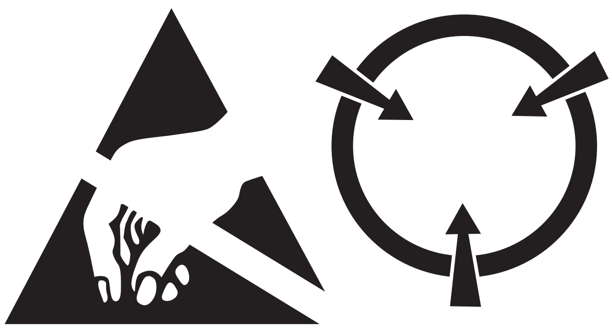


Figure 34-34. Electrostatic Sensitive Device Symbols

Electrostatic discharge is the most common cause of degradation (or destruction) of many electronic components, particularly integrated circuits (ICs), transistors, and semiconductors.

Handle electrosensitive devices (ESDs) with extreme care. A rate/approved wrist strap attached to the same ground potential as the desired circuit card, logic module, or component places a technician at the same potential, eliminating a discharge of electricity (and damage to equipment).

A typical discharge of electrostatic voltage is not seen or heard until it is in excess of 10,000 volts. This means that damage can occur without any indications to the operator until the device or component ceases to function. Most digital electronic components function on 5 VDC. Therefore, 100 volts of induced static electricity is more than enough to damage a component.

ESDs are clearly marked and all necessary precautions should be taken. To find the factory suggested handling procedures of ESD sensitive items refer to Chapter 20—"Standard Practices Airframe" in the *AMM*.

NOTES

34 NAVIGATION

Analog Signals and Digital Signals

An analog circuit is any circuit in which the output voltage and current values are considered significant over a continuous period of time.

Analog = continuous change of state.

A digital circuit is any circuit in which the output currents or voltages are interpreted as having two values.

Digital = two changes of state.

Generally, digital systems offer faster and more precise calculations than analog systems, and require less power to do so.

Analog systems still fill a gap where digital technology may fall short and would not be as practical (e.g., high-power applications).

A discrete signal is a positive switch, it can either be a change from open to short or no voltage to voltage. This type of a discrete signal is commonly used with the squat switch. A discrete signal is considered to be an analog signal even though it is not used to transmit data.

NOTES

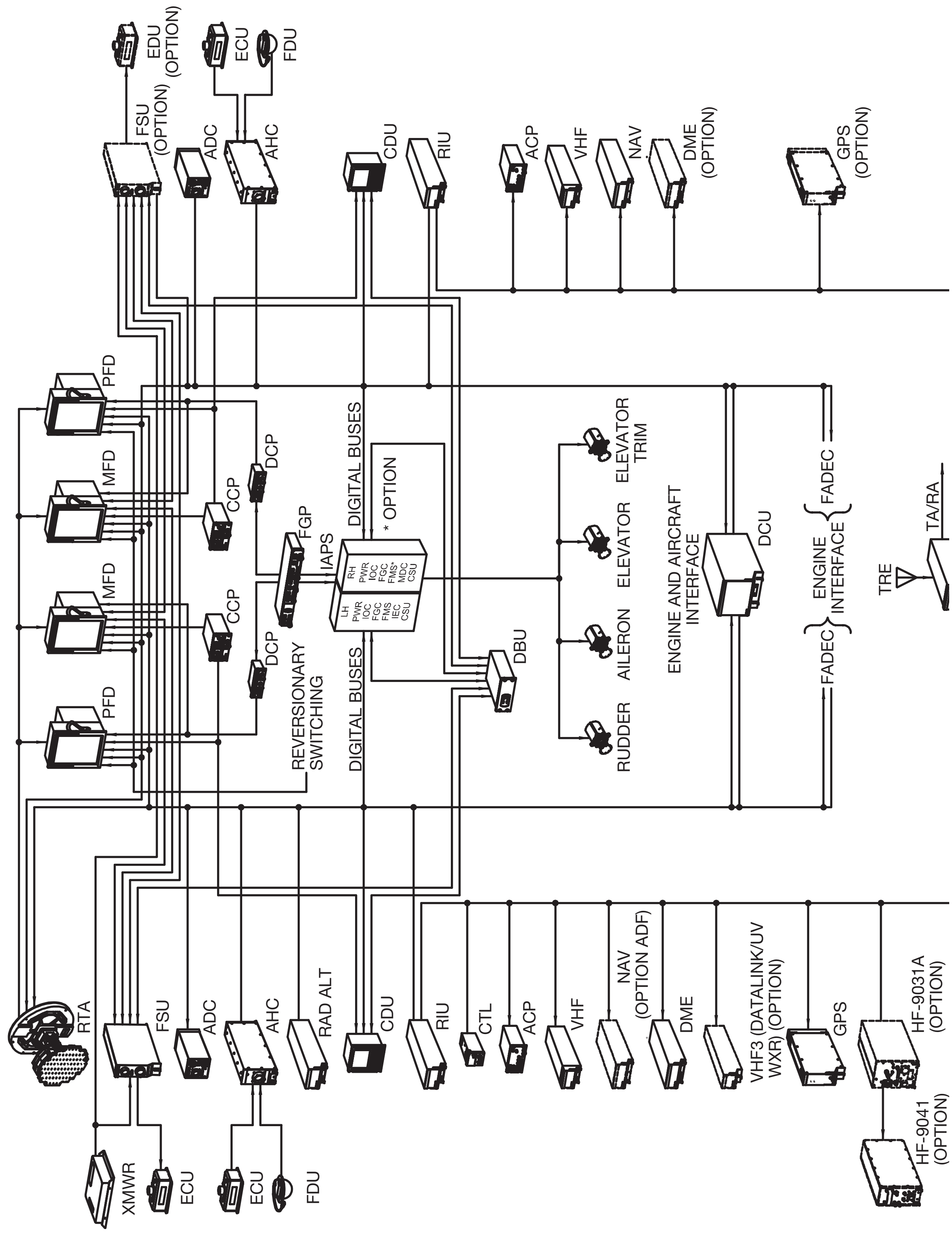


Figure 34-35. Avionics System Diagram

Data Communication

Data communication is a means by which avionics units communicate with each other to carry out programmed functions (Figure 34-35). Modern avionics units are capable of controlling other avionics units, sending and/or receiving information, and making complex decisions.

To allow these digital units to communicate with each other, a communications line has to be connected to each unit. This is known as busing.

There are two methods of data transmission down a data bus or communications line: serial and parallel:

- Serial data transmission-Information is sent down the bus single file. This is the slowest means of data transmission.
- Parallel data transmission-Information is sent side by side. Each bit of information arrives at the same time making this the fastest means.

In the Citation XLS+ aircraft, the most common means of data transmission is inside the avionics boxes using parallel busing. Outside of the avionics boxes, the preferred method is serial communications.

DATA BUSES

DESCRIPTION

The line replaceable units (LRUs) and line replaceable modules (LRMs) in the Collins Pro Line 21 avionics systems communicate with each other using data-bus lines (Figure 34-36). Physically the data buses consist of two wires that are twisted together and shielded from interference.

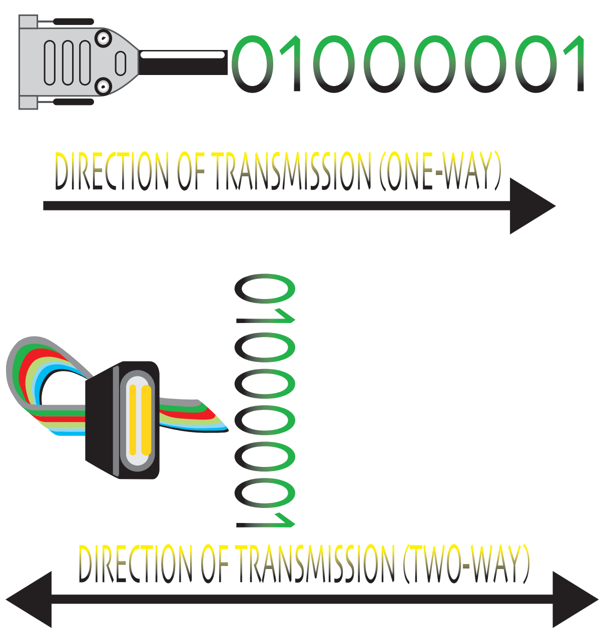


Figure 34-36. Data Bus Communications

High-energy radiated fields (HIRF) and electromagnetic interference (EMI) is unwanted energy interfering with aircraft electronics, causing a disruption of normal operation. Coupling may occur through the aircraft wiring or directly into the equipment itself. This unwanted energy may come in the form of a lightning strike or interference from other transmitters.

Electromagnetic comparability (EMC) is a condition when a signal transmitted by an onboard transmitter (or other electrical/electronic component) affects other system(s) in the aircraft. EMC caused by onboard transmitters occurs due to improper bonding of an access panel or other element common to the skin of the aircraft. The transmitted signal creates skin currents and is reradiated at the point of improper bonding and may bleed back into other system(s).

EMC caused by electronic equipment is a condition when the equipment case or wire shields connected to the equipment is improperly bonded. Without proper bonding the signal is radiated into other equipment or wiring.

Protection against HIRF, EMI, and EMC is accomplished through specific wire routing, proper grounding of equipment, and use of shielded wires (with the shield grounded at both ends of each wire segment).

On most shielded wires, the shield is grounded at both ends of each wire segment. An open shield at one end of a wire segment nullifies the HIRF, EMI, and EMC protection of the wire.

Shield grounding may be at a connector backshell or equipment rack. In some cases, the shields are bonded to a backshell or equipment rack with a band clamp. Some wire bundles are enclosed in a shield overbraid to provide additional HIRF/EMI protection. The shield overbraid is grounded at both ends with a band clamp. Splice shielded wires by using a braided solder sleeve splice.

HIRF, EMI, and EMC protection is designed with consideration for the wire bundle in which wires are routed. Relocation of a wire bundle may cause a change in the common mode impedance between wire conductors and the aircraft fuselage.

CAUTION

Wire bundles should not be rerouted in a manner that changes the relative distance between the aircraft skin (or structure) and the wire bundle.

ARINC 429 Data Bus

An aeronautical radio incorporated (ARINC) 429 bus system is comprised of transmitters and receivers connected by shielded/twisted wire pairs. ARINC 429 is the most common standard data bus used by the Collins systems.

The ARINC 429 consists of a 32-bit, binary coded decimal data word. The first 8 bits make up a label that categorizes the data, (e.g., pitch attitude information). Bits 9 and 10 make up the source destination identifier (SDl), which identifies either the left or right system. Bits 11 through 29 contain pertinent information (e.g., actual pitch attitude, in degrees, of the aircraft). Bits 30 and 31 make up the sign status matrix (SSM) and defines the overall system status. The remaining bit (32) is an odd parity bit used by the avionics input/output processors to ensure data integrity.

Each ARINC 429 transmitter can communicate with up to 20 receivers. Data flows only one way over an ARINC 429 bus. Bidirectional transmission between two LRUs must be accomplished by using two sets of transmitters, receivers, and twisted pair wires.

RS-422 Data Bus

The RS-422 data bus is an electrical specification as defined by the Electronics Industries Association (EIA). It is used where bidirectional communications are needed (e.g., between the displays and display controllers).

The data buses consist of a pair of shielded twisted wires.

RS-232 Data Bus

The RS-232 data bus is an electrical specification as defined by EIA. The RS-232 bus describes any connection between the avionics.

PITOT-STATIC SYSTEM

DESCRIPTION

Pitot/static systems have pitot tubes (heads), static ports, static moisture drain valves, tubing, hoses and fittings. The pitot/static lines use clamps to attach the tubing to the structure. Flex hoses are installed between tubing and the operating components/instruments operational.

Model 560XL has three independent pitot systems and three independent static systems.

The left (pilot) pitot system supplies pitot pressure to the left air data computer. The right (copilot) pitot system supplies pitot pressure for the right air data computer. The standby pitot system supplies pitot pressure for the standby altimeter/airspeed air data unit.

Pilot and copilot pitot tubes are symmetrically located on the nose of aircraft. The

standby pitot tube is located forward of the forward pressure bulkhead, on the nose of the aircraft.

Pitot tubes are separated and positioned to give total pressure input and minimize possibility of total pitot loss due to a birdstrike.

The left (pilot) static system supplies static pressure to left air data computer. The right (copilot) static system gives static pressure to the right air data computer. The standby static system supplies static pressure to the standby altimeter/airspeed air data unit, and cabin delta P indicator (differential pressure indicator).

Each static system has two static ports. One static port is on the left side and one is on the right side of the aircraft.

Interconnecting lines connect the left static port with the right static port and to correspon-ding components/indicators. The pilot static system uses the top static port on the right side and the bottom static port on the left side. The copilot static system uses the bottom static port on the right side and the top static port on the left side. The standby static system uses the aft static port on both sides.

Static ports are located on the lower side of the fuselage, below the pilot and copilot side window.

Moisture accumulation in each pitot system goes to the lowest point in the tube length. The pitot tubes are the lowest point, and moisture automatically drains from the pitot tubes when the aircraft is not in flight. Moisture in the static systems collects at three moisture drain. Two drain valves are located on the right side just forward of the forward pressure bulkhead and one is on the left. All drain valves must be drained at a regular interval.

OPERATION

During flight, air pressure (impact air) collects in the pitot tubes from the forward movement of the aircraft. Pressure in the pitot tubes is transmitted through tubing and hoses to the correct components and instruments. The pitot tubes have heating elements to prevent inlet icing. The pitot heaters are controlled by the Anti-Ice Pitot/Static switch, located on the instrument panel. For the pitot heater electrical circuit, refer to the applicable Wiring Diagram Manual.

The static system sends atmospheric pressure from ambient air outside the aircraft through static ports, tubing and hoses to the correct components/instruments. Static ports have heating elements to prevent icing. Static port heaters are controlled by the pitot/static switch. For the static heater electrical circuit, refer to the applicable Wiring Diagram Manual.

DIAGNOSTICS

Aircraft static systems are required by FAR 91.411 to have altimeter and static system tests. Persons and facilities authorized to perform altimeter and static systems tests are also identified in FAR 91.411. Pressure actuated (barometric pressure) encoding altimeter and static system tests are described in Appendix E of FAR Part 43.

AIR DATA SYSTEM

DESCRIPTION

The Collins ADS-3000 Air Data System has two independent ADC-3000 Air Data Computers (ADC). The pilot and copilot pitot-static systems supply ram and static air pressure to their respective Air Data Computers (ADC). The ADCs calculate their data independently and can supply the air data to PFD 1 or PFD 2. ADC 1 usually supplies its air data to PFD 1 while ADC 2 usually supplies its air data to PFD 2. The air data includes altitude, vertical speed, airspeed, and mach warning. If the usual ADC does not supply the air data to the PFD, the other ADC will automatically supply the air data.

OPERATION

The ADC-3000 is an instrument-grade air data sensor/computer that supplies air data parameters related to air mass.

The ADC 1 receives air pressure inputs from the pilot side pitot and static ports. The ADC 2 receives air pressure inputs from the copilot side pitot and static ports. The data from ADC 1 or ADC 2 can show on PFD 1, MFD 1, MFD 2, and PFD2.

The DCP-3310 Display Control Panel (DCP) has a BARO knob with PUSH STD switch to adjust the barometric pressure value for the ADCs. The pilot and copilot Display Control Panel (DCP) are used to set their respective ADCs.

The ADCs use a processor to calculate the air data from the pitot and static systems. The ADCs calculate and output pressure altitude, baro-corrected altitude, barometric pressure, total pressure, static pressure, vertical speed, airspeed, Mach, maximum airspeed, true airspeed, total air temperature, static air temperature, and ISA delta temperature data. This data is sent to the PFD and MFD for display. The data is also sent to the Attitude Heading Reference System (AHRS), Flight Control System (FCS), transponder, Traffic Collision Avoidance System (TCAS), Enhanced Ground Proximity Warning System (EGPWS), and navigation system.

A standby ADC is located in the copilot console.

GH-3000 ELECTRONIC STANDBY INSTRUMENT SYSTEM

The electronic standby instrument system (ESIS) has an electronic standby attitude indicator (ESAI) that gives pitch, roll, and skid/slip information with an internal three-axis inertial sensor cluster (Figure 34-37).

The electronic standby instrument system has an electronic standby attitude indicator (ESAI), a detachable configuration module (DCM), an air data computer and a magnetometer.

The DCM has sufficient memory to record information specific to the hardware and software configurations for each installation, such as panel angle, navigation interface, aircraft heading calibration, and display format. When the ESAI is removed from the aircraft, the DCM stays attached to the aircraft wiring harness that mates with the ESAI. When the DCM is attached to a new or replaced line replaceable unit (LRU) there is no need to reconfigure the unit.

Air data information is received from the ADC by a dedicated ARINC 429 data bus to the ESAI. The magnetic heading is received from the magnetometer by a dedicated RS-422 data bus from internal inertial sensors. Navigational data is received from the aircraft VHF Navigation system by a RS-422 data bus. All of the system components receive electrical power from the left feed bus in the left J-box. If normal power is unavailable, a standby battery in the left nose baggage compartment is used to supply the unit with electrical power.

Figure 34-37. GH-3000 ESIS

Figure 34-38. EHSI-4000

EHSI-4000 STANDBY INSTRUMENT SYSTEM

The EHSI-4000 is a standby navigation instrument used to show aircraft heading and navigation data and is installed in the center instrument panel (Figure 34-38). The EHSI-4000 has a flat panel active matrix liquid crystal display (AMLCD) and two knobs that turn with push buttons. The pushbuttons let the pilots change the navigational source, instrument mode, and user settings.

The EHSI-4000 usually operates in normal operation mode but also has three navigation modes and two submodes of operation. The Mode pushbutton and knobs on the front of the EHSI-4000 control the navigation and submodes. The EHSI-4000 also uses a data wraparound function that transmits and receives data with the GH-3000 Electronic Standby Instrument System.

COLLINS ALT-4000 RADIO ALTIMETER SYSTEM

Description

The Collins ALT-4000 Radio altimeter system is an integrated part of the Collins Pro Line 21 Avionics Suite. The Collins ALT-4000 radio altimeter system components include one ALT-4000 radio altimeter, one receive radio altimeter antenna, and one transmit radio altimeter antenna.

The Collins ALT-4000 Radio altimeter system gives the pilots altitude data during the approach phase of the flight and provides visual display of the aircraft height above the terrain. The radio altimeter display on the PFD will display a message when the aircraft goes below the preset decision height (DH). The system has one transmit antenna and one receive antenna installed on the bottom of the fuselage.

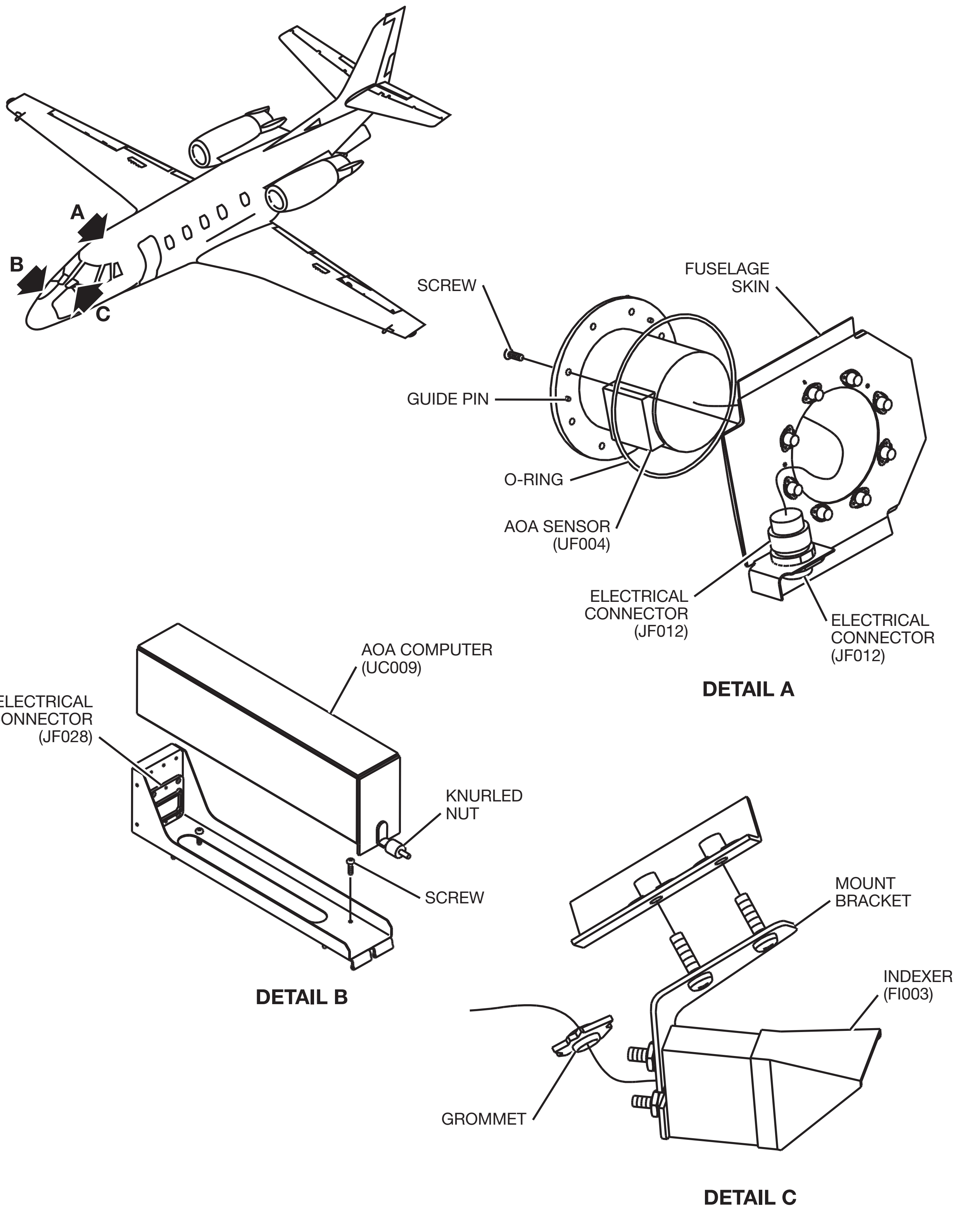


Figure 34-39. AOA System

Operation

The Collins ALT-4000 Radio altimeter system gives the pilots the aircraft height above the terrain from 2500 feet (762 m) to touchdown. The ALT-4000 radio altimeter system has high accuracy and resolution for the pitch guidance to touchdown and the decision height indication. The system is accurate within two feet (0.6 m).

The ALT-4000 radio altimeter is a digital receiver and transmitter in one unit. The ALT-4000 radio altimeter sends RF signals to the transmit radio altimeter antenna and then receives the return RF signal from the receive radio altimeter Antenna. The ALT-4000 radio altimeter processor changes the RF signal data from the transmit and receive radio altimeter antennas into digital data. This data is sent through ARINC 429 digital data buses to other avionics components such as TCAS, TAWS, PFDs, MFDs, and the flight guidance system.

The ALT-4000 radio altimeter is a digital receiver and transmitter in one unit. The radio altimeter sends RF signals to the transmit radio altimeter antenna and then receives the return RF signal from the receive radio altimeter antenna. The radio altimeter processor changes the RF signal data from the transmit and receive radio altimeter antennas into digital data. This data is sent through ARINC 429 digital data buses to other avionics components such as TCAS, TAWS, PFDs, MFDs, and the flight guidance system.

There are two radio altimeter antennas installed on the aircraft. One is for transmit and one is for receive. The transmit radio altimeter antenna sends the RF signals out to the ground below the aircraft. The receive radio altimeter antenna senses the return RF signals from the ground. The transmit radio altimeter antenna has a broad beam antenna system that illuminates a large area of the ground terrain. This give the system accurate data through the usual range of the aircraft pitch and roll axis with a fixed antenna system.

SAFE FLIGHT ANGLE-OF-ATTACK SYSTEM

Description

The angle-of-attack system utilizes the local airflow direction relative to the aircraft pitch direction, and flap position information to calculate a normalized angle of attack.

The angle-of-attack computer is installed in the nose. The computer provides compensating circuitry to adjust for installation variables in the airflow sensor, and logic lights which assist in troubleshooting the system.

The angle of airflow sensor, located on the right side of the fuselage at FS 165, WL 119.81, provides an input to the signal summing unit. The vane is electrically heated to provide anti-icing capability.

Angle-of-Attack Indicator, for Aircraft 5501 and on, the angle-of-attack indication data is shown on the PFDs.

The indexer lights, mounted on the center windshield post, provide a heads up display of angle-of-attack whenever the aircraft is in the air with the gear down and locked. There is a 20-second delay after the gear is down and locked before the indexer is illuminated to prevent operation at takeoff.

Operation

The angle-of-attack system determines the local air flow direction relative to the aircraft pitch axis and using flap position information, computes a normalized angle-of-attack for the angle-of-attack indicator, indexer, and low speed awareness (Figures 34-39 and 34-40). A reading of 1.0 on the indicator signifies stall attitude with 100% lift being used, and 0.0 indicates a 1G condition with no lift being used. A reading of 0.6 is equivalent to 1.3 times stall speed.

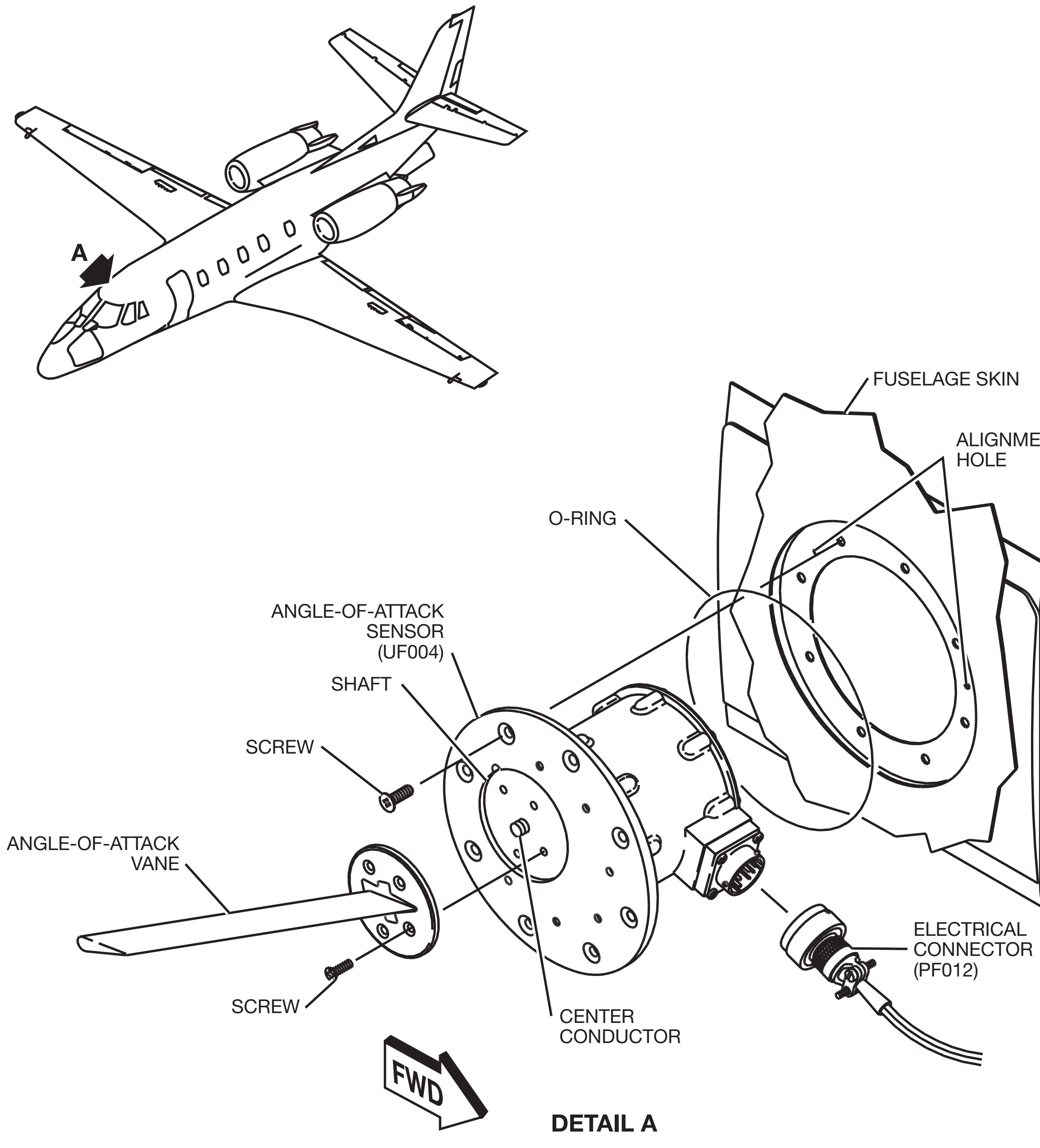


Figure 34-40. AOA Vane Installation

Full illumination of the green split ring on the indexer signifies the angle of attack setting is on the target speed corresponding to a reading of 0.6 on the indicator. Illumination of the red light signifies a speed below target and illumination of the amber light signifies a speed above target. The brightness of the indexer lights gradually increase or decrease proportional to the speed error.

Angle of attack information is also displayed as a low speed awareness bar on the airspeed tape on the primary flight displays. A white bar is the equivalent of 1.3 to 1.2 times stall speed, an amber bar 1.2 to 1.1 times stall speed, and a red bar less than 1.1 times stall speed.

COLLINS ATTITUDE HEADING REFERENCE SYSTEM (AHRS)

Description

AHC-3000 Attitude Heading Computer (AHC)—Solid-state strap-down reference system that supplies angular rate and linear acceleration about the axes of the aircraft. The AHC calculates this data digitally to get 3-axis angle, rate, and acceleration data. This data is supplied through a high-speed ARINC 429 data bus. The AHC has a directional gyro (DG) mode (free gyro) that you can use through discrete inputs.

NOTE

The pilot Attitude Heading Computer (AHC 1) is installed in the left side nose avionics compartment. The copilot Attitude Heading Computer (AHC 2) is installed in the right side nose avionics compartment.

ECU-3000 External Compensation Units (ECU)—Installed near the AHC and contains the specified aircraft AHRS alignment and compass correction parameters. The ECU is reprogrammed during the AHRS leveling procedure or compass swing procedures.

FDU-3000 Flux Detector Units (FDU)—2-axis magnetic sensor that senses the horizontal component of the earths magnetic field. The FDU uses a pendulous sensing element to sense the direction of the Earths magnetic field. Accurate alignment along the aircraft longitudinal axis and compensation adjustment gives the correct heading reference.

CAUTION

Only use nonmagnetic tools and screws for the installation of the FDU to prevent damage.

Operation

The AHRS system provides attitude and heading information to the flightcrew (Figure 34-41). AHRS 1 requires normal DC power to operate and is the normal system for the pilot and AHRS 2 is the normal system for the copilot. The yaw dampener and autopilot require both AHRS systems to be functional with no miscompares in attitude. During ground alignment do not taxi or tow the raircraft, move the flaps, or move the rudder pedals. Any of these can result in a faulty alignment.

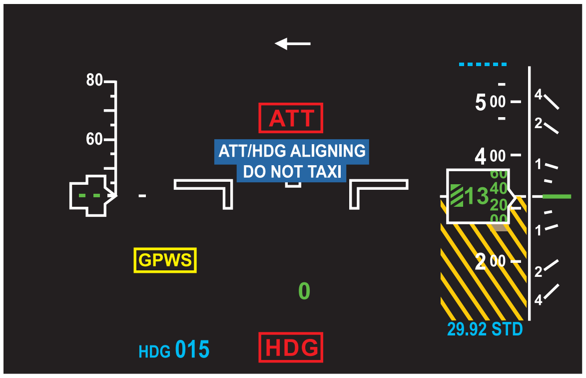


Figure 34-41. AHRS Alignment

The operation of the AH RS is automatic when you apply electrical power to the system and the initialization procedures are complete. After the system completes the power-up procedure, the Attitude Heading Computers (AHC) move to the AHRS mode.

34 NAVIGATION

The system is operating correctly when the compass card shows the correct magnetic heading and the ATT and HDG messages go off of the PFDs.

There is a switch on the left tilt panel to set the AHRS to automatic or manual operation (Figure 34-42). In automatic operation, the AHRS is slaved to the Flux Detector Unit. This is the primary heading mode during usual operations. In manual operation, the AHRS is set to the Directional Gyro with a left/right slew switch. If the MAN position is selected, the pilot can adjust heading with the L or R SLEW switch.

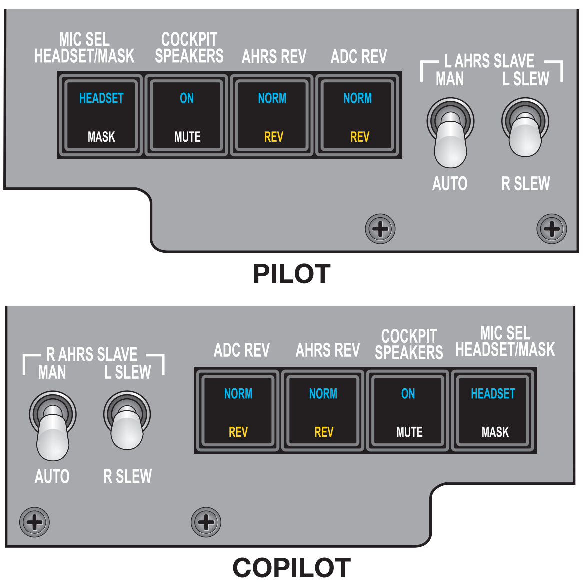


Figure 34-42. AHRS SLAVE switches

COLLINS INTEGRATED AVIONICS PROCESSOR SYSTEM (IAPS)

Description

The Collins Integrated Avionics Processor System (IAPS) is an integrated part of the Collins Pro Line 21 Avionics Suite. The IAPS supplies the integration function necessary to connect the avionics systems that are installed in the aircraft. This section gives the description and operation of the Collins IAPS components.

The Collins IAPS components consists of the following:

- ICC-3111 Integrated Card Cage (1)
- IEC-3001 IAPS Environmental Control Module (1)
- MDC-3110 Maintenance Diagnostic Computer Module (1)
- CSU-3100 Configuration Strapping Units (2)
- DCM-3100 Operation Configuration Modules (2)
- DC-4110 Input/Output Concentrator Modules (2)
- PWR-3000 Power Supply Modules (2)
- FGC-3000 Flight Guidance Computer Modules (2)
- FMC-3000 Flight Management Computer Modules.(2)

The ICC-3111 Integrated Card Cage (ICC) is a mounting tray for the IAPS modules which are installed in the right side nose avionics compartment.

Operation

EC·3001 Environmental Control Module - monitors temperature sensors and controls avionics heaters and cooling fans to keep a constant temperature.

MDC·3110 Maintenance Diagnostic Computer Module—Monitors line replaceable units (LRU) to find system failures, isolate faults to the LRU, and supply a history of fault data.

CSU·3100 Configuration Strapping Units—Supply a matrix of configuration shunts that program the integrated avionics processor system. The DCM-3100 Operation Configuration Modules are attached to their related CSU-3100 Configuration Strapping Unit.

IDC-4110 Input/Output Concentrator Modules—Give a data management function as a central data collection and distribution point for the data bus inputs from the air data computer, attitude/heading computers, flight guidance computers, flight management computer, COMM/NAV/pulse radios, radio altimeter, and engine data concentrator units. The IDC-411 0 Input/Output Concentrator Modules send data to the weather radar system, primary COMM/NAV radios, air data computer, flight control computers, flight management computer, and engine data concentrator units.

PWR-3000 Power supply modules—Supply independent power to the FGC-3000 flight guidance computer modules and IDC-411 0 input/output concentrator modules

FGC·3000 Flight guidance computer modules—Send flight guidance data to the autopilot system and flight displays.

FMC·3000 Flight management computer modules—Monitor sensor inputs to supply flight data for the aircraft.

COLLINS PRO LINE 21 ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

DESCRIPTION

The Collins EFIS (Figure 34-50) components include two AFD-3310 adaptive flight displays used as primary flight displays (PFD), two AFD-3320 adaptive flight displays used as multifunction displays (MFD), two DCP-3310 display control panels (DCP), two CCP-3310 cursor control panels (CCP), and a REVERSION pedestal switch assembly.

The Collins EFIS has four adaptive flight displays (AFD). Two of the AFDs are used as PFD for the pilot and copilot. The pilot PFD (PFD 1) is installed in the left instrument panel. The copilot PFD (PFD 2) is installed in the right instrument panel.

The other two AFDs are used as MFD for the pilot and copilot. The pilot MFD (MFD 1) is installed in the left side of the center instrument panel. The copilot MFD (MFD 2) is installed in the right side of the center instrument panel.

The AFDs are active matrix liquid crystal displays (AMLCD) that are 8 inches by 10 inches in size. The AFDs show data from the following systems:

- Data concentration unit (DCU)
- Full authority digital engine controller (FADEC)
- Attitude heading reference system (AHRS)
- Air data computer (ADC)
- VDR/LDC receivers
- Global positioning system (GPS)
- Traffic alert and collision avoidance system (TCAS)
- Weather radar
- Terrain avoidance warning system (TAWS)

DCP-3310 display control panels (DCP)—Give the pilots direct control functions and display menus to control the PFDs.

CCP-3310 cursor control panels (CCP)—Give the pilots direct control functions and display menus to control the MFDs. The CCPs are installed in the pedestal aft of the throttle quadrant.

REVERSION Pedestal Switch Assembly - gives the pilots a reversion control for the PFDs, MFDs, and Control Display Units (CDU). The REVERSION Pedestal Switch Assembly is installed aft of the throttle quadrant in the pedestal.

Figure 34-43. XLS+ EFIS

OPERATION

AFD-3310 primary flight display—Displays the primary flight data and receives data from the DCP. The two PFDs display data for aircraft altitude, airspeed, angle of attack, vertical speed, attitude, lateral acceleration, flight guidance, lateral and vertical deviation, TCAS, distance, temperature, UTC and elapsed time, COMM frequencies and transmit and ATC Ident, annunciations, compass rose, position, radar, bearing, lightning, and TAWS. The PFDs also show engine data, fuel quantity, and crew alerting system (CAS) messages in reversion mode.

AFD-3320 Multi Function Display—Displays weather, navigation, CAS messages, electronic charts, and other data. The MFDs receive input data from the DCP and the CCP. The two MFDs show data for the engine parameters, Crew Alerting System (CAS) messages, compass rose, map display, radar, TCAS, lightning, TAWS, checklist, maintenance data, ground speed, true airspeed, temperature, graphical weather, electronic charts, enhanced map overlays, and video window.

DCP-3310 Display Control Panel—Primary interface to control the PFD and gives the crew configuration and selection controls and shows the data on the PFDs and MFDs (Figure 34-46). The two DCPs configuration and selection controls include Baro Set, Vspeed/minimums references, PFD format, Nav/Bearing source selection, This overlay, terrain and weather overlays for map, and PFD/MFD terrain/weather range.

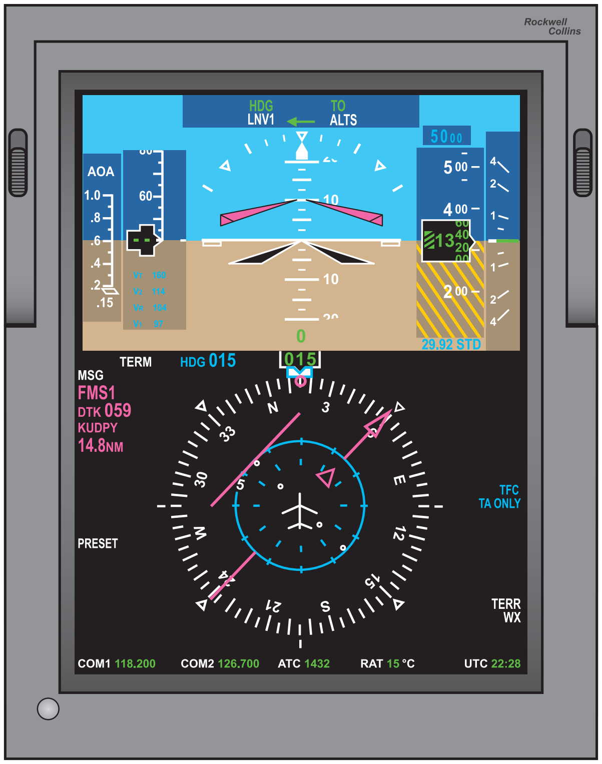


Figure 34-44. Primary Flight Display

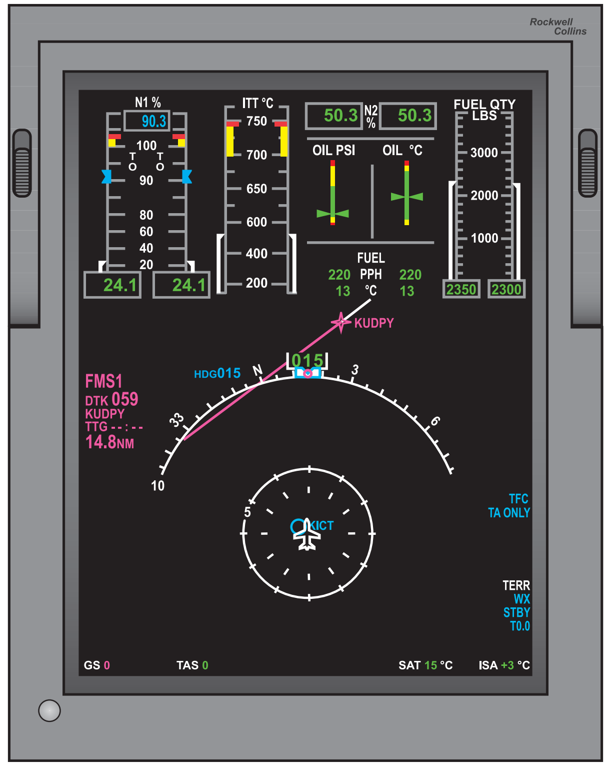


Figure 34-45. Multifunction Display

Figure 34-46. Display Control Panel

CAUTION

Do not pull on the knobs to remove the DCP from the instrument panel. You can cause damage to the DCP if you pull on the knobs.

CCP-3310 Cursor Control Panel—Primary interface to control the MFD and gives the crew main menus and submenus to control the display on the MFDs (Figure 34-47). The two CCPs menu and selection controls include terrain/weather/lightning, traffic, upper and lower menus, quick access keys, CAS messages, cursor, joystick, chart overlay, zoom, database, Nav Data overlay, and checklists.

REVERSION Pedestal Switch Assembly (Figure 34-48)—Provides control to move the display of data from the PFDs to the MFDs and from the MFDs to the PFDs if there is a failure. You can also move the data from CDU 1 to CDU 2 and from CDU 2 to CDU 1 if there is a CDU failure.

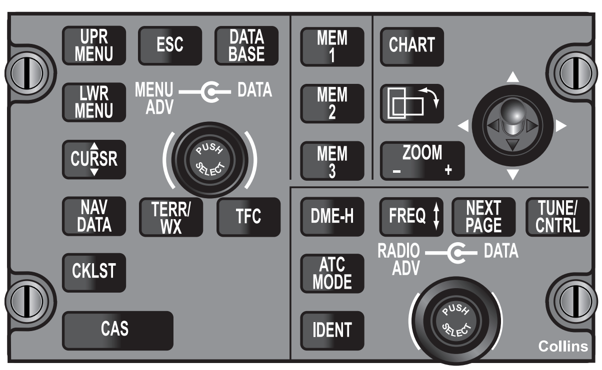


Figure 34-47. Cursor Control Panel

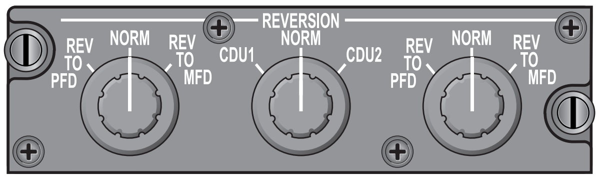


Figure 34-48. Reversion Switches

COLLINS NAV-4000/45000 DUAL VHF NAVIGATION SYSTEM

Description

The Collins Dual VHF Navigation (NAV-4000/4500) System is an integrated part of the Collins Pro Line 21 Avionics Suite. This section gives the description and operation of the Collins NAV-4000/4500 System components.

The Collins NAV-4000/4500 System components include two NAV-4500 navigation receivers, one NAV-4000 navigation receiver (optional), two navigation antennas, one navigation antenna coupler, one ANT-462A ADF antenna (optional), one marker beacon antenna, one marker beacon antenna coupler, one glideslope antenna, one glideslope antenna coupler, two CDU-3000 control display units (CDU), and one CTL-23D standby radio control.

The Collins Dual VHF navigation (NAV-4000/4500) system is installed in a dual system configuration. The NAV-4500 navigation receiver is the standard installation for NAV 1 and NAV 2. The NAV-4000 navigation receiver is an optional installation for NAV 1 only.

The NAV-4500 navigation receiver gives VDR data, localizer and glideslope, warning and flag signals, to/from indications, marker beacon light signals, and audio outputs for VDR, Localizer, and the Marker Beacon. The NAV-4000 navigation receiver gives all the data of the NAV -4500 and also gives ADF bearing data.

The data output from the navigation receivers is shown on the PFDs. NAV-4000 operates as an AM radio receiver in ANT mode or an automatic direction finding system in ADF mode. In ANT mode, the ADF only gives the audio part of the signal. In ADF mode, the ADF gives you audio and bearing data. The Collins dual VHF navigation system has two NAV antennas, an ADF antenna (optional with NAV-4000), a marker beacon antenna, and a glideslope antenna.

The NAV-4000/4500 navigation receivers give integrated ground-based navigation functions. The navigation receivers receive the navigational data from the NAV, ADF, marker beacon, and glideslope antennas. Then they process this data and send it to the pilot and copilot PFD and MFD.

The CDU-3000 Control Display Units (CDU) give the pilots the primary interface for radio frequency tuning with the NAV-4000/4500 System.

The CTL-23D standby radio control gives the pilots an independent control (backup) for radio frequency tuning on NAV 1 and COMM 1. The CTL-23D standby radio control has an LCD display with LED backlighting. In the STBY position the standby radio control operates as a repeater for NAV 1 and COMM In the ON position you cannot use the control display units (CDU) for radio frequency tuning on NAV 1 and COMM 1.

COLLINS TDR-94D DUAL TRANSPONDER SYSTEM

Description

The Collins TDR-94D Dual Transponder System is an integrated part of the Collins Pro Line 21 avionics suite and includes include two TDR-94D transponders, two transponder antennas, and two diversity transponder antennas.

The Collins TDR-94D dual transponder system is a solid-state, airborne, air traffic control (ATC) transponder system installed in a dual configuration. The transponder is a diversity version that can transmit and receive on two antennas and is fully compatible with the TCAS II system installation where antenna diversity is necessary. It operates in Mode A by responding to normal beacon radar interrogations. In Mode C (altitude encoding), the TDR-94D transponder supplies altitude data with the normal reply.

In Mode S, each aircraft is assigned a unique identification code that gives an automatic and unique interrogation and display of a specific aircraft. The unique identification code is related to the aircraft identification (tail) number and is necessary with TCAS operation. Eight different formats of interrogation in Mode S lets the ground controller monitor only the aircraft that are necessary.

The transponder system also has an enhanced surveillance function that sends more aircraft parameters to ATC to help give separation of aircraft. It has two antennas for each transponder. There is a transponder antenna installed on the nose and a diversity transponder antenna installed on top of the fuselage. The CDU-3000 control display units give the pilots the primary interface to control the transponder system.

Operation

The transponder system is an integral part of the Air Traffic Control Radar Beacon System (ATCRBS). It gives the identification of transponder-equipped aircraft on the ATC ground controllers plan position indication (PPI). An altitude encoder gives ATC the aircraft pressure altitude. The TDR-94D can operate in Mode S and it gives a unique aircraft identification code and air-to-air and air-to-ground interrogation replies.

The transponder can also send and receive data link messages that are necessary for ATC automation. The data link lets the TDR-94D Transponder do more ATC and aircraft separation assurance (ASA) functions. In Mode A or Mode C operation, the transponder replies automatically to radar pulses from the ground station. Those replies are decoded for identification and altitude data. In Mode S operation, the transponder replies automatically to radar pulses from the ground station or from other aircraft.

COLLINS DME-4000 DISTANCE MEASURING EQUIPMENT (DME) SYSTEM

Description

The Collins DME-4000 system is a three-channel unit that gives position navigation data. The DME-4000 measures the line-of-sight distance between the aircraft and the selected DME ground stations. The system reads the station identifier and calculates the rate of closure and time to reach the selected station.

Most DME channel assignments are paired with VDR or ILS stations and are selected by setting the related VDR or ILS frequency to the DME. DME frequencies that are not paired with VDR or ILS stations are randomly paired with a group off frequencies (133 to 135 MHz) in the VHF communications band.

Operation

The DME-4000 receives the navigation data through the DME Antennas. The data is then supplied to the pilot and copilot PFD and MFD. The CDU-3000s give the pilots the primary interface to control the DME-4000 System.

COLLINS FMS-3000 FLIGHT MANAGEMENT SYSTEM (FMS)

Description

The Collins FMS-3000 components include two CDU-3000 Control Display Units (CDU), two FMC-3000 Flight Management Computer (FMC) modules, two GPS- 4000S Global Positioning System (GPS) receivers, a GPS1/XM antenna, and a GPS 2 antenna.

The FMS-3000 controls the EFIS displays, FMS, radio tuning, and other aircraft functions. The FMS is a computer module in the IAPS that controls the flight data for use by the FMS. The FMC monitors sensor inputs to constantly find the flight parameters for the aircraft.

The CDU-3000 gives the pilots the interface with the FMS-3000 (Figure 34-49). The CDU has a color display to show the FMS related data and function modes and has a set of line select keys (LSK) and a set of function keys for the pilot input. The LSK are for mode selections and copying or transferring the displayed data. The function keys are for selections of the FMS functions and display modes. The CDU also has a full alphanumeric keypad for entering data.

The GPS-4000S is a 12-channel receiver that uses a minimum of four GPS satellites to find a position solution. With sufficient satellites in view, the GPS system will give Receiver Autonomous Integrity Monitoring (RAIM) for a nonprecision approach.

Figure 34-49. CDU-3000

Operation

The FMS-3000 is a multi-sensor navigation, performance, and flight planning system. The FMS gives lateral and vertical navigation for the enroute, terminal, and non-precision approach airspace through data from sensors. The sensors include Global Navigation Satellite System (GNSS), Distance Measuring Equipment (DME), DMENOR, and Inertial Reference System (IRS).

The FMS provides the data that follows: present position (latitude-longitude); desired track; cross-track deviation; time, distance, and bearing to the next waypoint; to-from; waypoint identifiers; waypoint ETA and ETE; total wind, crosswind, and headwind speeds; track angle; groundspeed; magnetic variation; vertical deviation; and time, distance, elevation, and ident of destination. The FMS data is shown on the CDU and EFIS.

The CDUs give the primary interface for the functions of the FMS-3000. The CDU has a color display to show the FMS related data and function modes. The line select keys around the display select the mode and copy or transfer the displayed data. The CDU has a set of function keys to directly select many of the FMS functions and display modes.

The FMC-3000 modules continuously monitor sensor inputs to supply flight data for the aircraft. The inputs include the GPS, DME, and AHRS.

The GPS-4000S receivers receive position data from satellites that orbit the earth. The receivers are a 12-channel unit that supplies the position data to the FMS and have Receiver Autonomous Integrity Monitoring (RAIM) for nonprecision approaches.

The GPS1/XM antenna receives position data from satellites that orbit the earth and transmits that data to the GPS 1 receiver. The GPS 2 antenna receives position data and transmits that data to the GPS 2 receiver.

COLLINS TTR-4000 TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEMS (TCAS II)

Description

The Collins TTR-4000 TCAS II is a traffic alert and collision avoidance system that monitors a radius of approximately 14 nautical miles around the aircraft. The system monitors the transponders of intruder aircraft to find if they are near your aircraft.

The TTR-4000 calculates range, differential altitude, bearing, and the closure rate of other aircraft that have a transponder. The traffic data is shown on the pilot and copilot PFDs and MFDs. You can hear the voice messages through the audio system.

Operation

The TTR-4000 gives two types of advisories. The type of advisory received is related to the distance to other aircraft. The TCAS II can give a Traffic Advisory (TA) or Resolution Advisory (RA). A TA is given to tell the pilots of a possible accident with an aircraft. A TA gives the pilots time to visually see the other aircraft and move out of the way. An RA is given to tell the pilots of a possible accident with an aircraft and which way to move the aircraft (climb or descend).

TCAS II also shows Proximity Traffic (PA) and Other Traffic (OT). PA and OT are not immediately a threat to your aircraft.

The system has two directional antennas. There is one antenna on the top of the fuselage and one antenna on the bottom of the fuselage. The antennas have four passive antenna components for direction.

HONEWELL MARK V ENHANCED GROUND PROXIMITY WARNING SYSTEM (EGPWS)

Description

The Honeywell Mark V EGPWS is a terrain awareness and warning system with terrain alerting and display functions to give the pilots aural and visual warnings if the flight path could cause an accident with the terrain.

The EGPWS uses aircraft inputs that include geographic position, altitude, airspeed, and glideslope deviation. These inputs are used with internal terrain, obstacle, and airport databases to predict a possible accident between the aircraft path and obstacles or terrain. If a possible obstacle or terrain accident is detected, the EGPWS will give a visual and/or audio caution or warning message. The EGPWS also gives you a message for too much glideslope deviation, too low altitude with flaps, gear not in landing configuration, and severe windshear.

Operation

The operation of the EGPWS is automatic when the AVIONICS switch is in the ON position and all related systems operate correctly. The EGPWS has the modes that follow:

Mode 1—Excessive Descent Rate

Mode 2—Excessive Terrain Closure Rate

Mode 3—Altitude Loss After Takeoff

Mode 4—Unsafe Terrain Clearance

Mode 5—Excessive Deviation Below Glideslope

Mode 6—Advisory Callouts

Mode 7—Windshear Caution and/or Warning Messages for Windshear

The Mark V EGPWS also has an internal database to give you more situational awareness for safety. The EGPWS internal database has the four (4) subset databases that follow:

- A worldwide terrain database of different degrees of resolution
- An obstacle database with obstacles that are 100 feet (30.5 m) high or higher in North America, parts of Europe, and parts of the Caribbean.
- A worldwide airport database with data on runways that are 3500 feet (1067 m) long or longer.
- An envelope modulation database to support the envelope modulation function.

The EGPWS can also have the Runway Awareness Advisory System (RAAS) function. RAAS uses GPS position data and the EGPWS database to give aural messages to the pilots. RAAS gives the pilots increased situational awareness during ground operations and on an approach to landing.

COLLINS RTA-852 WEATHER RADAR SYSTEM

Description

The Collins RTA-852 weather radar system has an RTA-852 Receiver/Transmitter Antenna (RTA) (Figure 34-50). The Display Control Panels (DCP) in the cockpit control the RTA. The radar data is shown on the PFDs and MFDs. Inputs to the RTA include pitch and roll data from the AHRS and radar control data from the PFDs.

Figure 34-50. RTA-852

The RTA is attached to the radar truss support assembly aft of the nose radome. The RTA is a solid-state 12-inch antenna with a 300-nautical-mile range. The RTA has 120 degrees of scan, +15 degrees and -15 degrees of tilt, and 27 degrees per second scan rate.

Operation

The RTA-852 senses precipitation along the flight path and ahead of the aircraft. The DCPs give the pilots the menu controls to select the radar modes. The radar modes include test, standby, weather, weather and turbulence, turbulence ground map, and ground clutter

suppression. The display can be set to show a maximum of a 300-nautical-mile range.

The radar system operates on a nominal output of 25 watts. Scan and tilt control circuits cause the motor to move the antenna horizontally and vertically. The transmitter sends X-band pulse signals to the antenna. When the transmitted signal finds precipitation, part of the signal reflects back to the antenna. When the receiver gets this signal return, it sends the data to the IAPS. The IAPS converts the data into a video format to be shown on the PFDs and MFDs.

The weather radar display is shown in four colors (green, yellow, red, and magenta). The four colors show an increasing intensity of rainfall. Magenta shows areas of very heavy rainfall at rates of 2 inches per hour or more. In addition, the Path Attenuation Compensation (PAC) Alert shows areas of unknown rainfall rates because of signal attenuation caused by areas of precipitation interference.

L3 WX-1000E STORMSCOPE SYSTEM

Description

The L3 WX-1000E Stormscope system is an airborne thunderstorm-mapping system. The system shows electrical discharges (lightning) in a thunderstorm on a map for 360 degrees around the aircraft at a maximum distance of 200 nautical miles. The Stormscope is a passive system that receives the electromagnetic and electrostatic signals with a Stormscope antenna.

The Stormscope antenna detects intra-cloud, inter-cloud, and cloud-to-ground electrical discharges in a radius of 200 nautical miles from the aircraft. The Stormscope antenna sends the discharge signals to the Stormscope processor which digitizes, analyzes, and changes the discharge signals into range and bearing data. This data is stored in the storm buffer and the results are shown on the EFIS displays as lightning bolt icons at the related locations.

The WX-1000E Stormscope Processor, installed in the upper tailcone, receives electromagnetic and electrostatic signals through the Stormscope antenna. The Stormscope antenna is installed on top of the aft cabin. The Stormscope Processor changes these signals and sends them through an ARINC 429 bus to the EFIS displays. During transmit on COMM 1, COMM 2, COMM 3, or HF, the Stormscope Processor is inhibited from receiving a signal to prevent false lightning indications.

NOTES

CHAPTER 35
OXYGEN

CONTENTS

ILLUSTRATIONS

CHAPTER 35
OXYGEN

INTRODUCTION

This chapter describes the oxygen system found on the Citation 560 XL/XLS/XLS+ aircraft. In addition to system descriptions, emphasis has been given to maintenance and servicing precautions, along with functional testing. References for this chapter and further specific information can be found in Chapters 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 35—"Oxygen," of the *Aircraft Maintenance Manual (AMM)*.

GENERAL

The oxygen system consists of the crew oxygen system and the passenger oxygen system. Oxygen is available to the crew at all times and is available to the passengers either automatically (above a predetermined altitude) or manually (at any altitude by a cockpit control). The oxygen system primarily provides emergency oxygen.

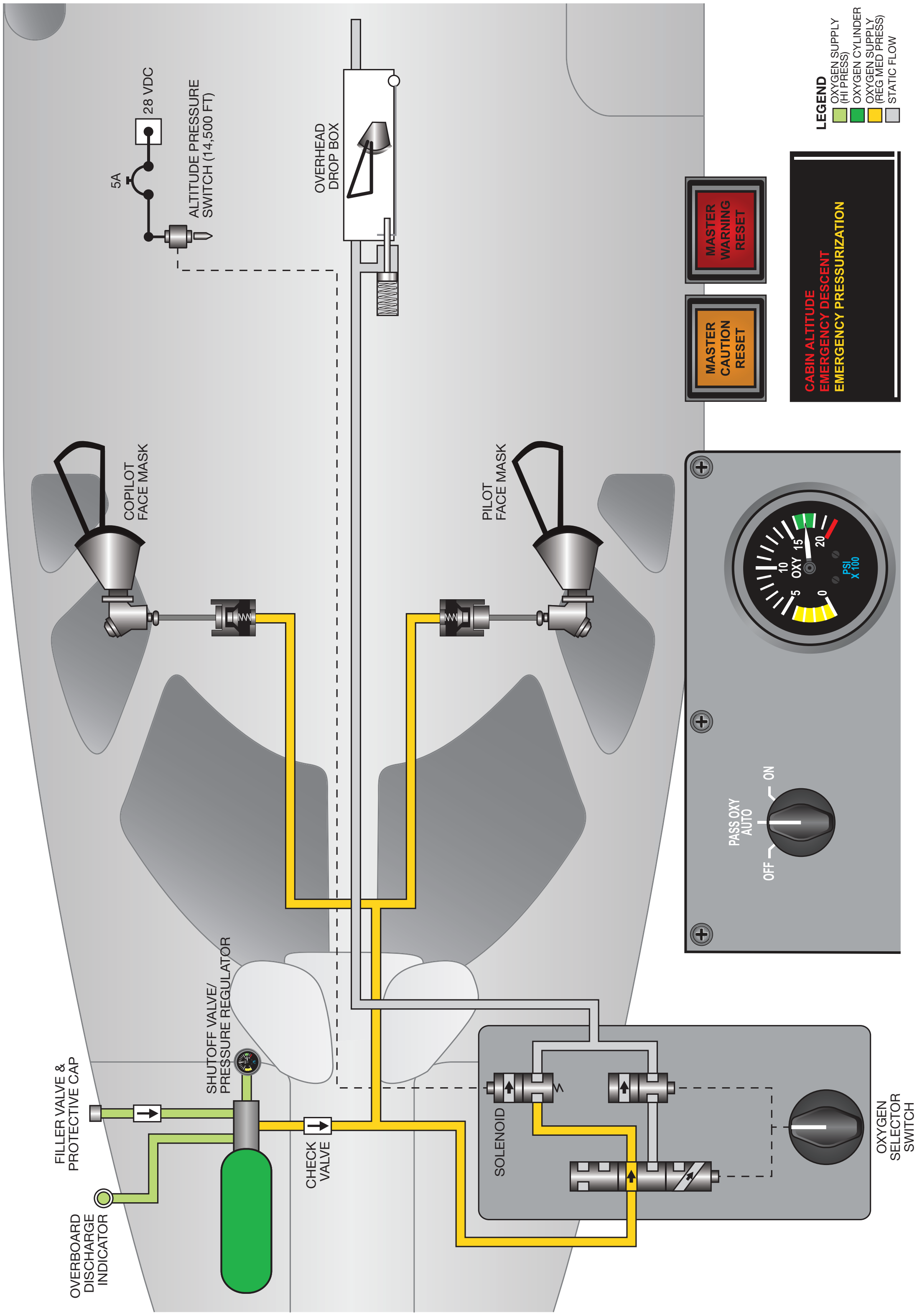


Figure 35-1. Oxygen Simplified

35 OXYGEN

OXYGEN SYSTEM

DESCRIPTION

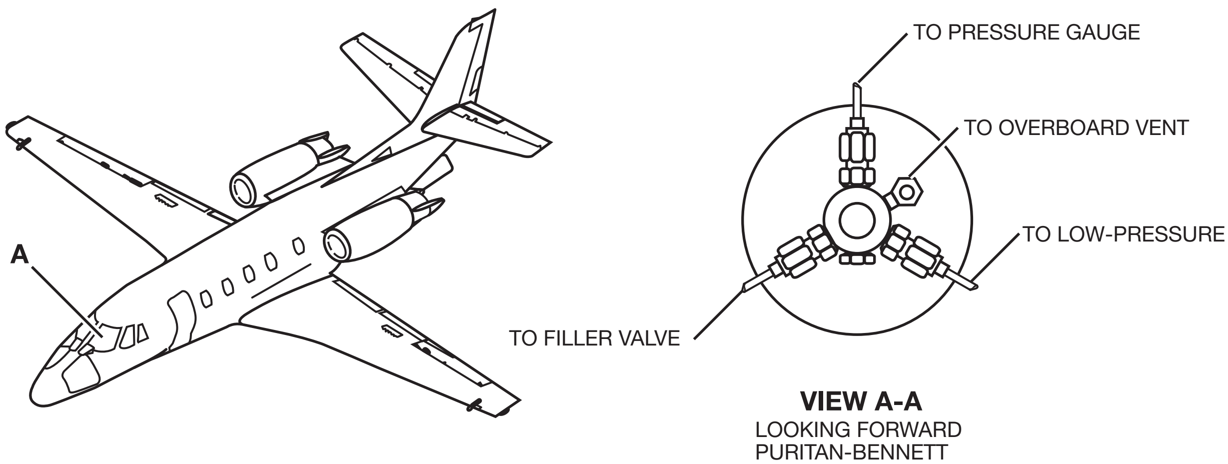
The oxygen system consists of (Figure 35-1):

- Pilot and copilot masks
- Dual passenger oxygen dropout boxes
- Altitude sensing pressure switch
- Composite oxygen bottle with a pressure regulator
  - 50 cubic foot—SNs 5001 through 5619
  - 77 cubic foot—SNs 5620 and Subsequent
- Oxygen pressure gauge
- Overboard pressure relief valve
- Oxygen filler valve
- Passenger oxygen control valve
- Necessary plumbing

Oxygen system flow schematics are provided in the crew and passenger oxygen system sections.

The oxygen system has a high-pressure and low-pressure side. The oxygen is stored in the high-pressure side in a composite 50 or 77 cubic foot bottle just to the right of the wheelwell in the nose compartment. The low-pressure side is controlled by a pressure regulator that attaches directly to the oxygen cylinder. The regulator is turned ON or OFF with the attached control handle. With the handle turned to the OFF position, the regulator functions as a shutoff valve while venting the low-pressure side of the system internally through the regulator. With the handle turned to the ON position and safety wired, the regulator provides a constant 70 ± 10 psi (482 ± 69 kPa) pressure to the low-pressure side of the system. The low-pressure side supplies oxygen to the crew outlets and to oxygen dropout boxes in the cabin.

The low-pressure side of the system of the system is split into two subsystems: one for the cabin and one for the cockpit. A passenger oxygen control valve separates the two halves of the system. Oxygen is always available to the crew through the crew distribution lines, but is blocked from entering the passenger system by the passenger oxygen control valve. The passenger oxygen control valve is a three-way, manual/solenoid-operated valve that has three positions: OFF, AUTO, and ON. Typically the valve is left in the AUTO position. With the valve in the AUTO position, when the cabin altitude increases to a pressure altitude of 14,500 ± 500 feet (4420 ± 152 m), the pressure altitude switch applies electrical power to the solenoid of the passenger oxygen control valve, opening the passenger oxygen control valve. When the passenger oxygen control valve opens, it allows 70 ± 10 psi (482 ± 69 kPa) of pressure to flow into the cabin oxygen system. This pressure is sufficient to deploy the doors on the oxygen dropout boxes and drop the passenger masks. In the event of a failure or at the discretion of the crew, the passenger system can be manually actuated by turning the passenger oxygen control valve to the ON position. The ON position manually opens the passenger oxygen control valve, deploying the doors on the oxygen dropout boxes and releasing the passenger masks. After deployment of the passenger masks, oxygen flow to the passenger is initiated by pulling the lanyard cord attached to the passenger mask, which in turn pulls the pintle pin. The control of oxygen flow into the passenger masks is achieved by a precision orifice between the supply line and the mask. When the passenger oxygen control valve is positioned to OFF, only the crew's oxygen system is operational and no oxygen flows to the cabin and the passenger masks.



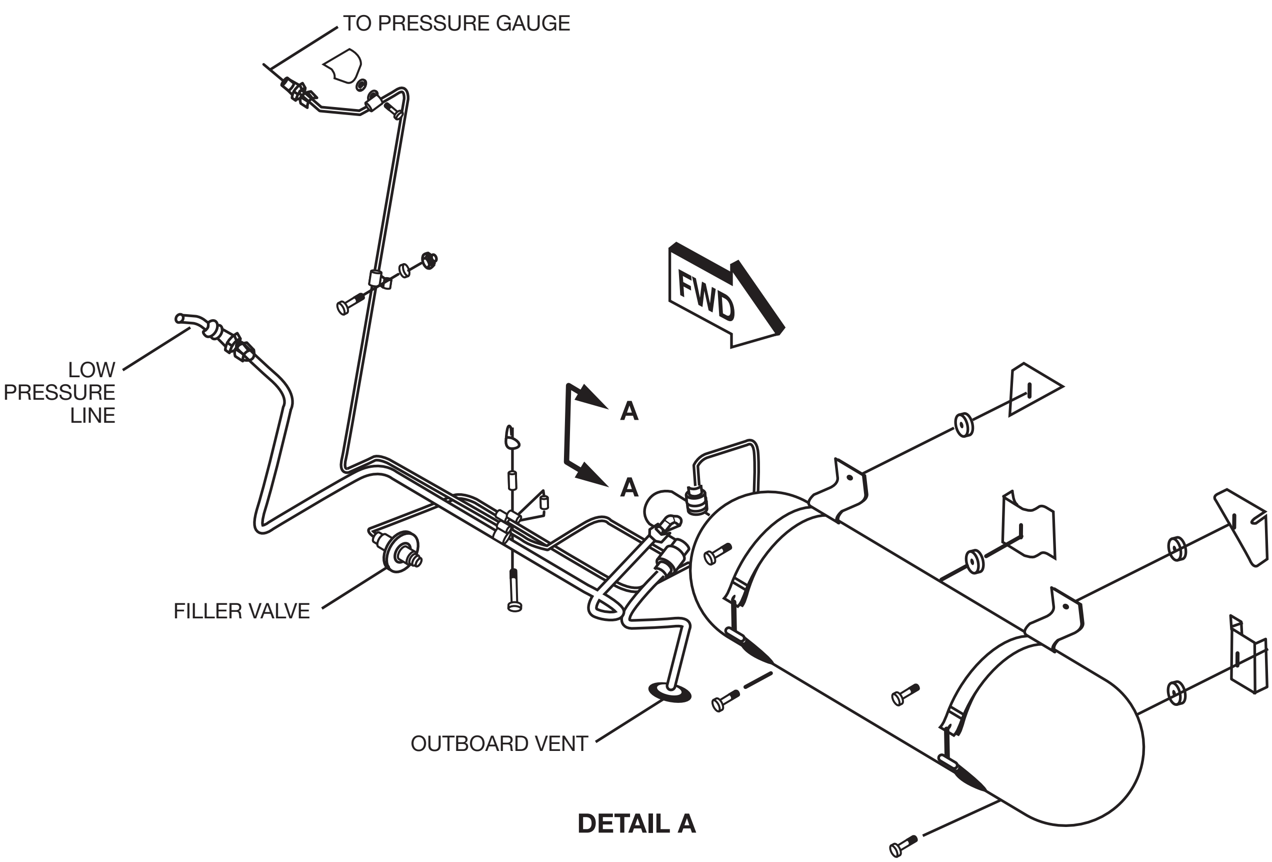


Figure 35-2. Oxygen Cylinder and Pressure Regulator Assembly

COMPONENTS

Oxygen Cylinder and Pressure Regulator Assembly

A single-oxygen cylinder and pressure-regulator assembly serves the entire airplane as a reservoir for breathing oxygen. It also reduces and regulates oxygen pressure to the various oxygen outlet assemblies (Figure 35-2).

The 50 or 77 cubic foot oxygen cylinder is in the nose section of the airplane (at FS 77.68 and RBL 10.25). It is a composite construction consisting of a thin aluminum shell which is tightly wrapped with Kevlar impregnated with epoxy resin.

The pressure regulator is on the oxygen cylinder. The regulator may be turned ON or OFF with its attached control handle. When the handle is turned to the OFF position, the regulator functions as a shutoff valve. With the regulator provides 70 ± 10 psi (482 ± 69 kPa) of pressure to the crew outlets and to the passenger oxygen control valve. The regulator contains separate ports for:

- Servicing (charging) the bottle
- Monitoring the bottle pressure with an instrument panel oxygen pressure gauge
- An overboard discharge line in the event of over-pressurizing the bottle
- A regulated low-pressure oxygen line

The regulator incorporates over-pressure protection by using a disc which ruptures if the pressure in the cylinder exceeds approximately 2600 psig (17,927 kPa). The rupture disc vents to an overboard port at FS 81.00 and RBL 10.25. The overboard port is covered with a green indicator. If the green indicator (disc) is missing, it indicates that the oxygen cylinder has been discharged and must be reinspected.

CAUTION

Oxygen cylinders and pressure regulators are furnished as assemblies by Cessna Parts Distribution. Attempting to remove, repair, or reinstall oxygen pressure regulators in the field provides opportunity for contaminants to enter the system. Faulty pressure regulators or pressure regulators which are otherwise in need of disassembly, must be exchanged for replacement oxygen cylinder and pressure regulator assemblies through Cessna Parts Distribution. The oxygen cylinder and pressure regulator assembly shall be disassembled, repaired, inspected, cleaned, hydrostatically tested, reassembled, and serviced by manufacturer or other FAA approved facility.

CAUTION

Connecting the oxygen pressure gauge line to the check valve unseats the check valve core, releasing high-pressure oxygen into the line. Ensure the oxygen pressure gauge line and oxygen pressure gauge are properly installed prior to connecting the oxygen pressure gauge line to the pressure regulator check valve.

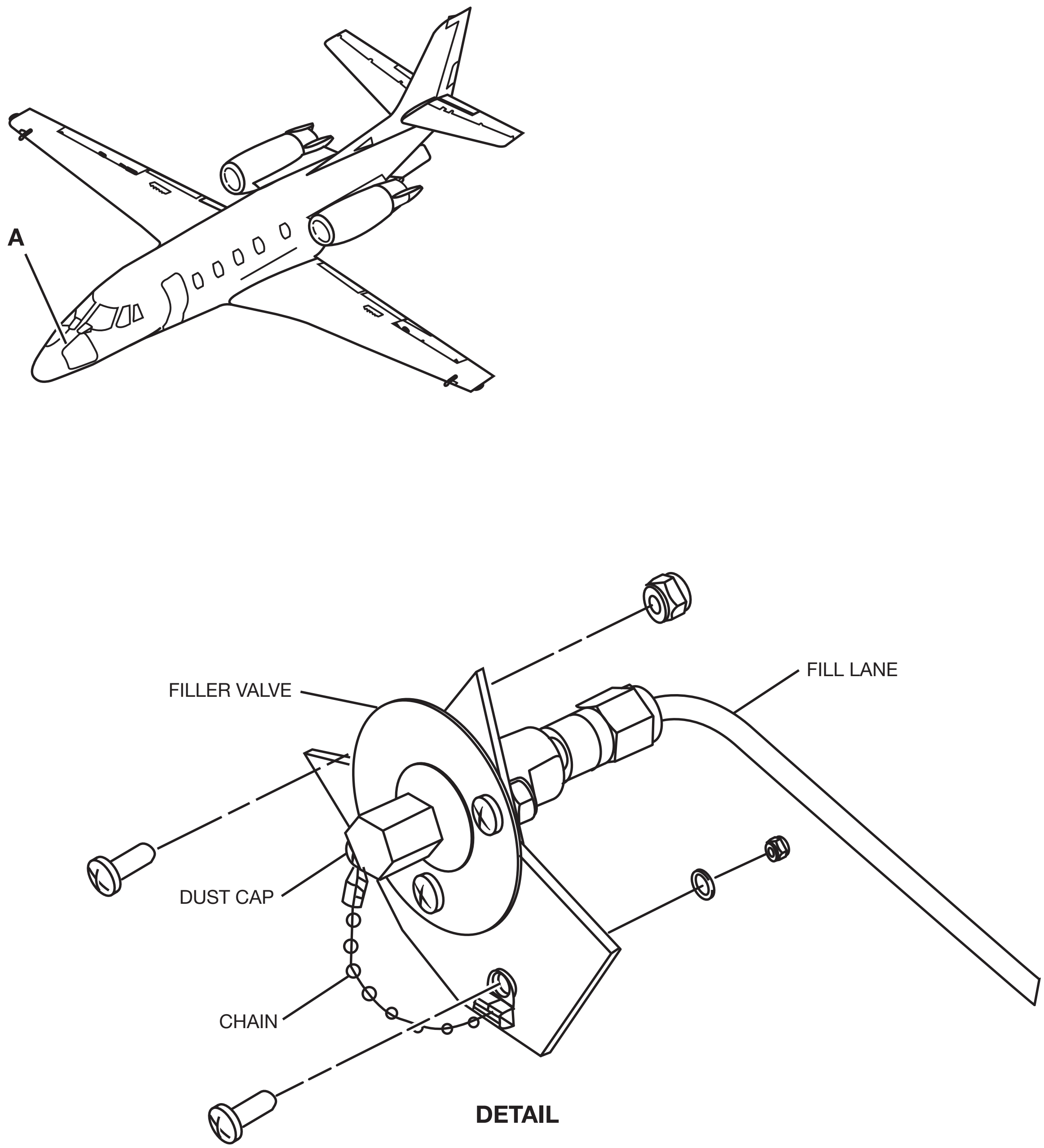
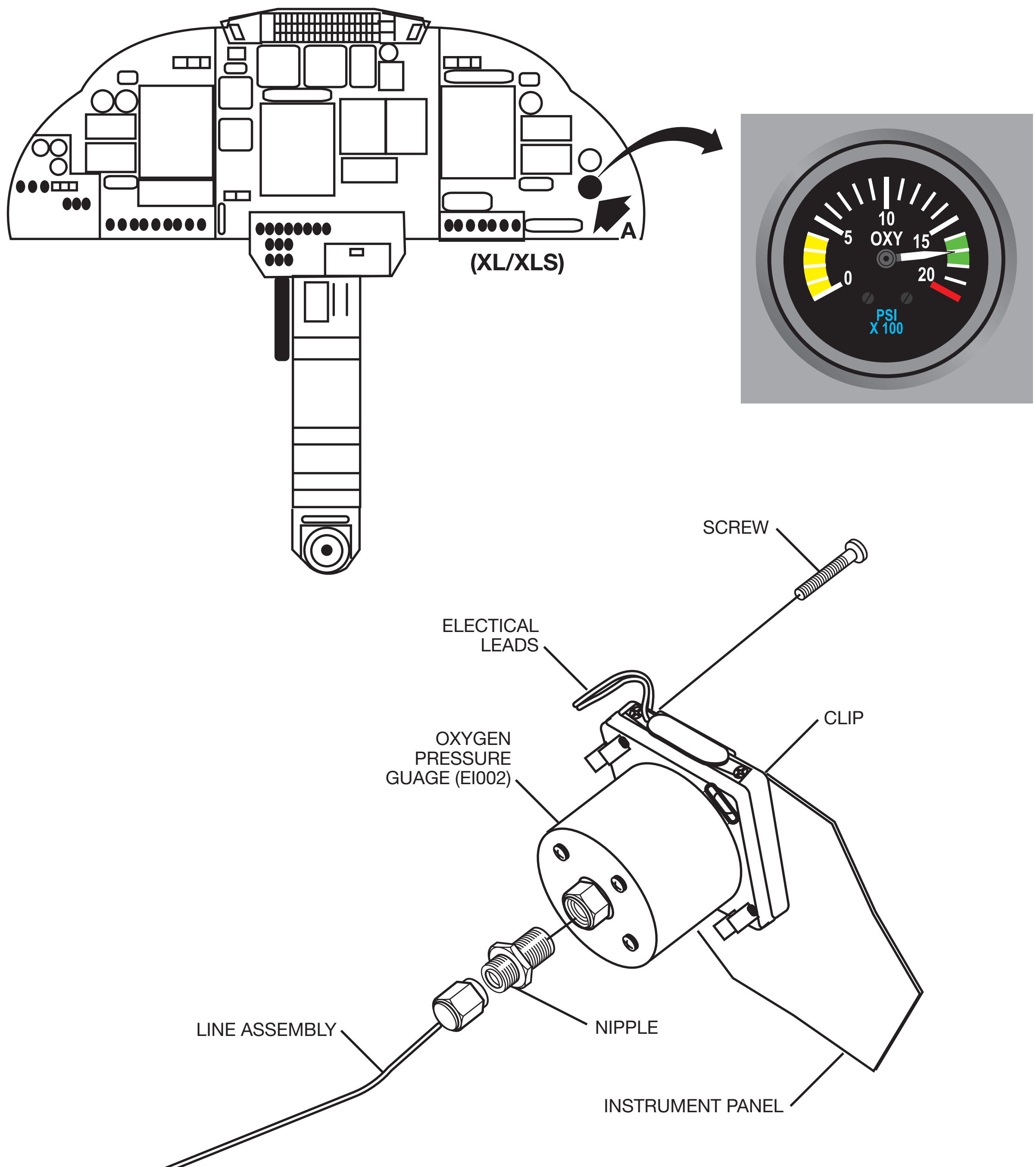


Figure 35-3. Charge Valve

Charge Valve

Servicing the oxygen cylinder and pressure regulator assembly is accomplished through the charge valve (remote fill port) in the airplane nose section at FS 90.20 and RBL 17-50 (Figure 35-3). The charge valve is provided with an integral filter and orifice; and functions as a check valve. It is connected to the oxygen cylinder and pressure-regulator assembly with a high-pressure copper line. A pressure-sealing cap is provided to prevent any contamination from entering the oxygen system. The charge valve is accessed by opening the right nose access door.

NOTES



DETAIL A

Figure 35-4. Oxygen Pressure Gauge

Oxygen Pressure Gauge

There is an internally-lighted oxygen pressure gauge in the right instrument panel (Figure 35-4). The oxygen pressure gauge indicates the pressure within the oxygen cylinder at all times. The oxygen pressure is monitored by a direct connection of the gauge to a high-pressure port on the pressure regulator (attached to the oxygen cylinder). Orifices limit the rate-of-flow through the lines.

At the connection between the high-pressure line and the pressure regulator there is a needle valve assembly that allows the oxygen pressure gauge line to be disconnected at the pressure regulator, without discharging the oxygen cylinder.

WARNING

When disconnecting lines from the pressure regulator, care must be taken to separate only the line from the pressure regulator and not loosen or remove the needle valve assembly from the pressure regulator.

WARNING

Do not remove fittings/check valves from any port of pressure regulator when oxygen cylinder is pressurized. The fittings contain check valves. The ports are always pressurized at cylinder pressure, except the low-pressure oxygen distribution port—which is regulated pressure.

CAUTION

The end of the oxygen pressure gauge line attaches to the pressure regulator check valve fitting incorporating a bayonet probe. When removing oxygen pressure gauge line from the check valve fitting, withdraw carefully to prevent damage.

NOTES

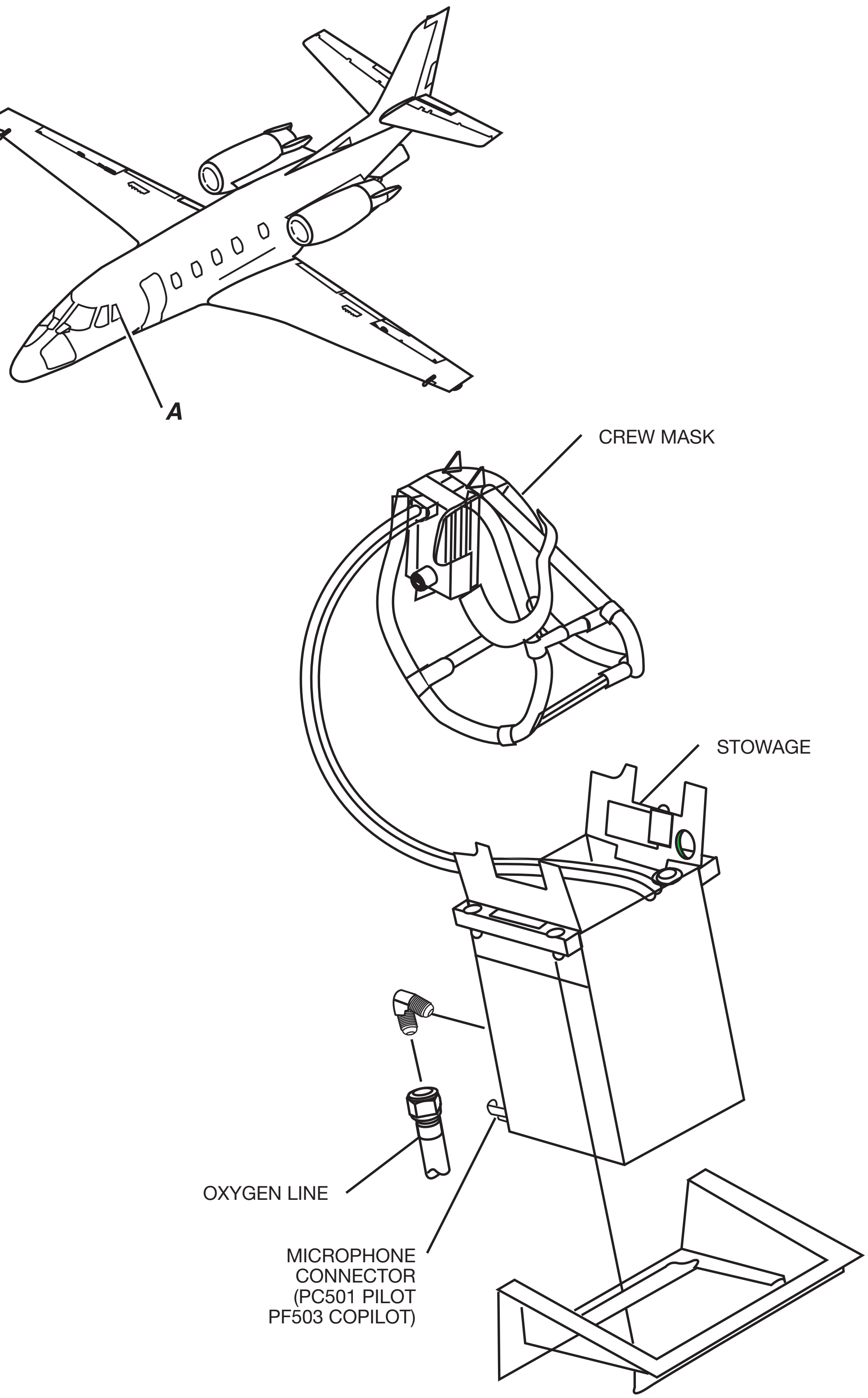


Figure 35-5. Stowage Box

CREW OXYGEN SYSTEM

This section includes only the components that are unique to the crew oxygen system. The crew oxygen system consists of oxygen lines, stowage boxes, and crew masks (Figure 35-5).

DESCRIPTION

There are two EROS brand masks in the flight crew compartment.

Each crew mask is housed in a stowage box in the pilot and copilot side consoles (at FS 132.75 and WL 11 8.34). These stowage boxes fully enclose the crew mask and associated hose connections. A bayonet mount inside the stowage box provides a quick disconnect for the oxygen hose. The stowage box also contains an RCA-type jack for microphone interface. From the stowage box, oxygen lines run directly to the oxygen cylinder and pressure regulator assembly, allowing for crew oxygen anytime the pressure regulator control handle is in the ON position.

The crew masks are quick-donning type with diluter demand, 100% demand, or pressure breathing capabilities (Figure 35-6).

In the diluter demand mode, cabin air is mixed with breathing oxygen based on cabin altitude, which is sensed by the crew mask's internal regulator. This function is selected by pressing the N (for normal) rocker switch on the front of the crew mask regulator.

In the 100% mode, all oxygen to the crew mask is supplied by the oxygen cylinder. This function is selected by pressing the 100% PUSH rocker switch on the front of the crew mask regulator.

In the pressure breathing mode, a positive pressure is created inside the crew mask seal. This positive pressure ensures that no toxic fumes or smoke are inadvertently breathed by the flight crew. This function is selected by rotating the EMERGENCY knob, on the crew mask regulator, in a clockwise direction.

The crew masks stored in the stowage boxes allow for system testing and leak-free verification without removing crew masks from storage.

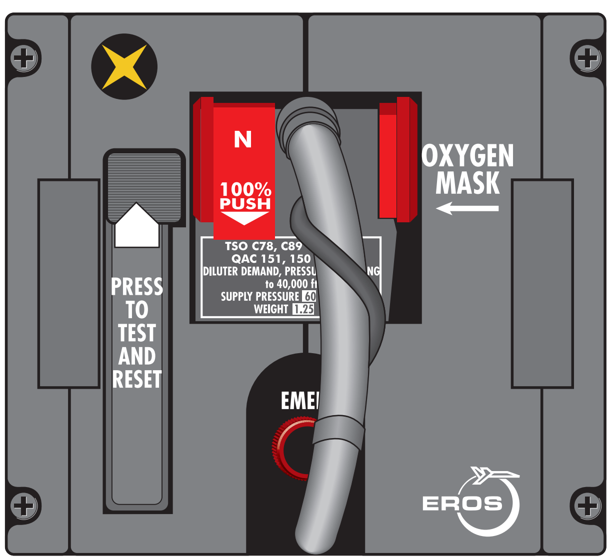


Figure 35-6. Crew Oxygen Mask—Stowed

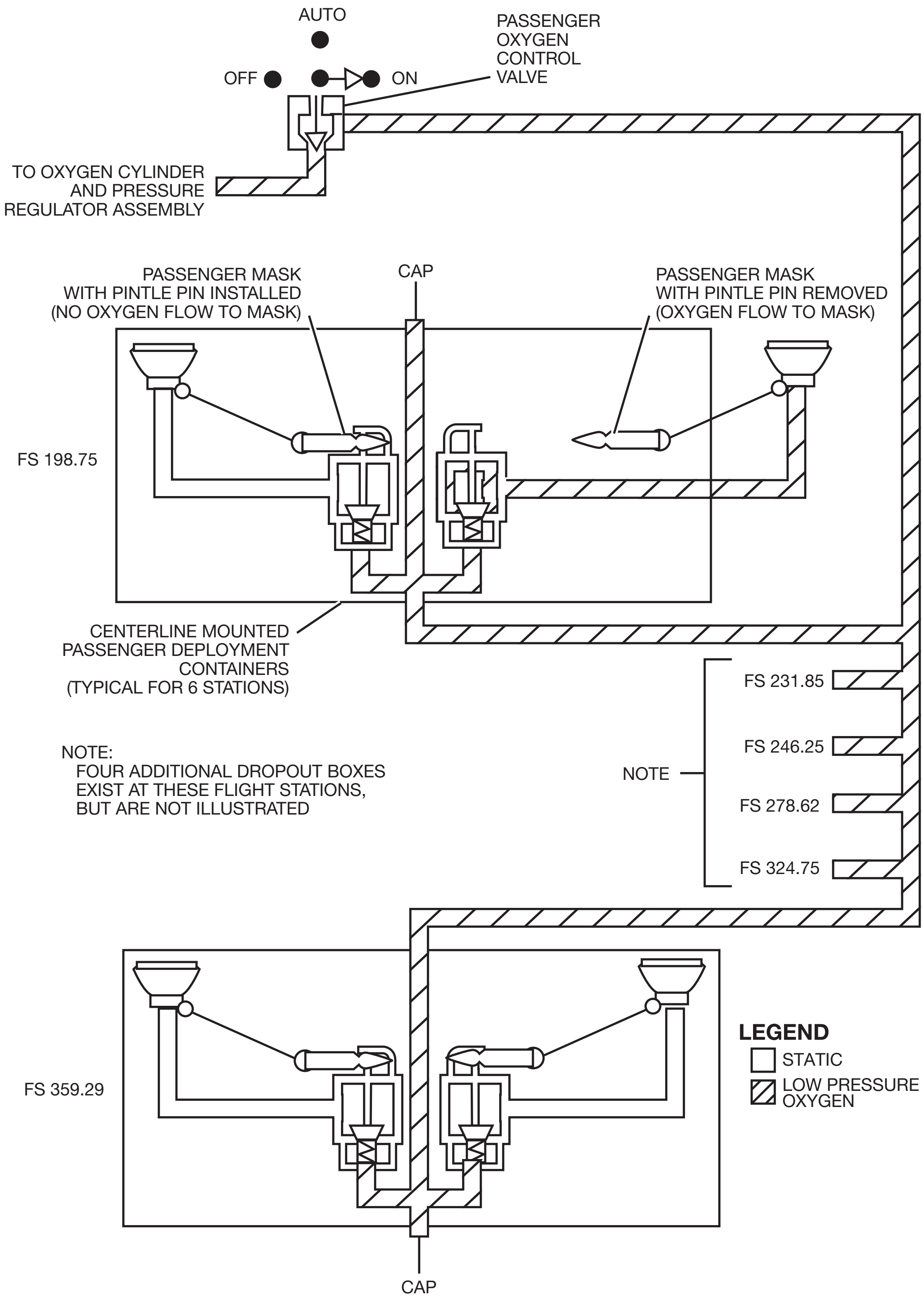


Figure 35-7. Passenger Oxygen System

35 OXYGEN

OPERATION

The crew oxygen system is pressurized at any time that the pressure regulator control handle is turned ON.

The crew masks are normally stowed with the control set to 100%. When the crew mask must be donned, oxygen is immediately available with no additional action required from the crew.

The crew masks incorporate a microphone, allowing uninterrupted communication with cabin and ground while wearing the crew mask.

DIAGNOSTICS

Crew Mask Cleaning

Mask cleaning and disinfecting is accomplished in a single step using SAN50 wipes. These wipes are available from Scott Aviation, distributor for EROS product. Refer to the manufacturer's technical publications.

PASSENGER OXYGEN SYSTEM

DESCRIPTION

The passenger oxygen system uses the same oxygen cylinder and pressure regulator assembly as the crew oxygen system (Figure 35-7). A single line runs from the oxygen cylinder and pressure regulator to the cockpit area. At the cockpit area, the line branches off into two branches. One branch supplies regulated oxygen to the crew masks. The other branch supplies the passenger system with oxygen, but only after passing through the passenger oxygen control valve.

The passenger oxygen system consists of:

- Passenger oxygen control valve
- Altitude pressure switch
- Oxygen lines
- Dropout boxes
- Passenger masks

COMPONENTS

Passenger Oxygen Control Valve

The passenger oxygen control valve is on top of the pilot side console (Figure 35-8). It is a three-position control which may be manually or electrically operated. In the ON position and OFF position, the flow of oxygen to the passenger system is manually regulated (initialized or stopped)—irrespective of cabin pressure altitude. In the AUTO position, the altitude pressure switch opens or closes the passenger oxygen control valve's electrically-operated solenoid, controlling oxygen flow to the passenger system.

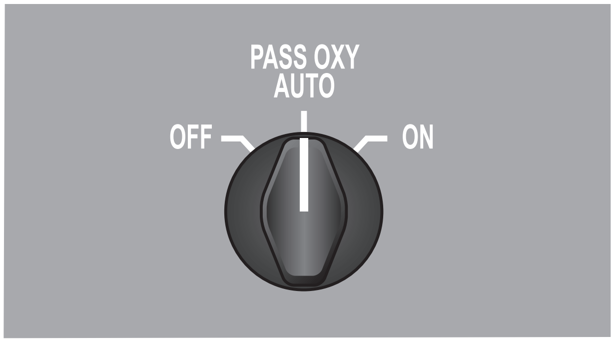


Figure 35-8. Passenger Oxygen Control Valve

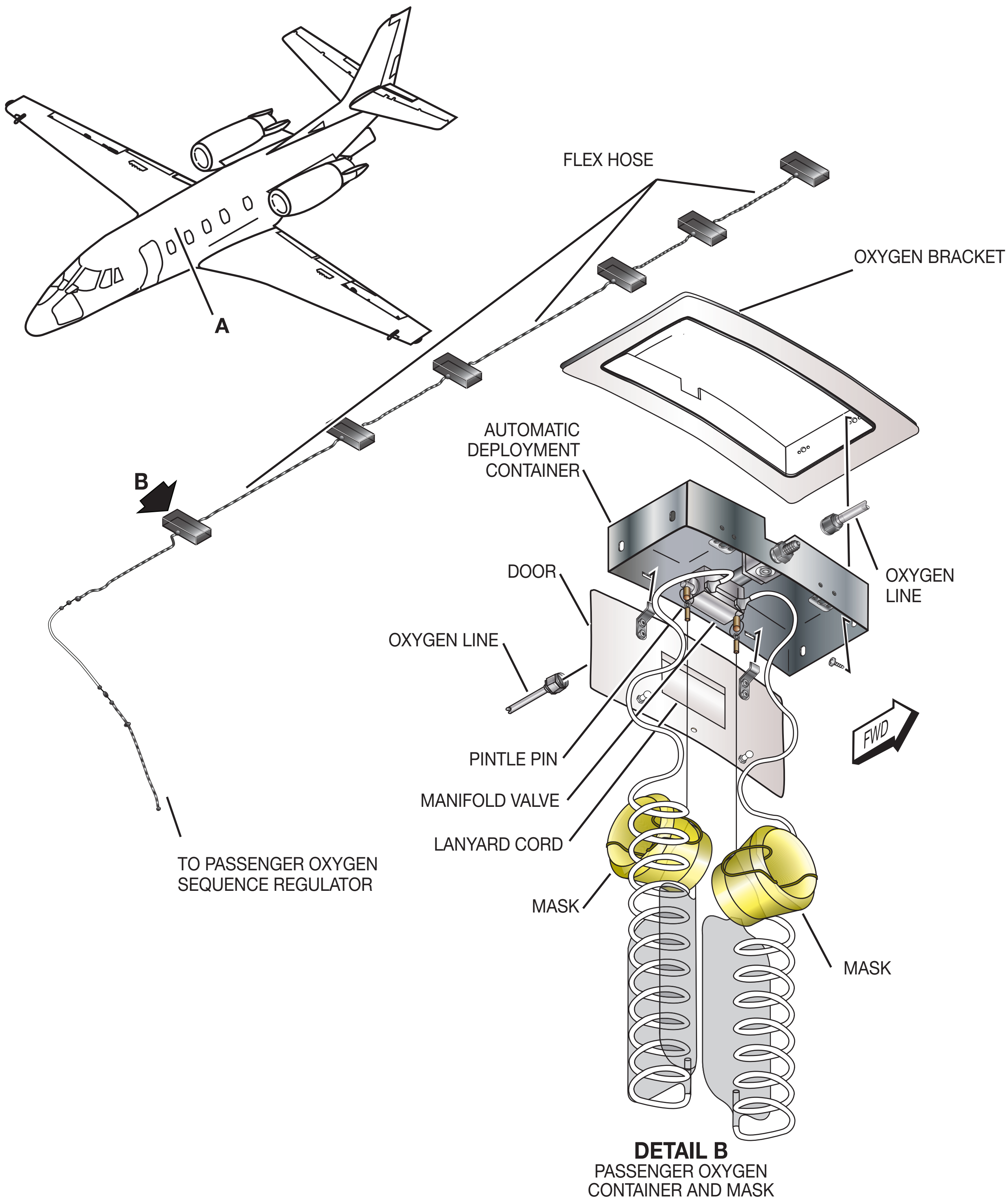


Figure 35-9. Passenger Masks

Passenger Masks

Oxygen masks are stowed in six dropout boxes, overhead in the passenger cabin area (Figure 35-9).

Each dropout box contains two passenger masks.

Whenever the passenger oxygen system is pressurized, a plunger (door actuator) is extended, forcing the dropout box doors open and allowing the passenger masks to drop. The passenger masks are of a constant-flow type with a lipless face piece cushion. They cover the nose and mouth and are secured to the face by an elastic headband. When the passenger mask is deployed, a lanyard cord supports the mask. Reinserting the pintle pin cuts off the flow of oxygen to that particular mask. The passenger masks are suspended in the dropped position by a lanyard cord. The other end of the lanyard cord is connected to a pintle pin. Pulling down on the lanyard unseats the pintle pin, allowing oxygen to flow to the passenger mask.

Dropout Box

The dropout boxes serve as storage units and as passenger mask deployment containers for the oxygen masks in the passenger compartment. There are six dropout boxes in the center overhead panel: one each at FS 198.75, FS 231.85, FS 246.25, FS 278.62, FS 324.75, and FS 358.29. Each dropout box houses two passenger masks which deploy to the left and right of the center overhead panel centerline. Oxygen masks are deployed when oxygen pressure in the lines actuates a plunger, causing the doors to open and the masks to drop out of the dropout boxes.

NOTES

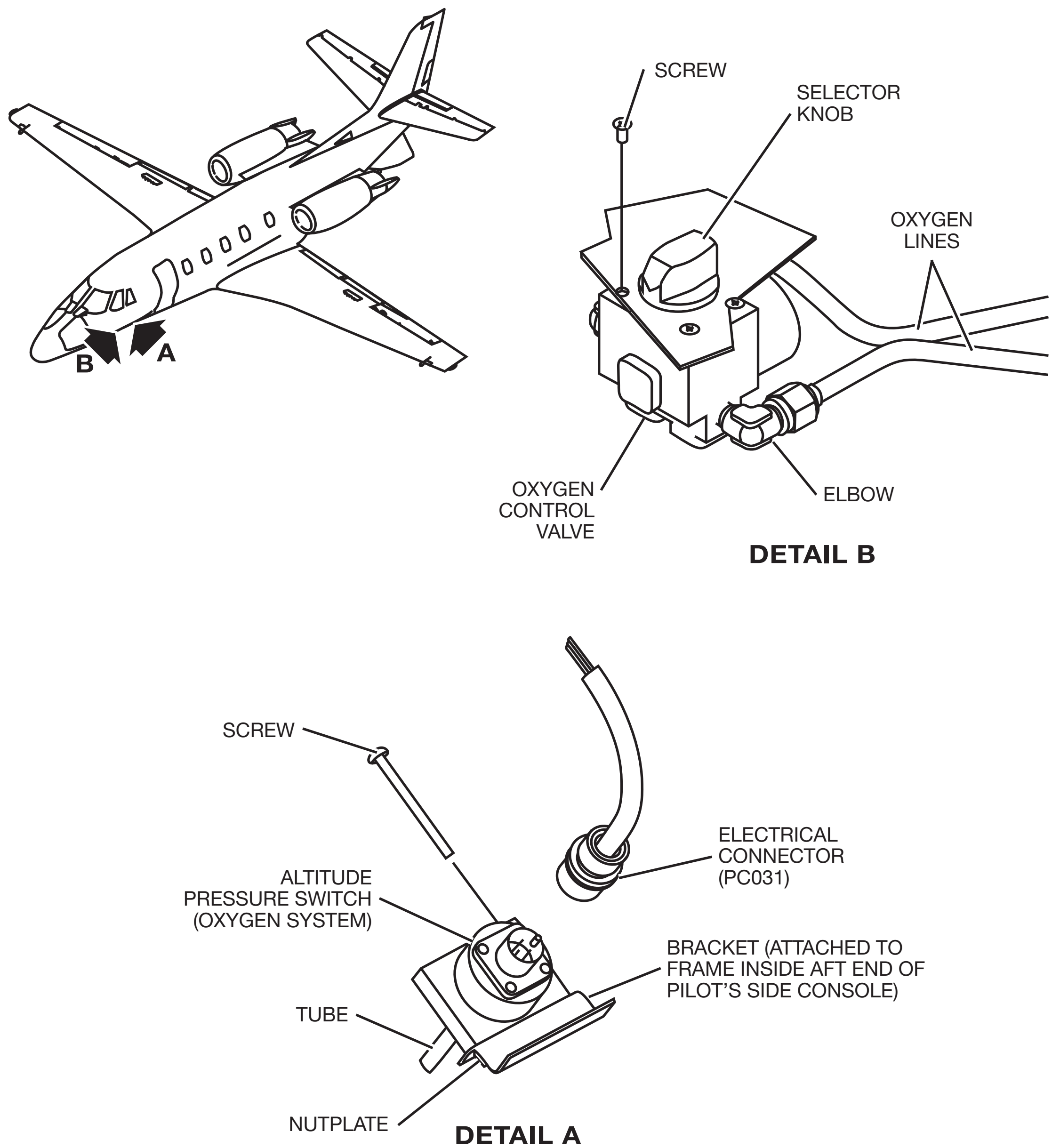


Figure 35-10. Passenger Oxygen Control Installation

Altitude Pressure Switch

The altitude pressure switch is in the pilot side console on a plate assembly at FS 155.00 (Figure 35-10). It senses the cabin pressure via a cabin static port tube (piccolo tube) assembly and electrically activates (opens) or deactivates (closes) the passenger oxygen control valve—based on the altitude pressure within the cabin. When a cabin pressure altitude of 14,500 ± 500 feet (4420 ± 152 m) is sensed, then the altitude pressure switch activates the passenger oxygen control valve. The altitude pressure switch deactivates the passenger oxygen control valve at a minimum cabin pressure altitude of 12,000 feet (3658 m).

OPERATION

When the oxygen cylinder's pressure regulator is in the ON position, cylinder pressure is reduced to 70 ± 10 psi (482 ± 69 kPa) and is plumbed to the passenger oxygen control valve. In case of decompression, the cabin altitude pressure switch senses the increased cabin pressure altitude. With the passenger oxygen control valve in the AUTO position and when a cabin pressure altitude of 14,500 ± 500 feet (442 ± 152 m) is sensed, the altitude pressure switch energizes a normally closed solenoid valve in the passenger oxygen control valve, and supplies regulated oxygen to the passenger system. The pressure in the passenger oxygen lines actuates the door release mechanisms in the dropout boxes, allowing the passenger masks to drop.

In case of electrical failure, or any time at the crew's discretion, oxygen is provided to the passenger system by positioning the passenger oxygen control valve to ON. The ON position of the passenger oxygen control valve bypasses the solenoid-valve operation, allowing oxygen to flow to the passenger system with the same operating sequence as in automatic actuation. After deployment of the passenger masks, oxygen flow to an individual mask is initiated by pulling on the lanyard cord supporting the passenger mask. This disengages a pintle pin, allowing oxygen to flow through the masks.

NOTES

DIAGNOSTICS

Passenger Mask Inspection and Check

NOTE

It is recommended that passenger masks are inspected anytime they are unstowed.

1. Check for stickiness of the economizer bag. Mask must not stick to the dropout box or to itself.
2. Check for contamination of the passenger mask or dropout box.
3. Check to see if excessive force is required to remove the lanyard pin.
4. Check for tears, cracks, or deterioration of the passenger mask, or the economizer bag (unfold bag, if necessary).
5. Check oxygen supply hose for kinked hoses.
6. Check the legibility and presence of the donning instructions label on the dropout doors.
7. Check for proper installation of the lanyard pin in the manifold valve.
8. Inspect the front and back manifold valve housing for cracks, breaks, or damage to valve seat.
9. Check the bag for torn or imperfect seams, holes, or mildew.
10. Check tubing for cracks, kinks, and security of installation.
11. Check the headstrap for corroded or distorted clips, elasticity, cleanliness; to see that they are securely installed.

Disinfect Mask and Container

1. The mask and container must be disinfected with an aqueous solution of Zephiran Chloride or QS4 disinfectant concentrate, in concentrations recommended by the manufacturer. Refer to Introduction, List of Manufacturers' Technical Publications.
2. After disinfecting and thoroughly drying the mask, lightly dust the outside of the facepiece with Neo-novacite powder.
3. Contamination can be removed with a mild soap and water solution.

NOTES

MAINTENANCE PRACTICES

DESCRIPTION

Maintenance of the oxygen system includes:

- System purging, following system maintenance, that requires plumbing disconnection
- System leakage and operational tests
- Troubleshooting a failed system
- Removal and installation of system components

Charging the oxygen system is required only if the system pressure is low.

NOTE

Do not allow the system to become fully discharged. If the oxygen bottle discharges fully, it must be removed and returned to the manufacturer or an FAA approved overhaul station for reconditioning.

Before maintenance is performed on the oxygen system, maintenance personnel must read, understand, and adhere to the following WARNINGS and CAUTIONS. Observing of these WARNINGS and CAUTIONS aids in the safe maintenance of the oxygen system.

Warnings

Oxygen System Warnings

WARNING

Do not permit smoking, open flame, or potential sources of electrical sparks near the airplane while maintenance is being performed on the oxygen system. Ensure that all electrical power is disconnected and that the airplane is properly grounded.

WARNING

Use extreme caution to ensure that all components are kept thoroughly clean of oil, grease, or solvent contamination. These or similar products may explode or burn spontaneously when in contact with pure oxygen.

WARNING

The entire oxygen system must be kept free of moisture, as the cooling produced by expansion of the compressed oxygen or low operating temperatures causes water to freeze in the small orifices of the system.

WARNING

Never attempt to remove or tighten oxygen system fittings while the system is pressurized. Personal injury could result.

WARNING

Do not blow out the lines with compressed air. Most air compressors are lubricated with oil, and small amounts of oil stay entrained in the air flow. Use only dry nitrogen or argon to blow out the lines.

WARNING

The oxygen cylinder is shipped with a partial to full charge. Care must be taken not to drop cylinder or otherwise damage the bottle or the pressure regulator.

OXYGEN SYSTEM FUNCTIONAL TEST

1. A successful oxygen system leakage test must be completed before proceeding with the functional test.
2. Verify that the oxygen cylinder's pressure regulator control handle is in the ON position, and safety wired.
3. Ensure that the oxygen cylinder is serviced to 1800 psig at 70°F.
4. Gain access to the pressure altitude switch in the left side console.
5. Attach the vacuum pump to the piccolo tube on the altitude pressure switch.
6. Tee the altimeter into the vacuum pump vacuum line.
7. Apply electrical power to the airplane. Verify that the OXY/SEAT BELT circuit breaker on the power junction box is engaged.
8. Position the passenger oxygen control valve to AUTO.
9. Plug a crew mask into the pilot oxygen outlet valve.
10. Place the mask regulator to the EMER position and verify oxygen flow.
11. Disconnect the crew mask and plug it into the copilot oxygen outlet valve.
12. Place the mask regulator to the EMER position and verify oxygen flow.
13. Very slowly, apply a vacuum on the piccolo tube (while monitoring the altimeter) until the passenger oxygen masks drops.
14. Verify that all the passenger oxygen masks were released and dropped at an altitude pressure of 14,500 ± 500 feet (4420 ± 152 m).
15. Pull the pintle pins for all the passenger masks.
16. Check the oxygen flow for each passenger mask by connecting a flowrator to a mask and measuring oxygen flow. Repeat this measurement for each passenger mask. The minimum flow for each passenger mask is 4.5 liters per minute (1.19 gallon per minute).
    (a) If a flowrator is not available, perform the following as an alternate method to check passenger mask oxygen flow:
        1 Verify that the economizer bag attached to the passenger mask is fully collapsed.
        2 Temporarily block the oxygen flow through the mask, at the point where the bag (economizer bag) attaches to the mask face piece.
        3 Note the time that it takes to fill the bag with oxygen.
        4 The maximum allowable time fully fill the bag with oxygen is 17.5 seconds. Repeat measuring oxygen flow for each passenger mask.
17. Slowly relieve the vacuum applied to the altitude pressure switch until the altitude pressure switch closes and stops oxygen flow to the passenger masks.
18. Verify oxygen flow to the passenger masks stops (altitude pressure switch closes) prior to, or at a pressure altitude reading of 12,000 feet (3658 m) minimum.
19. Turn the passenger oxygen control valve from AUTO to ON and verify that oxygen flow is again supplied to the passenger masks.
20. Insert the pintle pin for each passenger mask and verify that oxygen flow stops at each mask.
21. Remove the safety wire that secures the pressure regulator control handle in the ON position, and move the handle to OFF. Refer to Chapter 20—"Safetying: Maintenance Practices".

22. Rotate the passenger oxygen control valve to ON.
23. Pull a couple pintle pins to vent the low-pressure oxygen lines.
24. After venting the oxygen low-pressure system, install the removed pintle pins and rotate the passenger oxygen control valve to AUTO.
25. Stow the passenger masks in the dropout boxes and close the dropout box doors.
26. Remove the vacuum pump and altimeter.
27. Install left side console panels previously removed to access altitude pressure switch.
28. Remove electrical power from airplane.
29. Move the pressure regulator control handle to ON and safety wire.
30. Service the oxygen cylinder to 1800 psig at 70°F (21°C).

NOTES

QUESTIONS

1. The oxygen regulator reduces the pressure of the oxygen cylinder into the distribution system to:
   A. 30 ± 10 psi
   B. 70 ± 10 psi
   C. 1,800 psi
   D. 1,850 ± psi

2. The passenger oxygen door plunger, used to open the door, is operated by:
   A. Hydraulic actuator
   B. Electrical locking actuator
   C. 1,800 psi of oxygen pressure
   D. Regulated oxygen pressure from the oxygen regulator

3. Oxygen supply to the passengers is shut off by placing the PASS OXY selector to OFF?
   A. True
   B. False

4. Crew oxygen masks are diluter-demand or 100% type masks. The 100% position is normally used:
   A. Above 20,000 feet
   B. Below 20,000 feet
   C. Between 8,000 and 13,000 feet
   D. When oxygen pressure is greater than 2,000 feet

5. Selecting OFF with the shutoff valve portion of the oxygen regulator shuts off which lines?
   A. All lines from the regulator
   B. Fill line, gauge line, and overboard discharge line
   C. Low pressure line (±70 psi) to oxygen system
   D. Fill line only

6. The cockpit oxygen pressure gauge reads:
   A. Oxygen pressure at the crew masks
   B. Bottle pressure, electrically
   C. Bottle pressure, mechanically
   D. Requires DC power

7. Passenger masks are dropped as follows:
   A. Automatically with the PASS OXY selector in AUTO and when cabin altitude exceeds 14,500 feet
   B. If cabin altitude exceeds 13,500 feet, regardless of PASS OXY selector
   C. PASS OXY selector ON regardless of altitude
   D. Both A and C

8. If DC power fails, placing the PASS OXY selector in:
   A. ON deploys the passenger masks, regardless of DC power on or off
   B. ON deploys the passenger masks only if 14,500 feet cabin altitude is exceeded
   C. OFF does not restrict oxygen to the crew; only if the cabin altitude is above 14,500 feet
   D. None of the above

9. The purpose of the altitude pressure switch is to:
   A. Bypass oxygen flows directly to the passengers regardless of the PASS OXY selector position
   B. Open a solenoid at 14,500 cabin altitude, allowing oxygen flow to the passenger oxygen distribution system
   C. Close a solenoid valve at 14,500 feet cabin altitude, stopping oxygen flow to the passengers
   D Open a solenoid if the PASS OXY selector is in AUTO and the cabin exceeds 10,000 feet

CHAPTER 36
PNEUMATICS

CONTENTS

ILLUSTRATIONS

CHAPTER 36
PNEUMATICS

INTRODUCTION

This chapter presents the pneumatic system for the Citation XL/XLS/XLS+ aircraft, with discussion primarily limited to the delivery and control of bleed air into the tail cone area and the service air system. Each bleed-air user system is covered in detail within the appropriate chapters of this training manual. General maintenance considerations are included, with an introduction to functional and operational checks. References for this chapter and further specific information regarding components or operation can be found in Chapter 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 36—"Pneumatics" in the *Aircraft Maintenance Manual (AMM)*.

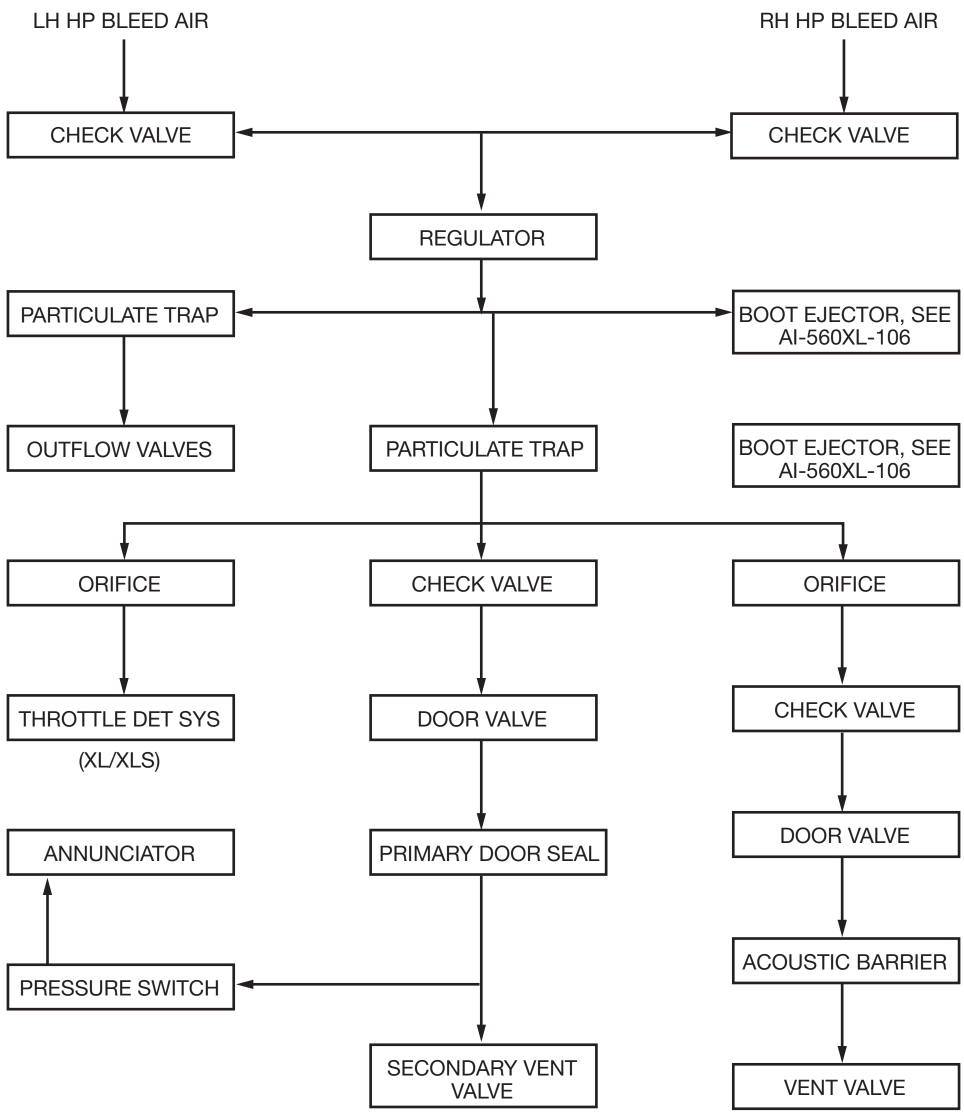


Figure 36-1. Pneumatics Service Air Flow Diagram

GENERAL

This section describes the extraction, control, and distribution of bleed air from the engines to the connecting components of systems that utilize bleed air and service air for temperature and/or pressure purposes (Figure 36-1).

This section provides description and maintenance information on the pneumatic distribution system components in the nacelle, pylon, and tail cone. This includes hardware required to duct the bleed air through the pylon into the tail cone, but does not include the engine inlet anti-ice system, the wing anti-ice system or the air-conditioning system.

PNEUMATICS

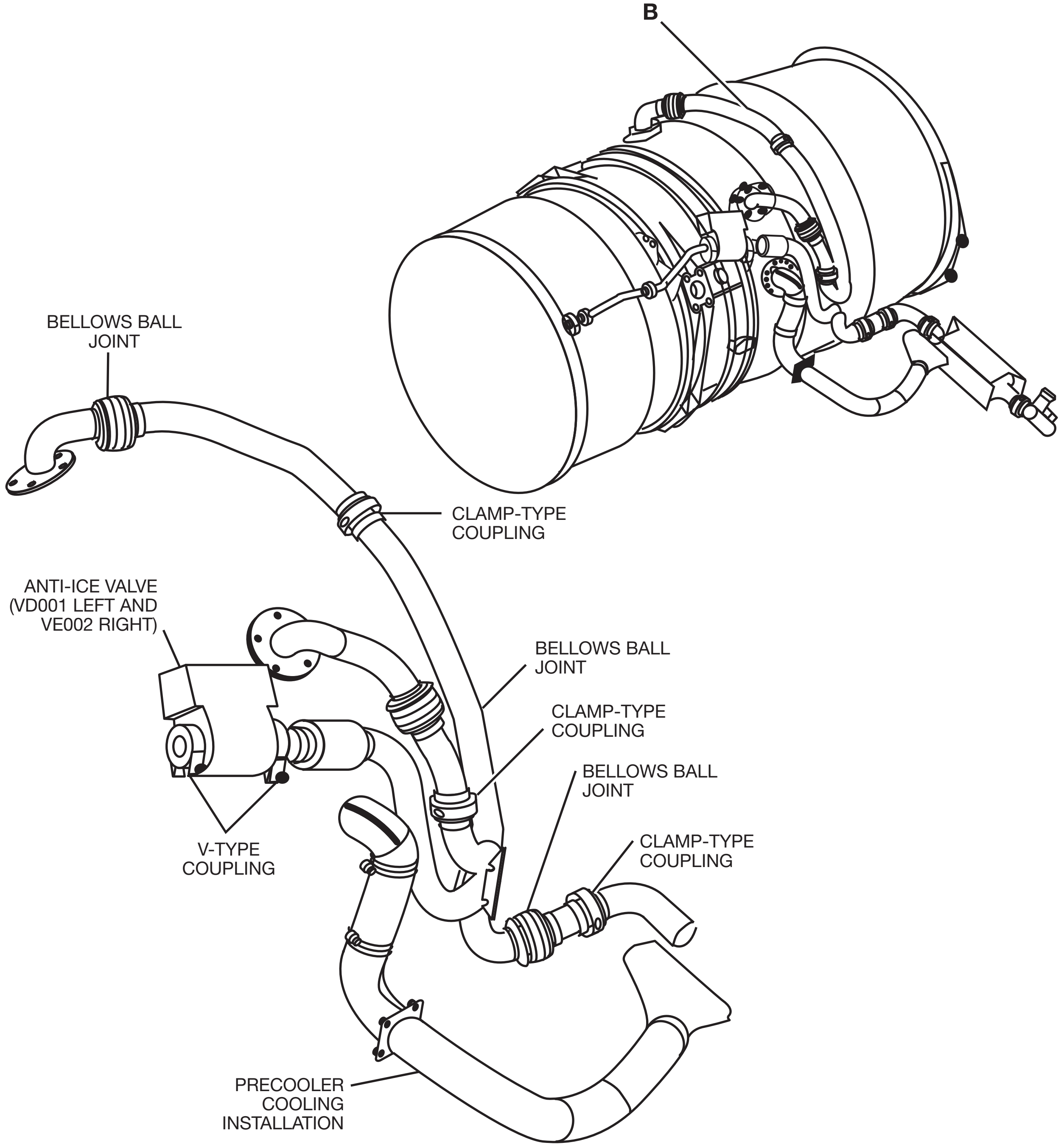
DESCRIPTION

Airplane systems that utilize engine bleed air and service air are as follows:

- Air-conditioning, cabin pressurization, and cabin temperature control systems
- Wing leading edge anti-ice system
- Horizontal stabilizer deice boot system
- Engine inlet anti-ice system
- Cabin door primary seal
- Cabin door acoustic seal
- Throttle detent system
- Rudder bias system

Bleed air is extracted from the engine at all times when the engine is operating. Usage of the bleed air depends on the position of the control valve of individual systems using bleed air.

NOTES



DETAIL B

Figure 36-2. Engine Bleed-Air Plumbing

COMPONENTS

Engine Plumbing

Engine bleed air extracted from the high-pressure compressor is referred to as primary bleed air or just bleed air (Figure 36-2). Components used to transfer the bleed air from the compressor to the exterior ducts on the engine limit the volume of air available for using systems.

High-temperature (primary) bleed air is extracted from the high-pressure compressor delivery. Transfer tubes are at the 1:00 o'clock and 11:00 o'clock positions on the engines. The bleed-air transfer tubes are designed to a specific inside diameter size to draw a percentage of bleed air from the high-pressure compressor delivery. Components for extracting the bleed air are engine components.

Bleed air is collected from the engine bleed-air transfer tubes through ducts. Couplings and ducts connect to the collecting ducts and route the bleed air through the pylon and into the tail cone bleed air distribution system. Part of the bleed air is routed to the engine inlet anti-ice system.

NOTES

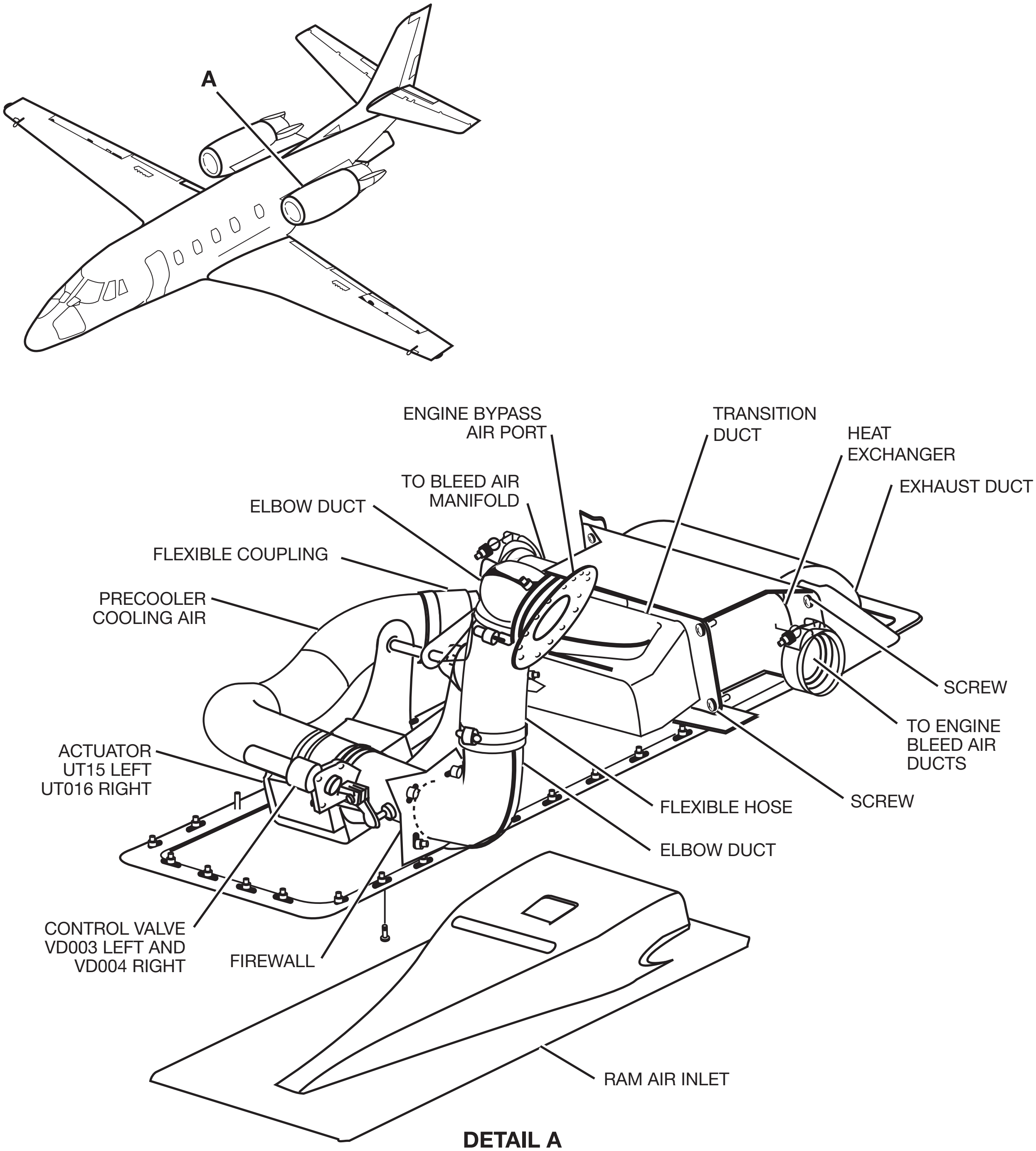


Figure 36-3. Precooler Installation—XL

Precooler

The purpose of the precoolers is to take the hot bleed air (up to 800°F) and cool it to a temperature that makes it useful in the aircraft anti-ice, environmental and other pneumatic systems (Figures 36-3, 36-4, and 36-5). Precoolers are the primary means of regulating temperature of bleed air going to the wing anti-ice system. The precoolers are conventional cross flow heat exchangers in the pylons. They have two primary flow paths, the bleed air flow (hot) and the cooling air flow (cold). On the ground, cooling air is provided from engine fan air. In flight (XL only), cooling air is forced across the precoolers by the use of a NACA scoop on the bottom of the pylon. The cooling air exit is just aft of the NACA scoop on the bottom surface of the pylon. The XLS/XLS+ use fan air for cooling on the ground and in flight.

Precooler Temperature Controls (XL)

The crossflow cooling is controlled to provide fan air and deactivates the in-flight proportional scoop while the aircraft is on the ground. The fan air is linked by the "ground-on-ground" from the squat switch. The scoop is deactivated to prevent fan air from exiting through the scoop instead of across the precooler. Conversely, in flight, the fan air is deactivated and the proportional scoop provides the cooling. The purpose of deactivating the fan air in flight is to conserve thrust. The proportional scoop uses the "ground-in-air" to activate the system from the squat switch.

On the ground, the bleed air temperature from the exit to the precoolers is controlled to a range of 380°F to 405°F by allowing fan air to be used for crossflow. Fan air is approximately 100°F above ambient conditions. The temperature sensor is closed until 405°F at which time it opens and applies voltage to a gas heated linear actuator, which in turn opens the fan air butterfly valve. The sensor then closes at 380°F, which in turn closes the fan air butterfly valve. The gas-heated linear actuator damps out oscillations in the operation.

NOTES

XL

XLS/XLS+

Figure 36-4. Precooler Installation Doors—Bottom View

XL

XLS/XLS+

Figure 36-5. Precooler Installation—Top View

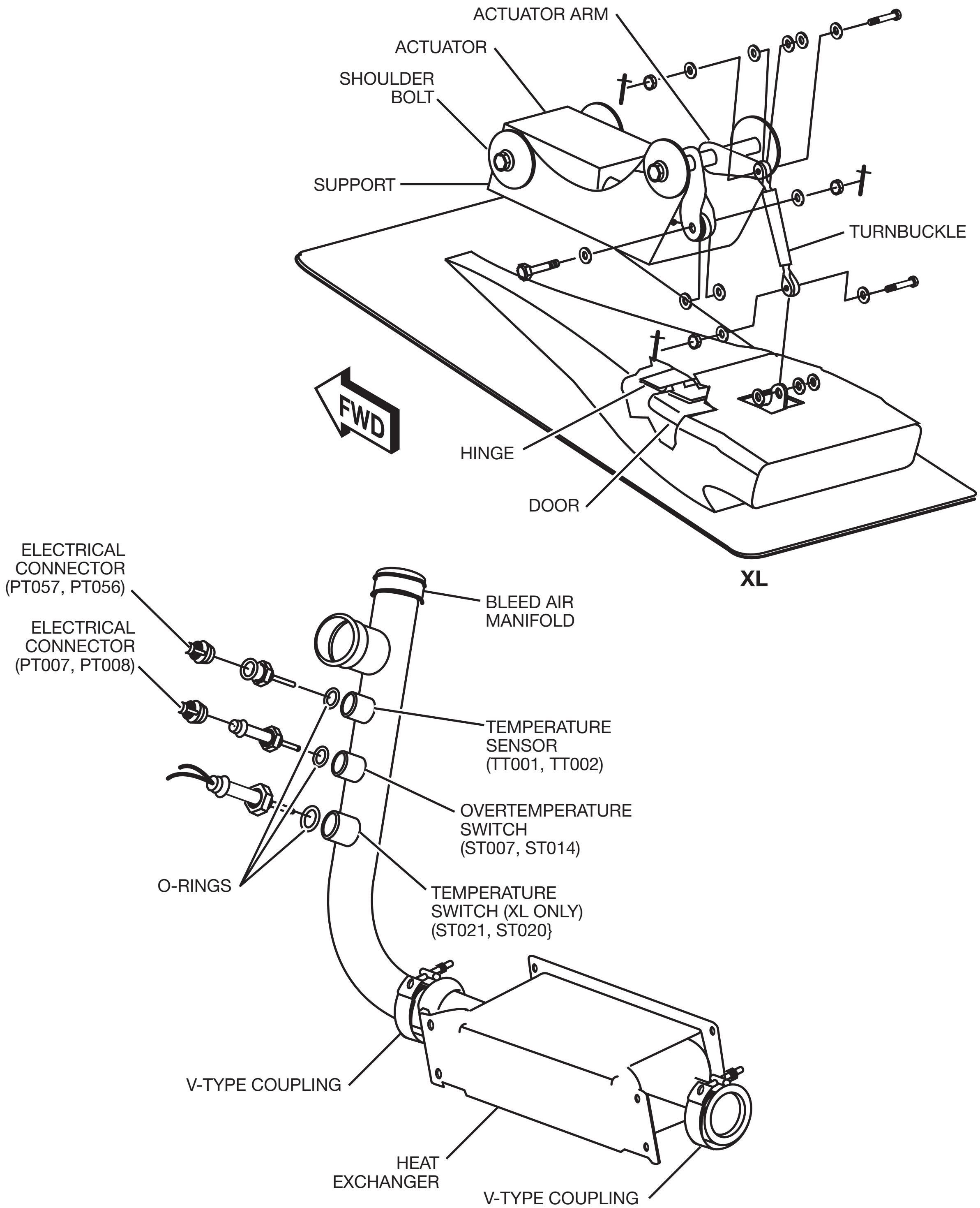


Figure 36-6. Bleed-Air Temperature Control

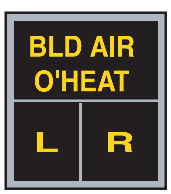
At the exit of the precoolers the bleed-air temperature is controlled to between 475 ± 15°F by modulating the amount of cooling air passing across the precooler (Figures 36-6 and 36-7). This modulation is accomplished by using the floor or ramp portion of the NACA scoop as a door that restricts the flow of air through the cooling air circuit. This ram air system is disabled on the ground.

The temperature control system is an electronic system that utilizes a sensor downstream of the precooler to measure the temperature of the bleed air. The system incorporates a "dead band" from 460°F to 490°F. If the temperature of the bleed air is within this range, no corrective action is required. If the measured temperature is between 490°F to 530°F or between 420°F to 460°F, the actuator compares the measured temperature to a reference temperature of 475°F. The controller then moves the actuator in proportion to the error. For example, if the controller detects that the bleed air is at 500°F, the controller supplies a couple pulses of voltage to the actuator. If the controller is detects 525°F bleed air, it might send 10 pulses. If the measured bleed air temperature is under 420°F or over 530°F, the controller sends a 100% extend or retract signal to the actuator.

NOTE

The XLS/XLS+ utilizes the same actuator controls for continuous operation on ground or in flight, but uses fan air only, without the aid of a ram air NACA scoop.

If for any reason the bleed air temperature exceeds 560 ± 10°F (293 ± 5°C), the overheat switch causes the L–R BLD AIR O'HEAT annunciator (XL/XLS) or BLEED AIR OVERTEMP L–R CAS message (XLS+) to illuminate and closes the respective wing anti-ice pressure regulating shutoff valve.



L/R BLD AIR O'HEAT
Annunciator flashes if the bleed air precooler (engine pylon mounted) output temperature has exceeded 560°F, activates MASTER CAUTION lights. If the wing A/I is ON, the respective wing A/I, L or R, will automatically shutdown.

XL/XLS ANNUNCATOR

BLEED AIR OVERTEMP L-R			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	20 Second

This message is displayed when the supply bleed air from the engine is too hot. A temperature switch in the supply duct provides a 28V signal to the EICAS, which posts the message after 20 seconds. When the supply temperature is normal, the switch provides an open to the EICAS, which removes the message.

XLS+ CAS MESSAGES

Figure 36-7. Bleed-Air Annunciation

36 PNEUMATICS

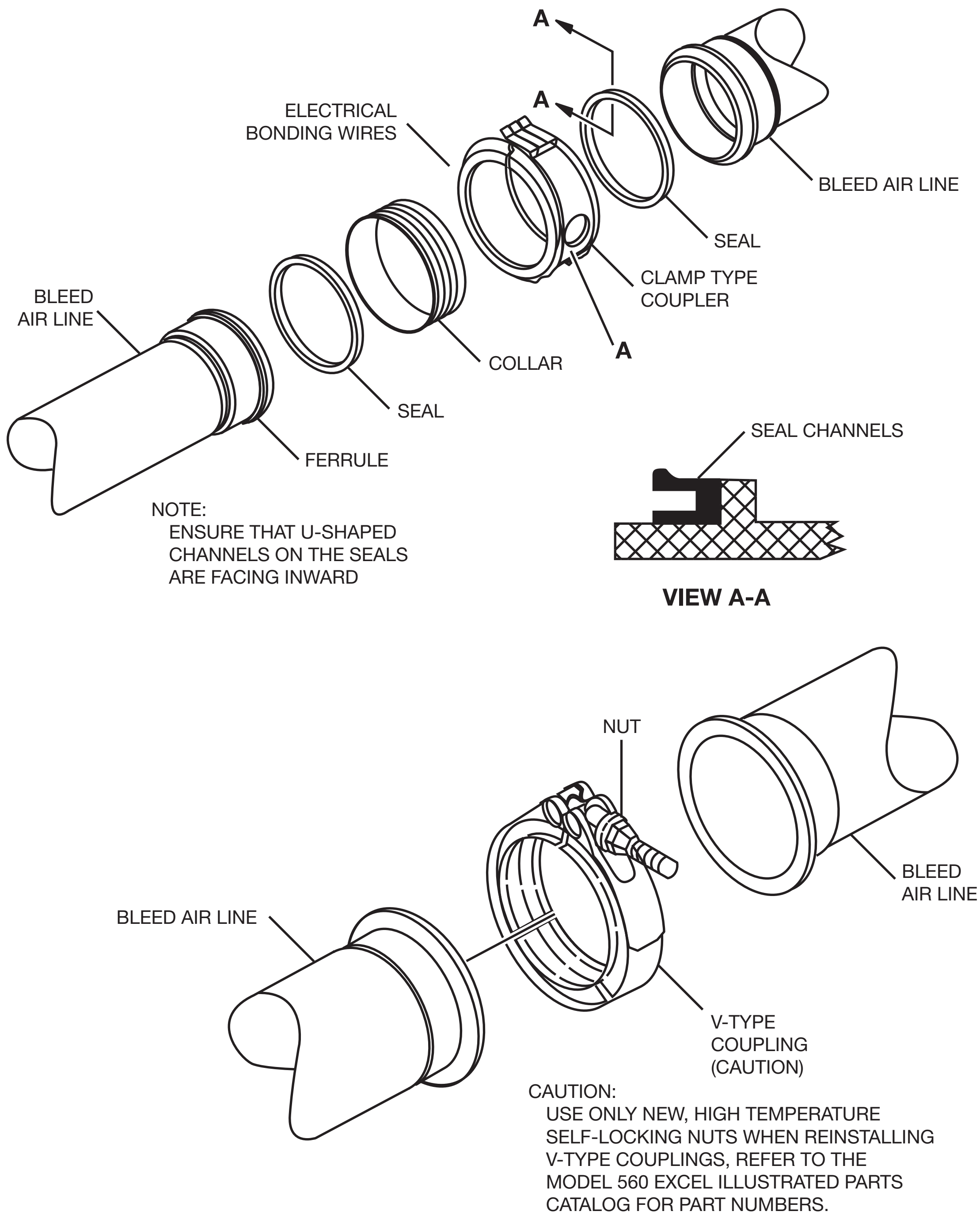


Figure 36-8. Bleed-Air Couplings

Couplings

Two types of couplings are used: for tube-to-tube and component-to-tube connections (Figure 36-8).

The clamp-type coupling assembly consists of a collar, two seals and a retainer (clamp). Use of this assembly requires ferrule ends on both bleed air lines. The retainer (clamp) assembly incorporates electrical bonding wires and is secured by a double-locking latch.

NOTE

Ensure that the seal U-shaped channels are facing inward, toward the joint.

The V-type couplings are designed to be used on special joints and do not require gaskets.

NOTES

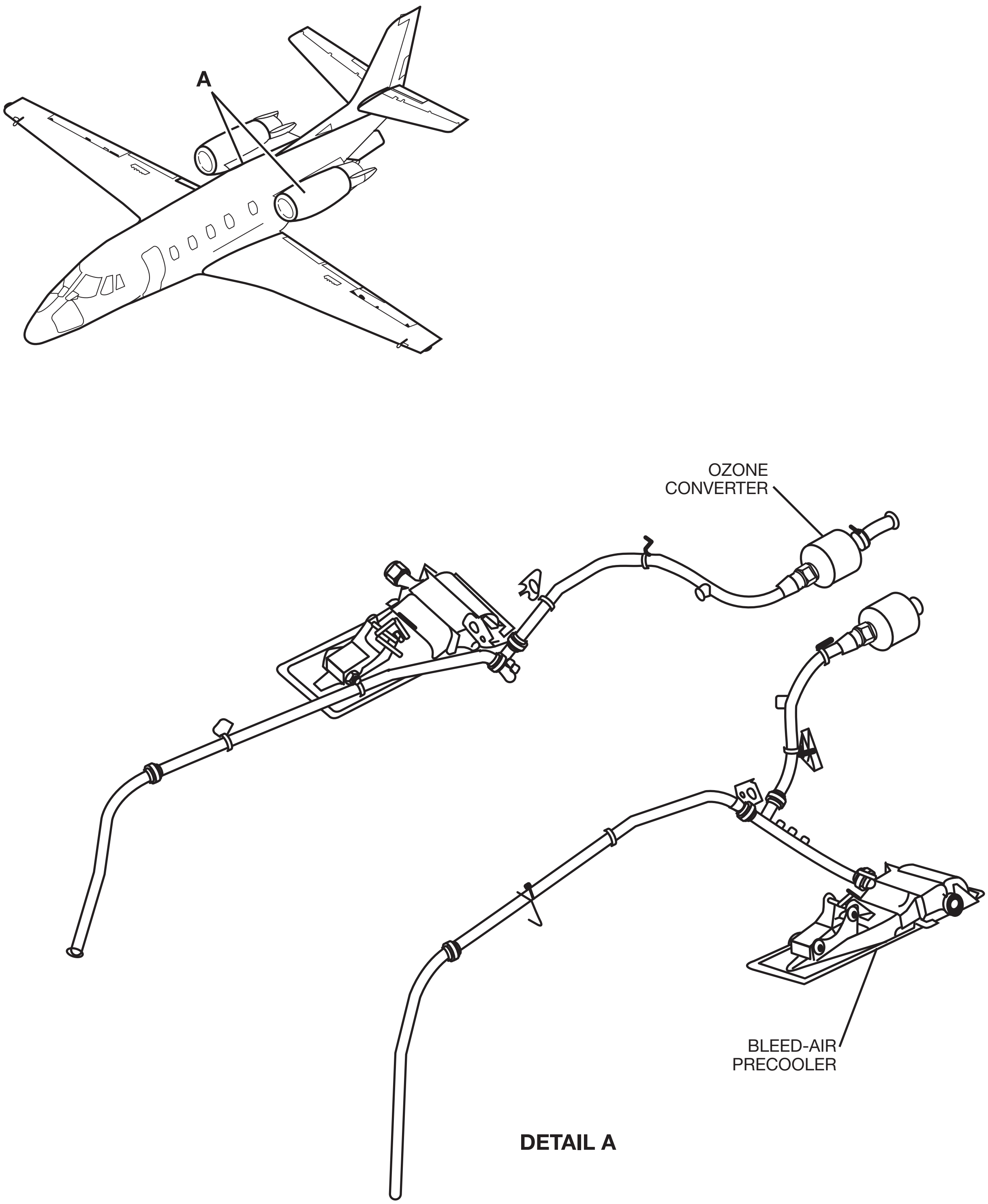


Figure 36-9. Tail Cone Bleed-Air Plumbing and Components

Ozone Converter

Two ozone converters (one left and one right) are plumbed so that all engine bleed air is filtered through the converter before flowing into the environmental control unit (ECU). The ozone converter is a one-piece assembly consisting of a corrosion resistant steel housing (Figure 36-9). Inside the housing, a ceramic honeycomb core is shock-mounted with corrosion-resistant woven steel pads. The housing is a welded assembly with flanges that connect that the bleed air system. The plating on the ceramic honeycomb converts ozone to oxygen by catalytic action.

DIAGNOSTICS

Troubleshooting the pneumatic distribution system primarily involves checking fittings, couplings and pneumatic lines from the engine to the bleed-air manifold and the service air system.

Troubleshooting may be needed if the distribution system is inoperative or if bleed-air systems are sluggish.

- Broken or damaged distribution ducting/pneumatic lines are the probable cause for inoperative or sluggish bleed-air system(s). This type of failure usually affects more than one system.
- Malfunction of bleed-air flow control units normally causes erratic operation, or failure of a directly related system.
- A defective distribution system check-valve may be evident if the horizontal stabilizer deice and cabin door seal systems are inoperative (with only one engine operating) but these systems operate adequately with both engines running.

NOTES

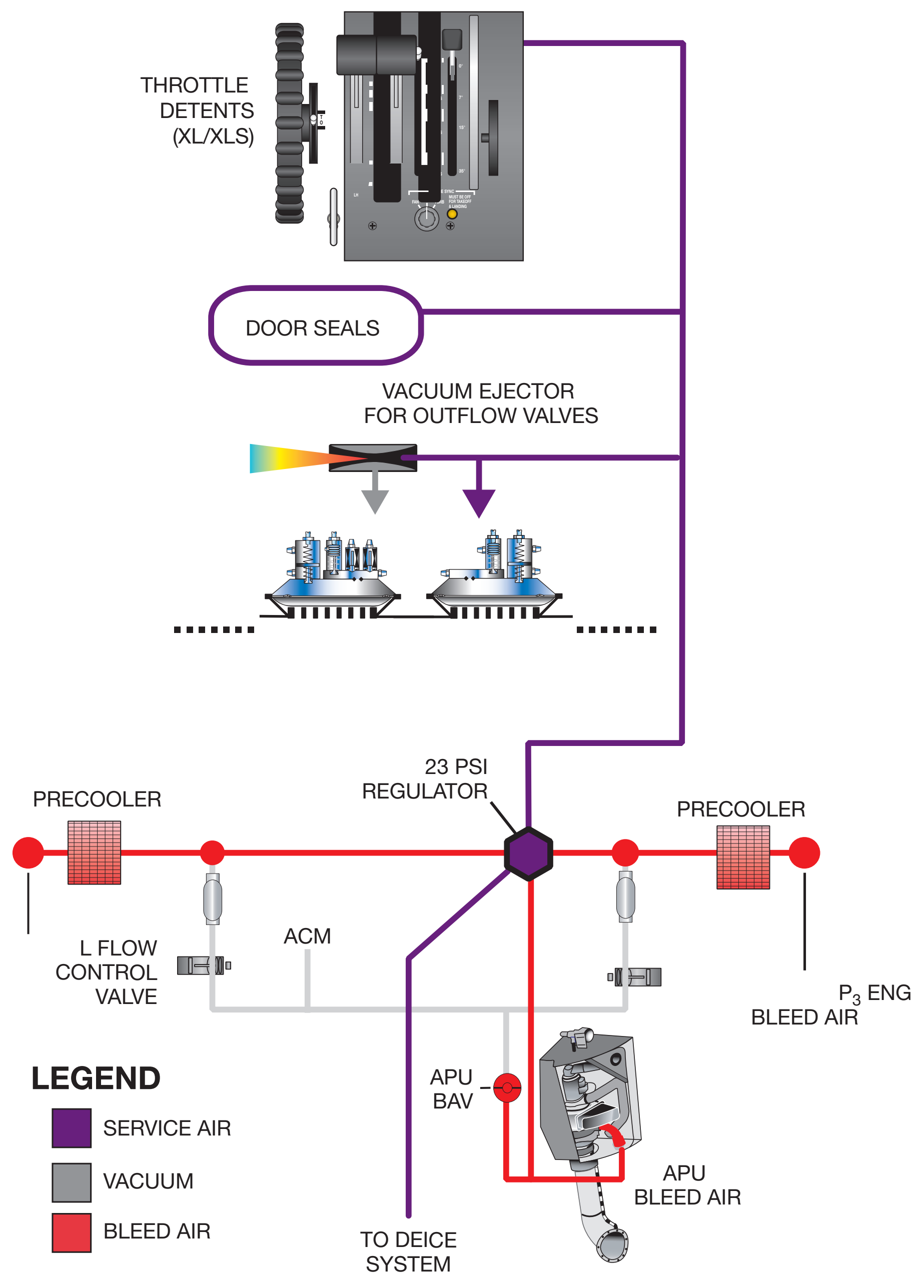


Figure 36-10. Service Air System Diagram

SERVICE AIR

DESCRIPTION

This section covers the service air portion of the bleed-air system (Figure 36-10). The purpose of service air is to distribute regulated air pressure for the following functions:

- Horizontal Stabilizer Deice System—Service air is used to inflate the deice boots on the horizontal stabilizers.
- Vacuum Ejector Jet—Service air is used to operate the vacuum ejector jet, which serves as a vacuum source in the cabin pressurization control system.
- Door Seal Pressurization—Service air is used to pressurize seals on the cabin entry door.
- Throttle Detents (XL/XLS)—Service air is used to control the throttle detents in automatic and manual modes.

NOTES

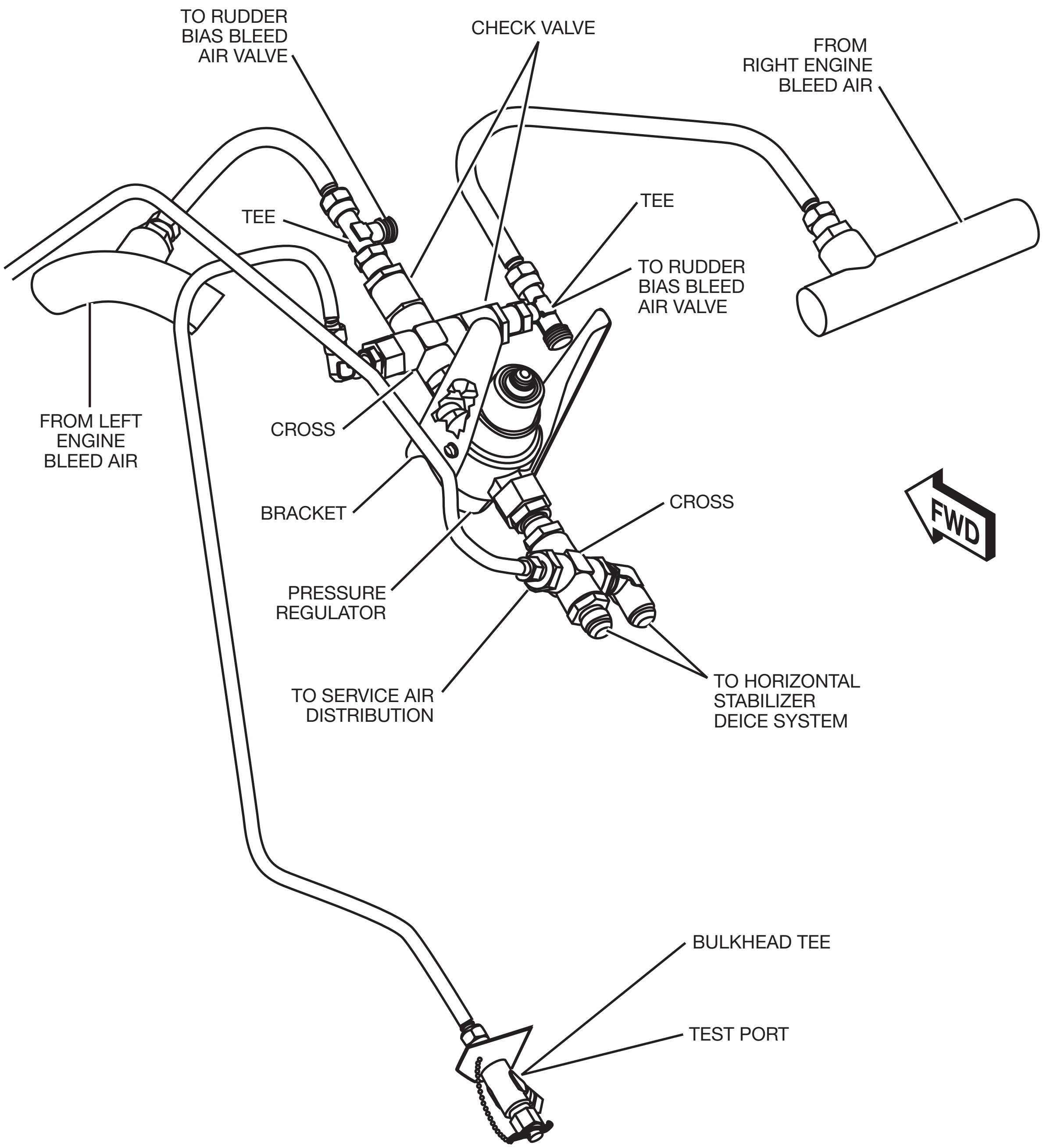


Figure 36-11. Service Air Supply and Regulation

Supply

Service air originates at the bleed-air tubes on the supply side of the ozone converters (Figure 36-11). The two bleed-air supplies are routed to check valves then into a cross fitting. The cross fitting is attached to the pressure regulator and the test port line. The test port is directly under the hydraulic reservoir (at FS 479.5). The test port is a capped tee-fitting used for connecting shop air to test or check the service air system. The regulator supplies pressure-regulated bleed air to another cross. This cross supplies regulated service air to the horizontal stabilizer deice boots and service air to the vacuum ejector jet on the primary outflow valve, the cabin door seals and the throttle detent system.

NOTES

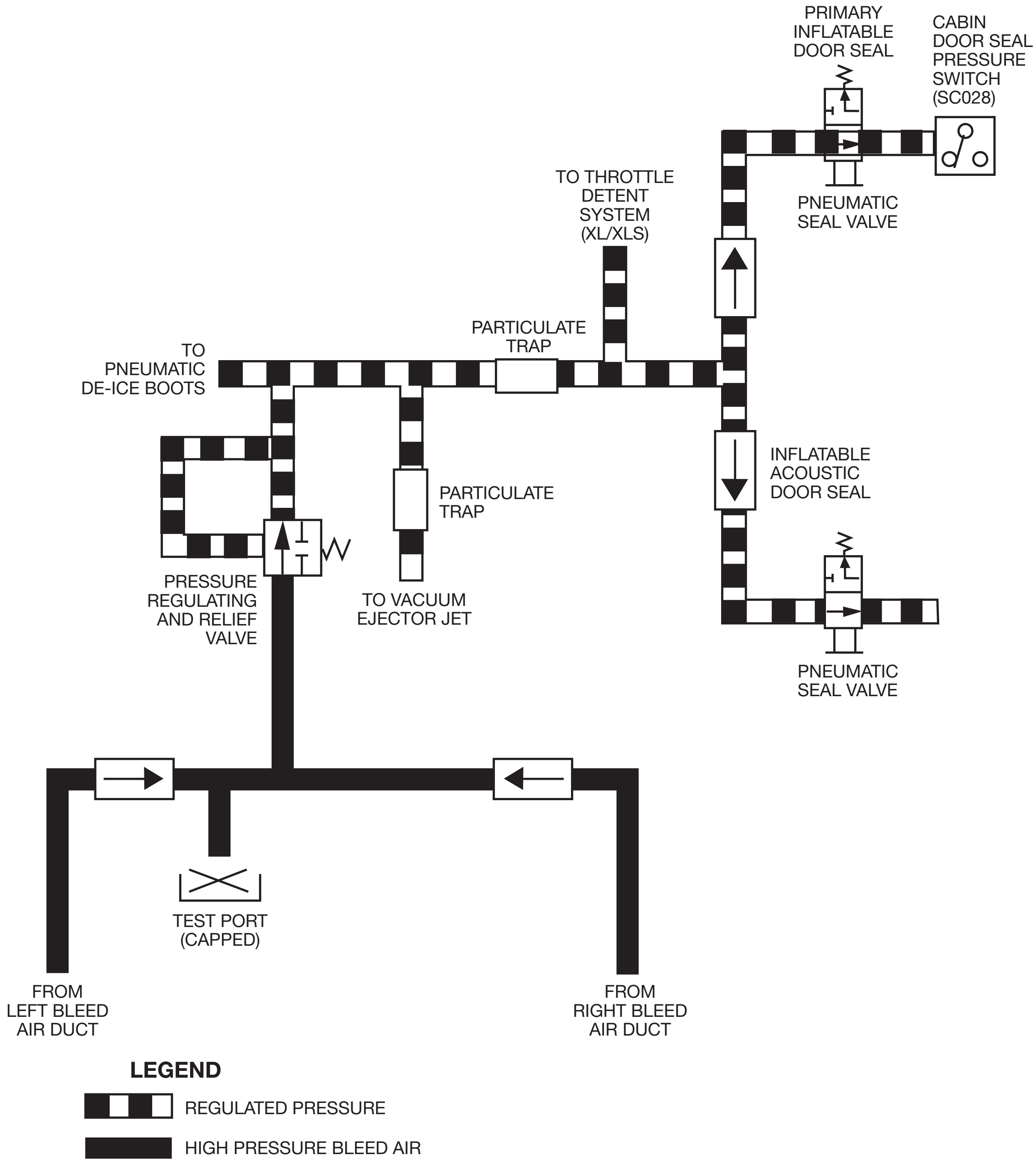


Figure 36-12. Service Air System Flow Diagram

Distribution

The service air line runs from the regulator forward to approximately FS 136.20 (Figure 36-12). This line penetrates the pressure vessel near the outflow valve area. The line tees off just forward of the aft pressure bulkhead to provide air to the vacuum ejector jet on the primary outflow valve. The line again tees off just aft of the cabin door under the left cabin floorboard to provide air to the door seal system. The door seal line also tees, providing service air to the primary inflatable door seal and the inflatable acoustic door seal. There are check valves in each line ahead of the respective door seal valve. The line for the throttle detent system (XL/XLS) continues on from the door seal system tee and runs under the left cabin floorboard to the throttle detent system (below the center pedestal).

The service air system contains two particulate traps downstream of the regulator, which trap any solid material before service air reaches the end-use systems. One is in the line going toward the primary outflow valve vacuum ejector jet. It is forward of the aft pressure bulkhead and below the secondary outflow valve. The other trap is in the line going toward the cabin door seal and throttle detent systems (XL/XLS). It is below the left cabin floorboard (aft of the cabin door).

COMPONENTS

Regulator/Relief Valve

The regulator/relief valve regulates the service air system pressure to 23 ± 1 psig. The relief valve prevents regulated pressure from exceeding 27 ± 1 psig in case of a regulator malfunction. The regulator is above the aft end of the baggage compartment on the left side of the airplane (at FS 461.50).

Vacuum Ejector Assembly

Service air is routed to the vacuum ejector jet for pressurization control. It provides a pressure source and generates a vacuum for the primary outflow valve solenoid operation. This will be discussed in further detail in Chapter 21—“Air Conditioning.”

NOTES

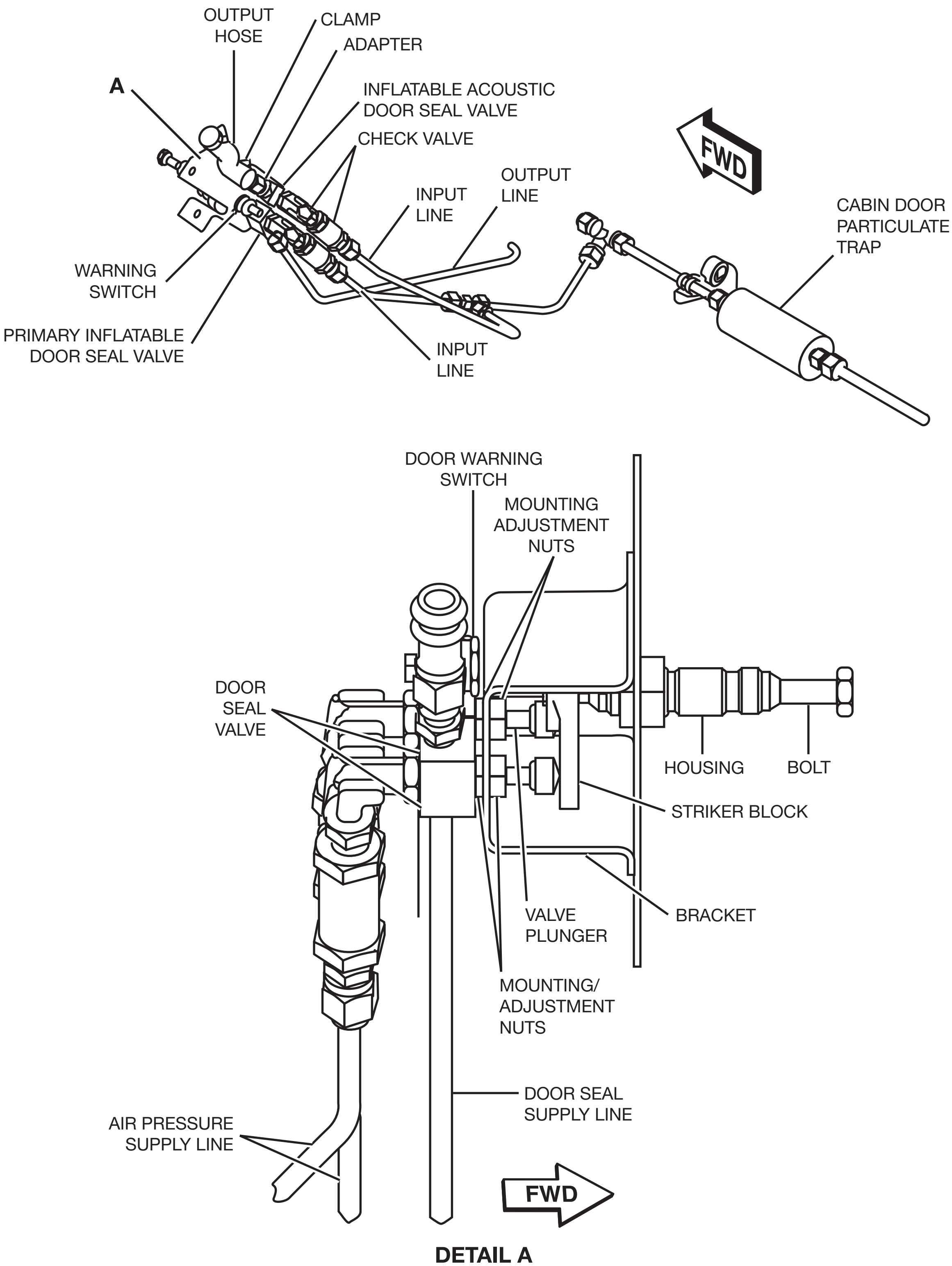


Figure 36-13. Entrance Door Seal System

Entrance Door System

The pneumatic entrance door seal system operates with service air (Figures 36-13 and 36-14).

The system consists of:

- Primary inflatable door seal valve
- Inflatable acoustic door seal valve
- Primary inflatable door seal
- Inflatable acoustic door seal
- Pressure switch

There are entrance door seal valves at the lower aft doorframe. The lower aft door lock pin actuates the valve in the door-locked position, allowing bleed air to inflate the door seal. There is a check valve in the input line connected to the primary inflatable door seal valve (and the input line to the inflatable acoustic door seal valve). This prevents loss of inflation pressure if the pneumatic air source is lost. When the door is unlocked, the spring-loaded valves deactivate, closing off bleed-air pressure and allowing air trapped in the seal to deflate through a vent in the valve body. A second valve, connected to the primary door seal, is actuated by the door handle mechanism, allowing the pressure in the seal to vent. This deflates the primary door seal.

A differential pressure switch (in the lower center of the entrance door) monitors door seal pressure. If door seal pressure drops below 5.5 ± .5 psid, the DOOR SEAL annunciator (XL/XLS) or the CABIN DOOR SEAL CAS message (XLS+), flashing along with the MASTER CAUTION warning light. Pressing the MASTER CAUTION light extinguishes it and the DOOR SEAL annunciator to illuminates steadily. The DOOR SEAL annunciator extinguishes when pressure in the door seal increases to 3 psi above the pressure at which the light illuminated.

DOOR SEAL
Annunciator steady on ground, flashes in flight, if the door seal pressure drops below 5.5 psi, activates MASTER CAUTION lights. Annunciator will extinguish if door seal pressure increases to approximately 8.5 psi.

XL/XLS ANNUNCATOR

CABIN DOOR SEAL			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard

This message is displayed when the pressure in the cabin door seal is less than 5 PSI. There is a pressure switch connected to the cabin door seal. Normally, the switch is closed, causing a ground to be presented to the EICAS system, which displays the message. When the pressure goes above 5 PSI, the switch opens, removing the ground input, which removes the message.

XLS+ CAS MESSAGES

Figure 36-14. Door Seal Annunciation

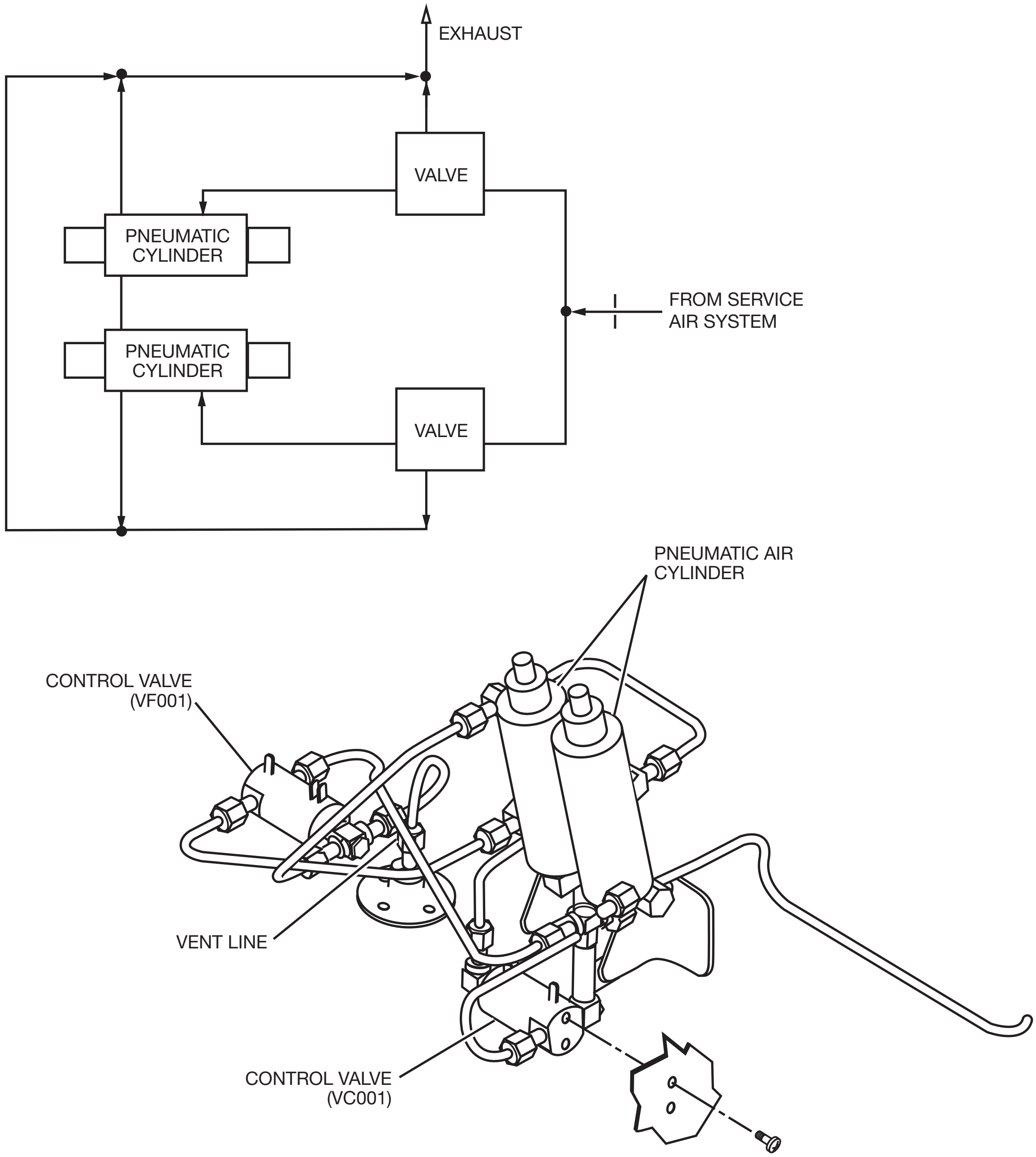


Figure 36-15. Throttle Detent System (Units 0001 through 0016)

Throttle Detent System (XL/XLS)

The throttle detent system consists of air-operated cylinders and valves that engage or disengage the throttle detents as commanded by the engine control system (Figure 36-15). Service air to the throttle detent system passes through an orifice to maintain the primary-door seal pressure in the event of pressure loss in the throttle detent system. The air-operated cylinders are below the throttle quadrant.

DIAGNOSTICS

Maintenance of the service air system consists primarily of replacing faulty components. Once maintenance has been completed, it is possible to test system integrity without engine operation by utilizing the shop air test port.

The pressure regulating valve is set to regulate bleed-air pressure at 23 ± 1 psig. The regulator pressure relief is set to release at 27 ± 1 psig. To perform an operational test of the regulator, connect a calibrated air-pressure gauge to the output side and apply air pressure to the input side. Adjustment of the regulator is performed in accordance with the manufacturer's component maintenance manual.

Bleed-Air Systems Test

1. Remove cap from the service air test port on the tee fitting directly below the hydraulic reservoir. Connect the shop air hose.

CAUTION

Shop air must be dry filtered air with a maximum pressure of 100 psig.

2. Connect a calibrated air pressure gauge to one of the cross ports on the outlet side of the pressure regulating and relief valve.
3. Check for leakage on the bleed-air check valves connected to the pressure regulating and relief valves and verify that shop air pressure is not flowing to the air conditioning system.

NOTE

Malfunction of the check valves may be caused by improper installation procedure. The check valves should be installed with the flapper valve hinge up.

NOTE

If a check valve has failed, the check valve must be removed and inspected for missing pieces. If pieces are missing from the check valve, the pieces must be located and removed from the pneumatic plumbing.

4. Upon completing the necessary system test(s), remove the shop air hose and pressure gauge and connect the distribution cluster to its proper configuration. Install the caps, remove tee fitting (if installed for test) and connect any bleed-air lines that were disconnected for the test.

QUESTIONS

1. Twenty-three psi regulated service air provides:

A. Cabin temperature control
B. Wing anti-ice capability
C. Pressurization vacuum
D. High pressure air to the ACM

2. Illumination of the DOOR SEAL annunciator is initiated by:

A. Cabin door seal valve
B. <5 psi pressure switch
C. Door locking microswitch
D. Door handle microswitch

3. The purpose of service air through vacuum ejector(s) is:

A. Provide cabin door seal vacuum
B. Provide vacuum for the pressurization system and the deice boots
C. Provide vacuum for the door acoustic seal
D. ACM water separator vacuum

4. Twenty-three psi service air provides operating pressure for:

A. Throttle detents and deicer boots
B. Standby gyro pressure
C. Emergency release of the gear uplocks
D. For the wheel brakes accumulator

5. Bleed air from the engines is precooled by:

A. Ram air on ground and in flight
B. Ram air in flight and fan bypass air on ground, on Excel units 5001 through 5372
C. Heat exchanger located in the tail cone ECU
D. Fan air in flight only

6. If the precooler fan air temp control valve fails closed on ground:

A. Bleed air will be too cool
B. System operates normally, due to the squat switch allowing ram air to be used
C. BLEED AIR O'HEAT L or R annunciator may illuminate and the respective wing anti-ice valve closes
D. Only the BLEED AIR O'HEAT annunciator illuminates

7. The 23 ± 1 pressure regulator regulates pressure to:

A. Pneumatic deice boots
B. Cabin door seal
C. Outflow valve ejector and throttle detents (XL/XLS)
D. All of the above

8. The primary door seal utilizes:

A. Two mechanical door seal valves; one in the door frame to direct 23 psi to the seal and one at the end of the seal to trap the pressure
B. One electric door seal valve, actuated when the door is closed
C. One mechanical door seal valve only
D. 23 psi from the APU only

9. Throttle detents (XL/XLS):

A. Use control valves which are energized with the EECs in manual
B. Use control valves which are deenergized with the EECs in auto
C. Use control valves which direct 23 psi to the detent cylinders
D. Operate with the EECs in manual only

CHAPTER 38
WATER AND WASTE

CONTENTS

ILLUSTRATIONS

CHAPTER 38
WATER AND WASTE

INTRODUCTION

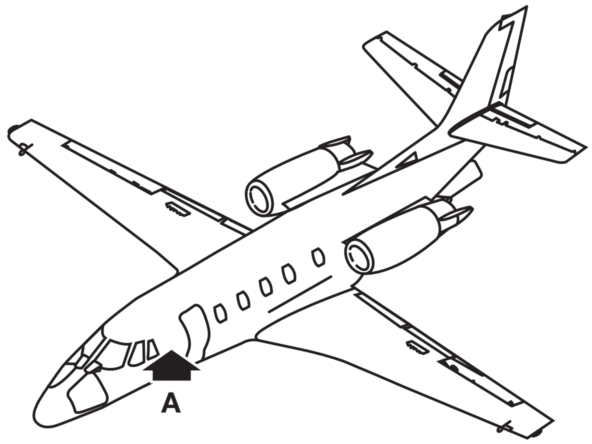
This chapter provides information on the water/waste system on the Citation XL/XLS/XLS+ aircraft. Information includes fixed units and components that store and deliver fresh water for use, and fixed components that store and furnish a means of removal of water and waste. Interior arrangement and options may cause variation in equipment description and installation.

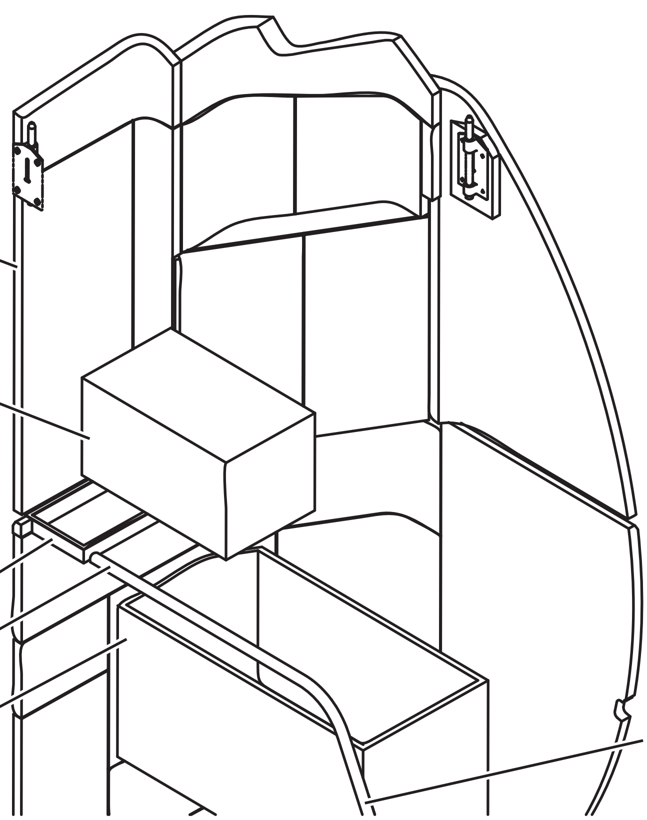
GENERAL

Potable water provisions are made available through the forward refreshment center in the left forward cabin compartment between FS 161.54 and FS 183.55 and the aft vanity/closet installation in the aft end of the cabin compartment at FS 370.60.

Wash water is available through a wash basin in the aft/vanity closet installation at the aft end of the cabin compartment at FS 370.60.

Waste disposal is accomplished with a flush-type toilet in the aft cabin on the left side of the aircraft. The flush toilet is self-contained to the extent that it requires only a 28 VDC electrical power source for operation.





DETAIL A

Figure 38-1. Forward Refreshment Center

POTABLE WATER

DESCRIPTION AND OPERATION

Potable water tanks are in the left forward refreshment center (Figure 38-1) and the aft vanity/closet area. The container is basically the same in each of the different cabinet styles available.

One stainless steel water tank is in the left forward refreshment center and one in the aft vanity/closet area. The stainless steel water tanks are electrically heated to provide hot water for hot drinks. An ice drawer stores ice for cooling drinks.

Access to the water tanks is usually made by removing a cover plate or lid, depending on cabinet style.

The water tanks are drained through a pushbutton-type drain valve. The valve assembly is spring-loaded in the closed position. The hot water tanks have a latch on each end so the cover may be removed.

The hot water tanks have a receptacle that connects with a mating plug at the back of the shelf. When the tank is in place, the circuit is completed to allow the heat to be turned with a switch on the front of the refreshment center.

NOTES

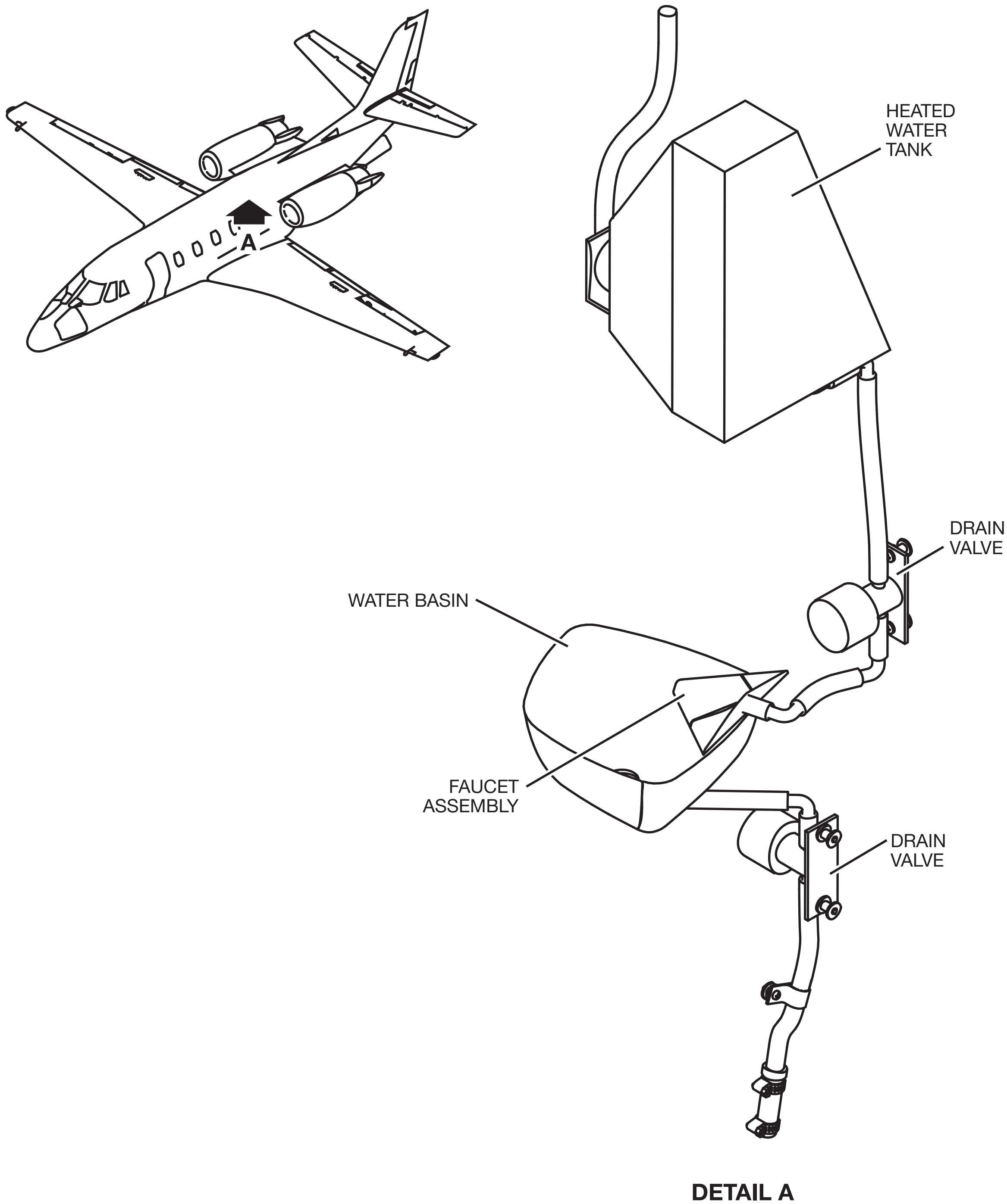


Figure 38-2. Vanity Water Assembly

WASH WATER

DESCRIPTION

Hot and cold running water is available through the aft vanity wash basin. The vanity may or may not incorporate a closet. Vanities without a closet have a water system in a drawer below the wash basin (Figure 38-2). Vanities with a closet have the water system behind a panel in the closet. For servicing the water storage storage tanks and for freeze protection of the water system in cold weather, refer to Chapter 12—"Vanity Water Supply—Servicing" in the *Aircraft Maintenance Manual (AMM)*.

COMPONENTS

Vanity

The typical vanity water system for the wash basin includes:

- Hot and cold water storage tanks
- Two electrically driven pumps
- Pump switch
- Hot water heater (incorporated in the hot water storage tank)
- Heater switch and relay
- Voltage regulator
- Two high-pressure switches
- Two low-pressure switches
- Two time-delay relays

The location of the water system components depends on whether the vanity has a closet or not. Electrical power for the vanity originates at the interior junction box. Circuit breaker CB146V is connected to the terminal board TB29V at the vanity. A separate CB4F (WATER) circuit breaker at the vanity protects the hot water heater power circuit while the control circuit for the heater and water system are protected by a common circuit breaker. The 12-volt electrical system, including the pumps, is protected by 4-amp fuses.

Vanity Closet

The aft vanity/closet incorporates a gravity-feed water system for the wash basin. The water storage tank is in the upper right side of the aft vanity/closet and connected to the faucet with a hose. A quick-disconnect allows separation of the tank from the water disconnect without loss of water. The water tank also has an electrical disconnect that separates when the tank is removed.

A pressure transducer is in-line mounted in the water hose and is electrically connected to the water level indicator on the temperature controller. Temperature sensors in the water tank are also electrically connected to the temperature controller on the vanity.

Other electrical components for controlling the water temperature and the indicating systems include:

- Heater power relay
- Overheat relay
- Pressure transducer
- Signal conditioner in the compartment below the water tank

The water hose quick-disconnect at the faucet permits separation of the hose from the faucet during removal of the vanity or vanity top.

The hot and cold water storage tanks in the aft vanity/closet have a capacity of 2.00 gallons (7.75 liters) each. Vanities without closets have a single storage tank with a capacity of approximately 1.45 gallons (5.5 liters).

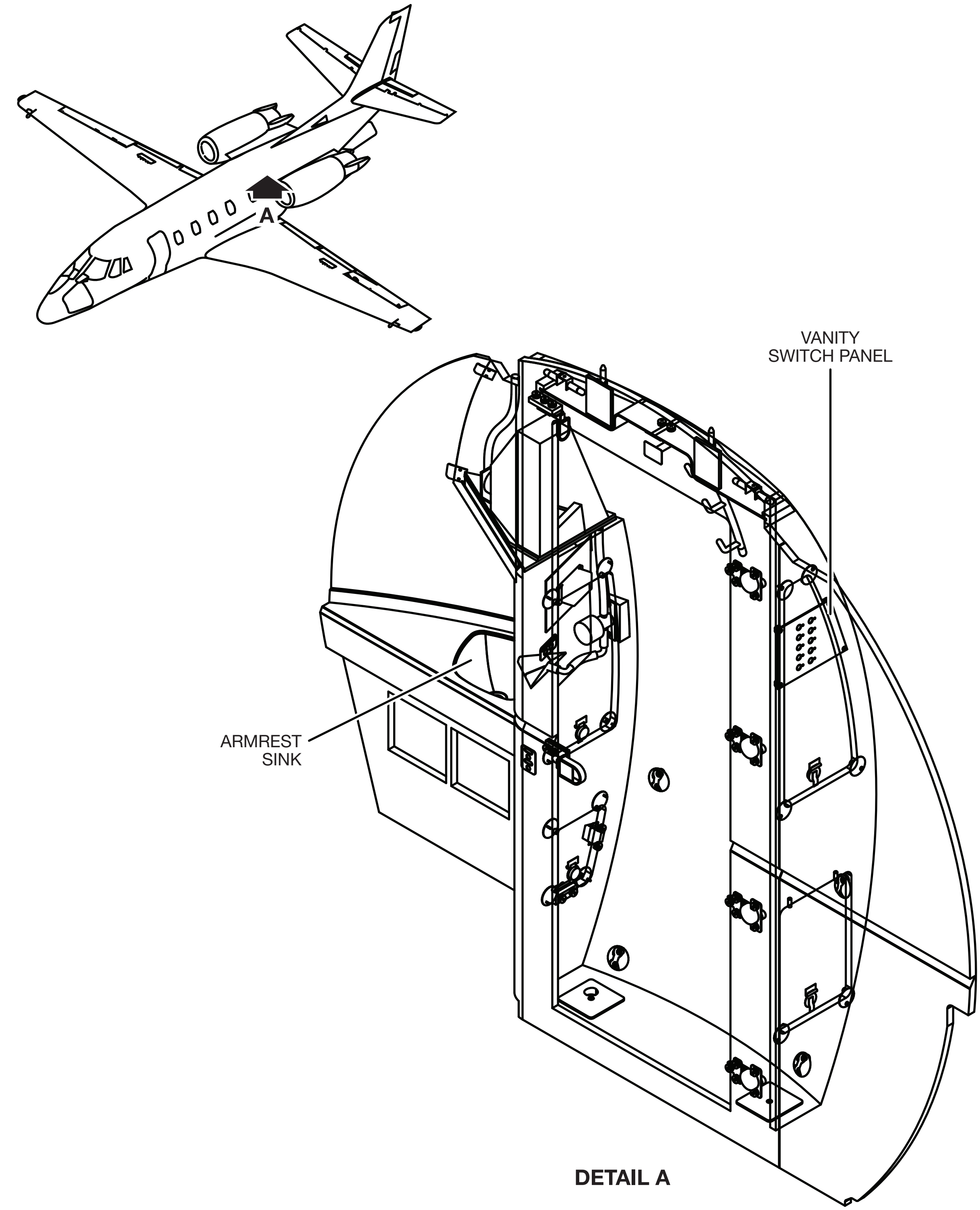


Figure 38-3. Vanity Switch Panel

Aft Bulkhead Closet With Armrest Sink

Maintenance of the aft bulkhead closet with armrest sink consists mainly of removal/installation of the components for heating and controlling the wash water temperature. The electrical components for heating and controlling the wash water are the water switch and vanity switch panels (Figure 38-3) mounted on the front of the aft bulkhead closet and above the sink, including the hot water tank power relay, sink drain relay, water control valve, and water drain valve. The timer control relay is in the compartment below the hot water storage tank. The heating elements and temperature sensors are incorporated into the water tank.

OPERATION

The water heater is controlled with the HEATER switch. The switch illuminates when the aircraft is powered with 28 VDC. When the HEATER switch is turned on, the switchlight extinguishes and three power lights illuminate indicating that electrical power is being applied to the heater relay. With the heater relay energized, 28 VDC activates the heater element. The hot water tank heater has the capability of heating 60 to 70°F (16 to 21°C) water to 130°F (54°C) in approximately 15 minutes. It is thermostatically controlled to cycle and maintain 130°F (54°C) until turned the HEATER switch is turned off.

Operation of the water system is controlled by the PUMP switch. With 28 VDC power on the aircraft, the switchlight is illuminated. Turning the PUMP switch ON extinguishes the switchlight and three power lights illuminate indicating 28 VDC is being applied to the voltage regulator, and 28 VDC is being applied to the two time-delay relays through the two low-pressure switches respectively.

The voltage regulator changes 28 VDC to 12 VDC to operate the hot water and the cold water pumps through the high-pressure switches. The pumps are provided a ground circuit through their respective time-delay relay. The high-pressure switches open at a water pressure of 6 psi (41.4 kPa) to shut down the respective pump. When the water pressure drops to 3 psi (20.7 kPa) or below, the low-pressure switch opens to cut off 28 VDC to the respective time-delay relay. If water pressure does not increase to 3 psi (20.7 kPa) or above within 6 seconds (such as a holding tank out of water), the time-delay relay opens to shut off the respective water pump. Each pump is protected by a 4-amp fuse in case of a short in the pumps or the pump electrical circuit.

If replacement of the water pump(s) is necessary, use the plastic fittings supplied with the pump to prevent breakage of the pump housing. Wrap the threads of the fittings with teflon tape and do not overtighten the fittings.

The operation of the gravity-feed water system includes instant water flow at the faucet, water temperature control, water temperature digital indicator, and water level bar-type indicator. For additional information on the aft vanity/closet electrical system, refer to the *Model 560 EXCEL Wiring Diagram Manual*.

The gravity-feed water system incorporates a single storage tank with approximately 2-gallon capacity and is hose connected to the aft vanity/closet wash basin faucet.

The temperature controller incorporates the controls for the water heater and vanity lights. The HEATER switch energizes the heater power relay through the temperature controller. Temperature sensors (thermal switches) in the storage tank maintain the water temperature as set with the control on the face of the controller. If an overheat condition exists, the controller energizes the overheat relay to disconnect heater power to the heating elements. The aft vanity/closet lights are controlled by the vanity switch panel on-off switch and a dimming control knob on the controller.

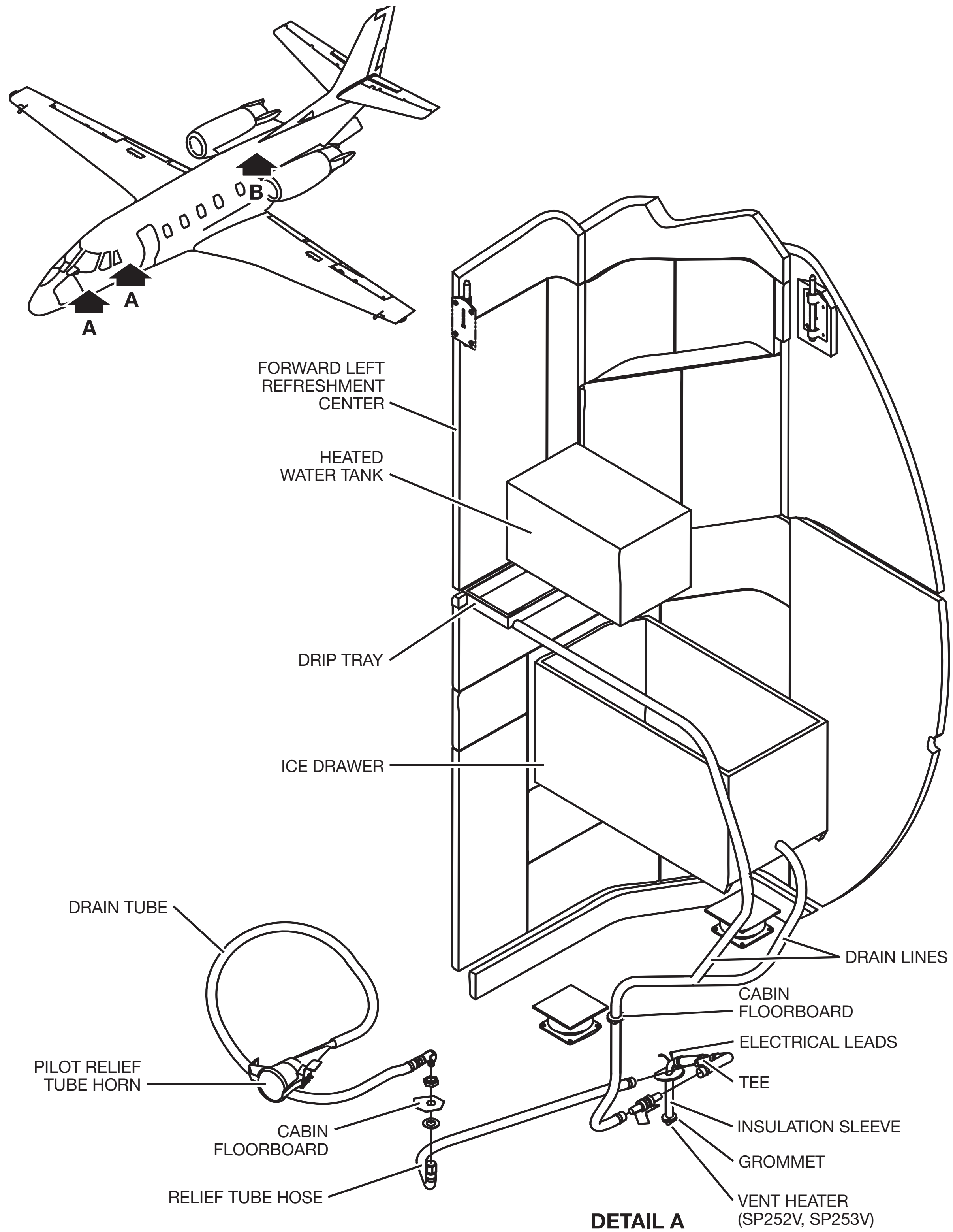


Figure 38-4. Water Disposal System (Sheet 1 of 2)

The temperature of the water displays digitally on the face of the controller. Also on the face of the controller is a lighted vertical bar scale to indicate the water level in the storage tank. The water level indicator receives signals from the pressure transducer through the signal conditioner unit. As the water level in the tank drops, the lighted bar proportionally lowers.

WATER/WASTE DISPOSAL

DESCRIPTION

This section describes the water disposal system (Figure 38-4) of the:

- Potable water from the left forward refreshment center
- Wash water from the aft vanity

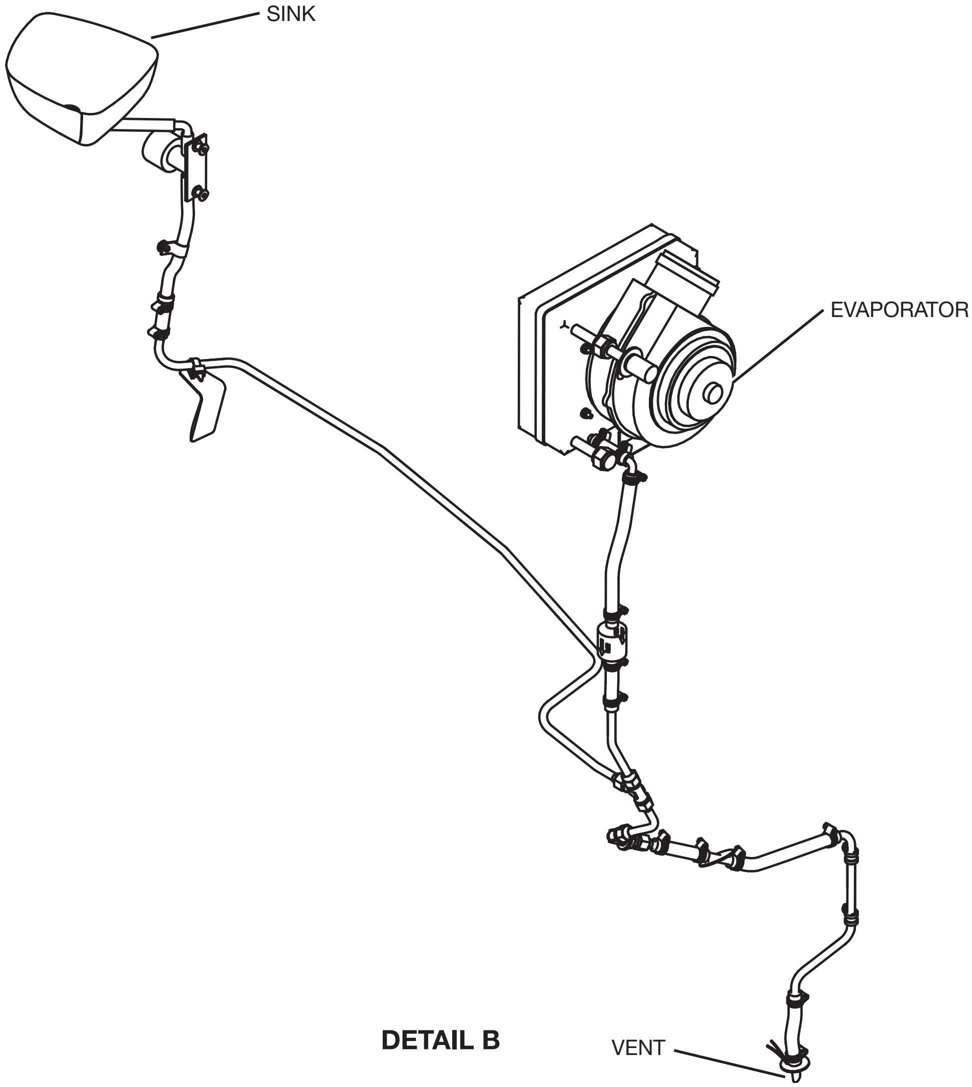


Figure 38-4. Water Disposal System (Sheet 2 of 2)

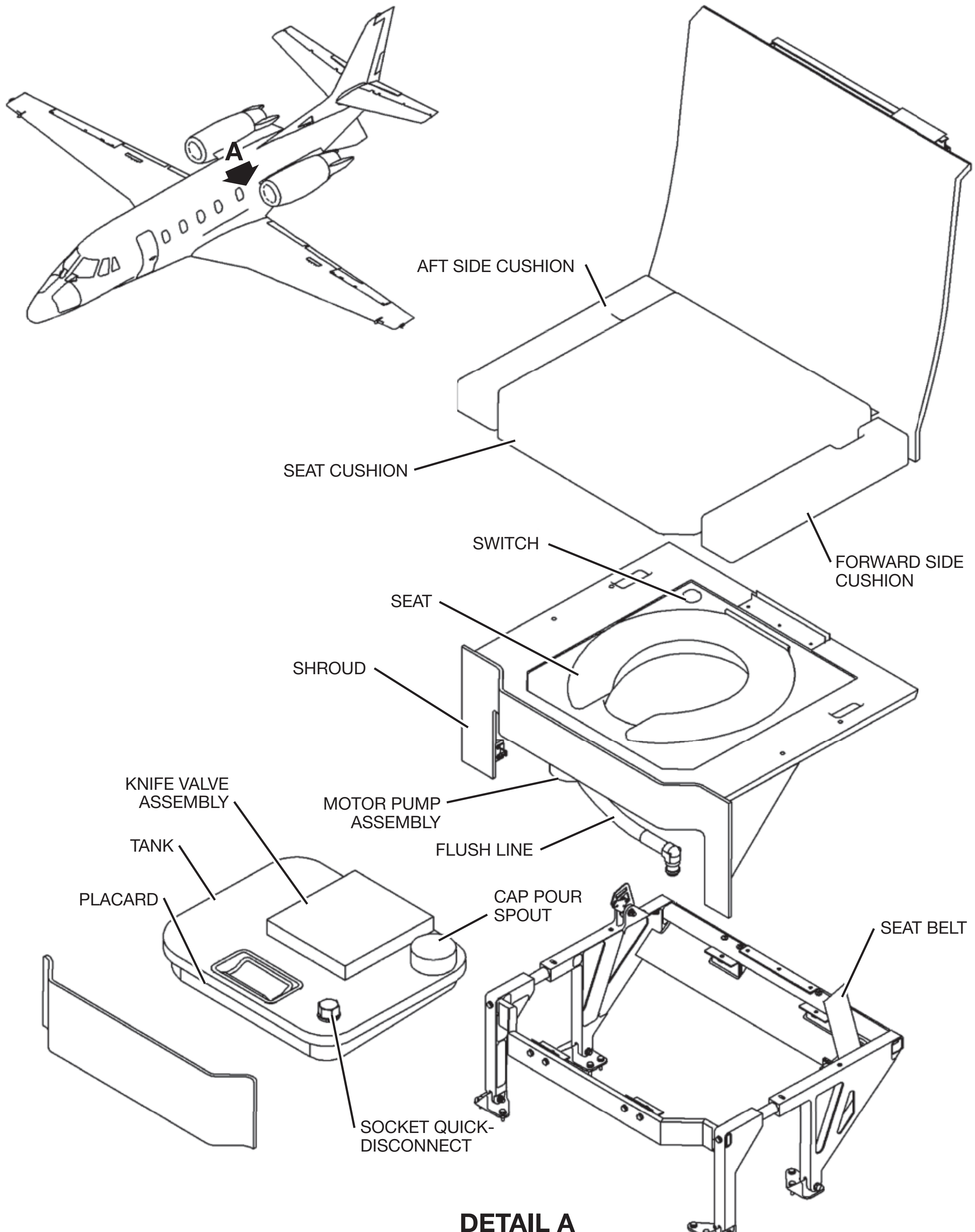


Figure 38-5. Carry Out Flush Toilet Assembly

Information on maintenance and cleaning of the waste disposal equipment is included as well as information on the aft toilet. Maintaining the equipment in a serviceable, clean, and sanitary condition is essential.

The left forward refreshment center has a drain system from the tray below the hot water tank and cup dispenser and a drain below the ice drawer to the outside through a heated outlet. The water from the wash basin in the aft vanity is drained through the bottom of the airplane through a heated outlet. Refer to Chapter 6—"Airplane Drains Location" and "Drains Location" in the *AMM*.

COMPONENTS

Aft Carry-Out Flush Toilet

The aft carry-out flush toilet (Figure 38-5) is in the aft cabin compartment on the left side of the aircraft between FS 345.06 and FS 370.50. A placard with servicing instructions is on the side of the toilet assembly. Servicing instructions are also given in Chapter 12—"Servicing" of the *AMM*.

The aft carry out flush toilet unit is a light weight, compact, electrically operated flush toilet, which utilizes ordinary tap water and a germicidal deodorant.

The standard seating option includes a aft carry out flush toilet in the aft cabin compartment on the left side of the aircraft between FS 345.06 and FS 370.05. This toilet is equipped with lap and shoulder belts and is approved for takeoffs and landings. A life vest is stowed in the aft bulkhead closet.

The aft carry-out flush toilet is a completely self-contained unit, requiring only the external connection of 28 VDC electrical power. The toilet assembly is permanently installed in the aircraft, requiring only the removal of the holding tank when servicing is desired.

The aft carry-out flush toilet assembly consists of a seat and shroud assembly, which rest on the toilet mounting plate. Attached to the mounting plate are the polished stainless steel bowl assembly, the motor and pump assembly and the PRESS TO FLUSH switch. Mounted to the bottom flange of the bowl is the slide assembly into which the removable holding tank assembly is installed.

NOTES

The removable holding tank assembly consists of a storage tank on which the knife valve, flush line quick-disconnect, and carrying handle are located. Extending through the cover of the knife valve is a manually operated actuator to open or close the knife valve, sealing the tank contents prior to removal of the tank. The position of the knife valve may be observed through the opening at the bottom of the bowl.

The holding tank assembly detaches from the toilet at the front of the unit. Two Pres-Loc fasteners, one on each side of the knife valve, secure the installed tank in the sealed position against the bottom of the bowl. By detaching and draining the flush line at the quick-disconnect, depressing the two Pres-Loc fasteners, and pulling the carrying handle, the tank is easily removed for servicing.

The flush cycle is initiated by pressing the PRESS TO FLUSH button on the seat and shroud assembly. The push button switch applies 28 VDC power to the motor section of the motor and pump assembly. Flushing continues until the push button is released. During the flush cycle, flushing fluid is pumped from the holding tank to the bowl by the self-priming pump section of the motor and pump assembly. The flush fluid enters the bowl through a nozzle in the upper rim and washes the inner surface of the bowl in a swirling pattern. Waste is carried to the holding tank through the knife valve below the bowl. When desired, the removable holding tank may be removed from the toilet for servicing after closing the knife valve.

The aft carry out flush toilet is located in the aft cabin compartment on the left side of the between FS 345.06 and FS 370.50. A placard with servicing instructions is located on the side of the toilet assembly. Servicing instructions are also defined in Chapter 12, Aft Carry Out Flush Toilet–Servicing.

Operation of the externally serviceable flush toilet is accomplished with 28 VDC electrical power. The toilet has a PRESS to FLUSH button and motor-pump assembly. Servicing the externally serviceable flush toilet is accomplished by using a ground service cart that cleans and replenishes the toilet with a chemical solution. Servicing the aft flush toilet (carry out) is accomplished by removing the holding tank and cleaning and replenishing it with a chemical solution.

NOTES

Aft Carry Out Flush Toilet

Troubleshooting

Troubleshooting procedures are given as an aid in isolating and identifying malfunctions of the aft flush toilet. Refer to the *560 Excel Wiring Diagram Manual*.

Adjustment/Test

Operational test:

1. The flush cycle is initiated by pressing the PRESS TO FLUSH button on the seat assembly.
2. The pushbutton switch applies 28 VDC to the motor section of the motor and pump assembly.
3. Flushing continues until the pushbutton is released.
4. During the flush cycle, flushing fluid is pumped from the holding tank to the bowl by the self-priming pump assembly.
5. The flush fluid enters the bowl through a nozzle in the upper rim and washes the inner surface of the bowl in a swirling pattern. Troubleshooting procedures are given as an aid in isolating and identifying malfunctions of the aft flush toilet.

NOTES

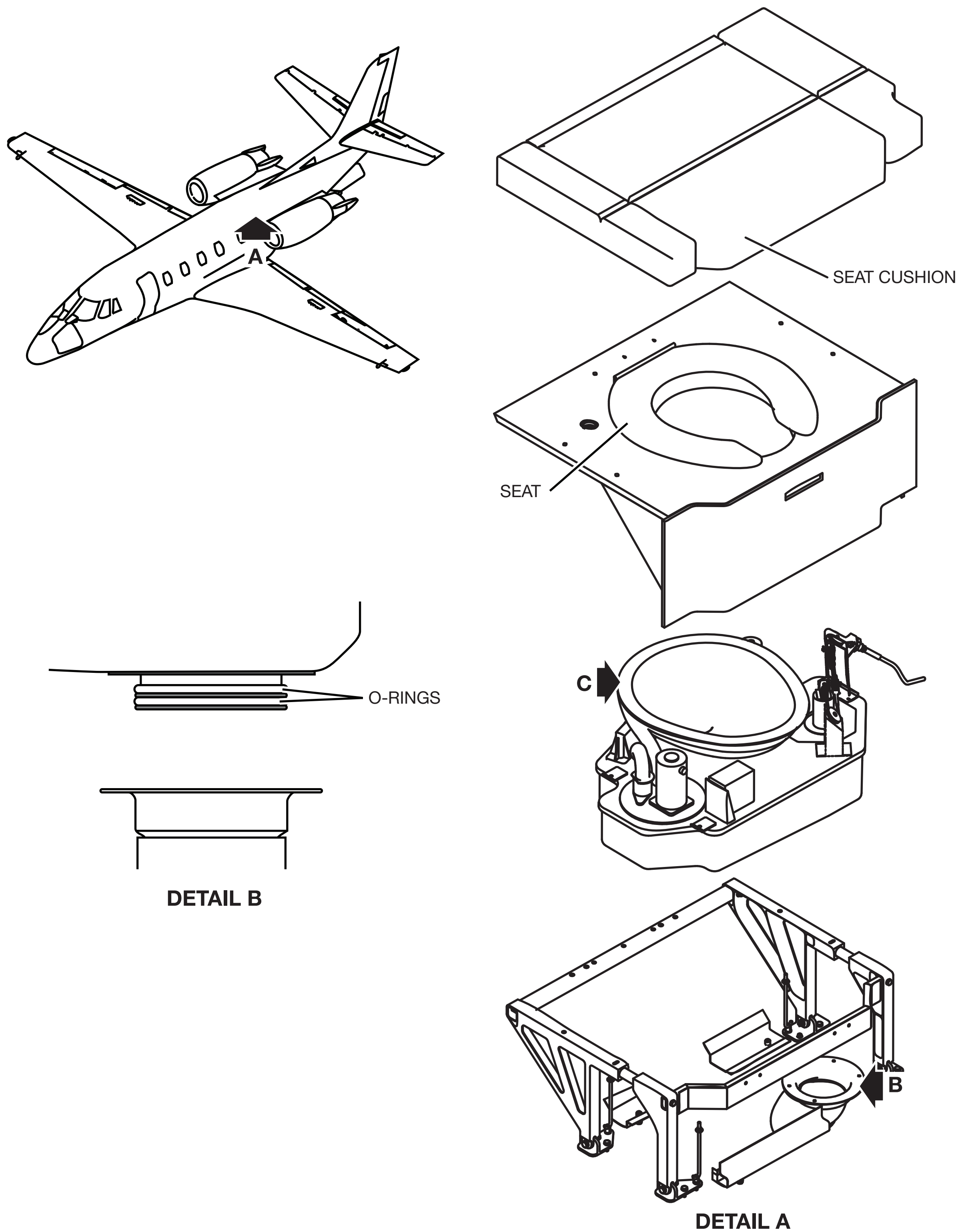


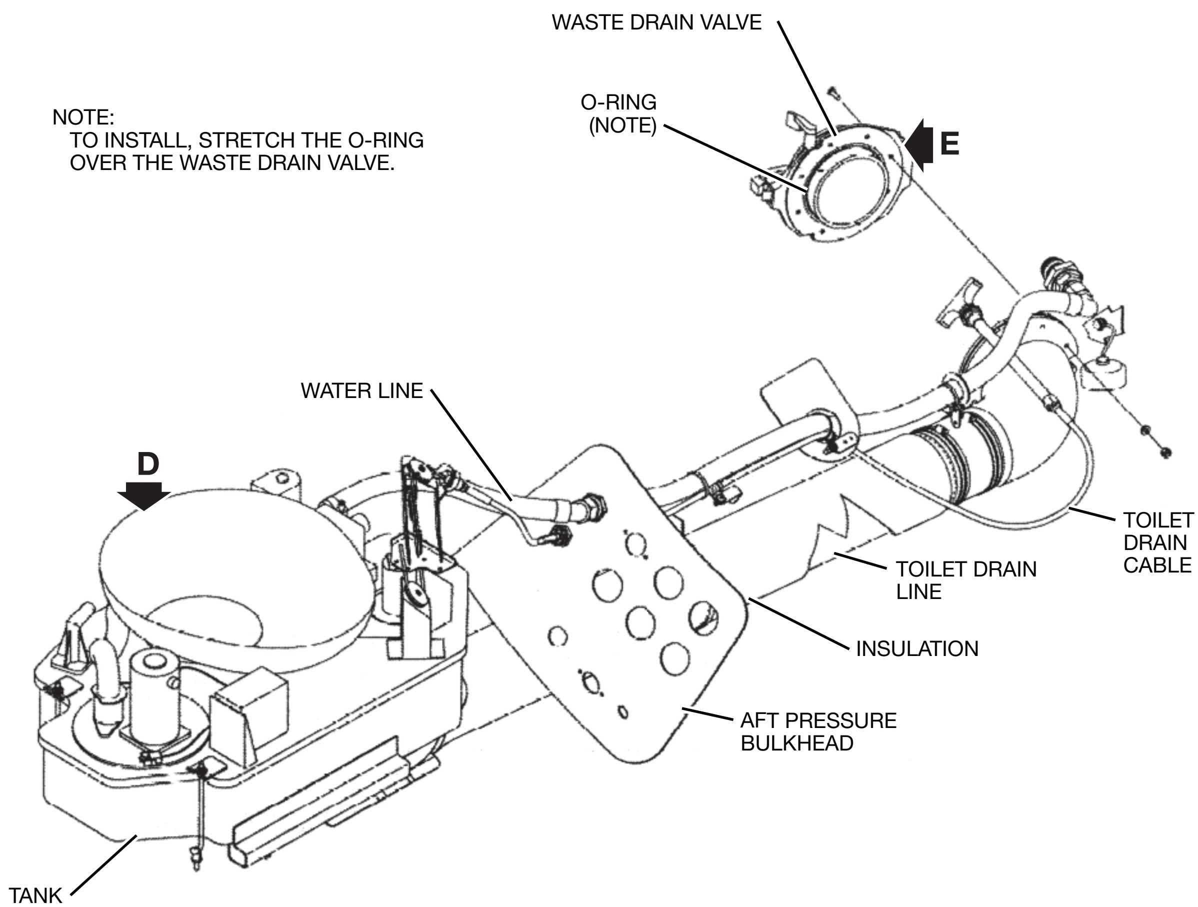
Figure 38-6. Externally Serviceable Toilet Assembly (Sheet 1 of 3)

Externally Serviceable Flush Toilet

The externally serviceable flush toilet unit is a compact, electrically operated flush toilet that utilizes ordinary tap water and a germicidal deodorant (Figure 38-6). The externally serviceable flush toilet is in the aft cabin compartment on the right side of the aircraft between FS 345.06 and FS 370.50. The externally serviceable toilet is a self-contained unit requiring 28 VDC electrical power and connects with the external service panel. The toilet assembly is removable for maintenance and is serviced through the service panel.

The toilet is mounted under a removable seat frame that supports a shroud, toilet ring, and a cushioned seat.

The external service panel is on the right side of the fuselage below the engine nacelle between FS 408.45 and FS 422.03. The service access door opens to expose the service valve assembly, a water service connection, and a flush handle.



DETAIL C

Figure 38-6. Externally Serviceable Toilet Assembly (Sheet 2 of 3)

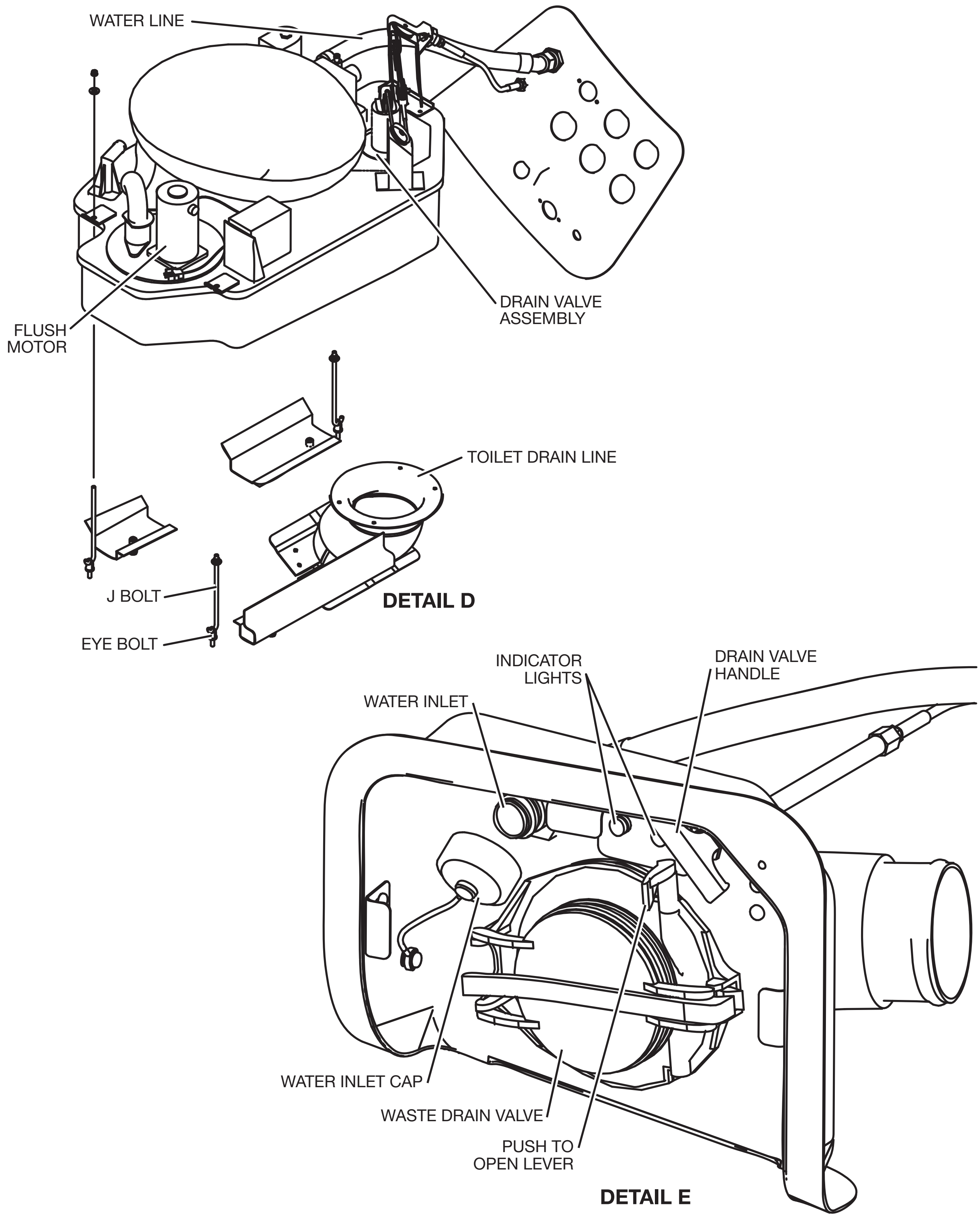


Figure 38-6. Externally Serviceable Toilet Assembly (Sheet 3 of 3)

The flush cycle is initiated by pressing the PRESS TO FLUSH button next to the toilet seat. The flush cycle lasts approximately 8 seconds. During the flush cycle, flushing fluid is pumped from the holding tank to the bowl through a nozzle in the upper rim, washing the bowl in a swirling pattern. Waste is stored in a holding tank below the bowl.

The unit is serviceable with a portable ground unit. Prior to removal of the toilet assembly, it must be serviced externally and be empty of any flushing fluid or water. A placard with service instructions is located on the external service panel. For servicing information, refer to Chapter 12—“Servicing” in the *AMM*.

For maintenance of the externally serviceable toilet components, refer to the vendor’s manual in the list of *Manufacturer’s Technical Publications*.

Externally Serviceable Flush Toilet Adjustment/Test

Flush Cable Adjustment and Tension Test

1. Adjust the flush cable travel at turnbuckle. The pull handle should extend out 4.50 inches (114.3 mm).
2. Check the toilet handle pull. Handle pull should be 50 ± 10 pounds (222.4 ± 44.5 N).

Toilet Operation Check

1. Place battery switch on the DC POWER subpanel to the ON position. Flush the toilet a minimum of three flush cycles. Each flush cycle should last approximately 8 seconds.
2. Place battery switch to the OFF position.

Indicator Level Lights and Automatic Water Shut-off Test

1. Turn inlet water on and fill the tank with 1.5 ± 0.1 gallons (5.7 l ± 0.38 l) of water. Check if the precharge level light is activated.
2. Continue to add water to fill toilet tank.
   a. Overfill level light activates between 2.0 and 4.0 gallons (7.6 and 15.14l) of water.
   b. Automatic shutoff valve deenergizes.
   c. Water shuts off preventing tank from overfilling.
   d. Turn the inlet water off.
3. If the automatic shutoff does not activate after 4.0 gallons (15.14 l) of water has been added to the system, stop filling to avoid flooding the aircraft. This indicates that the overfill protection feature has a problem.

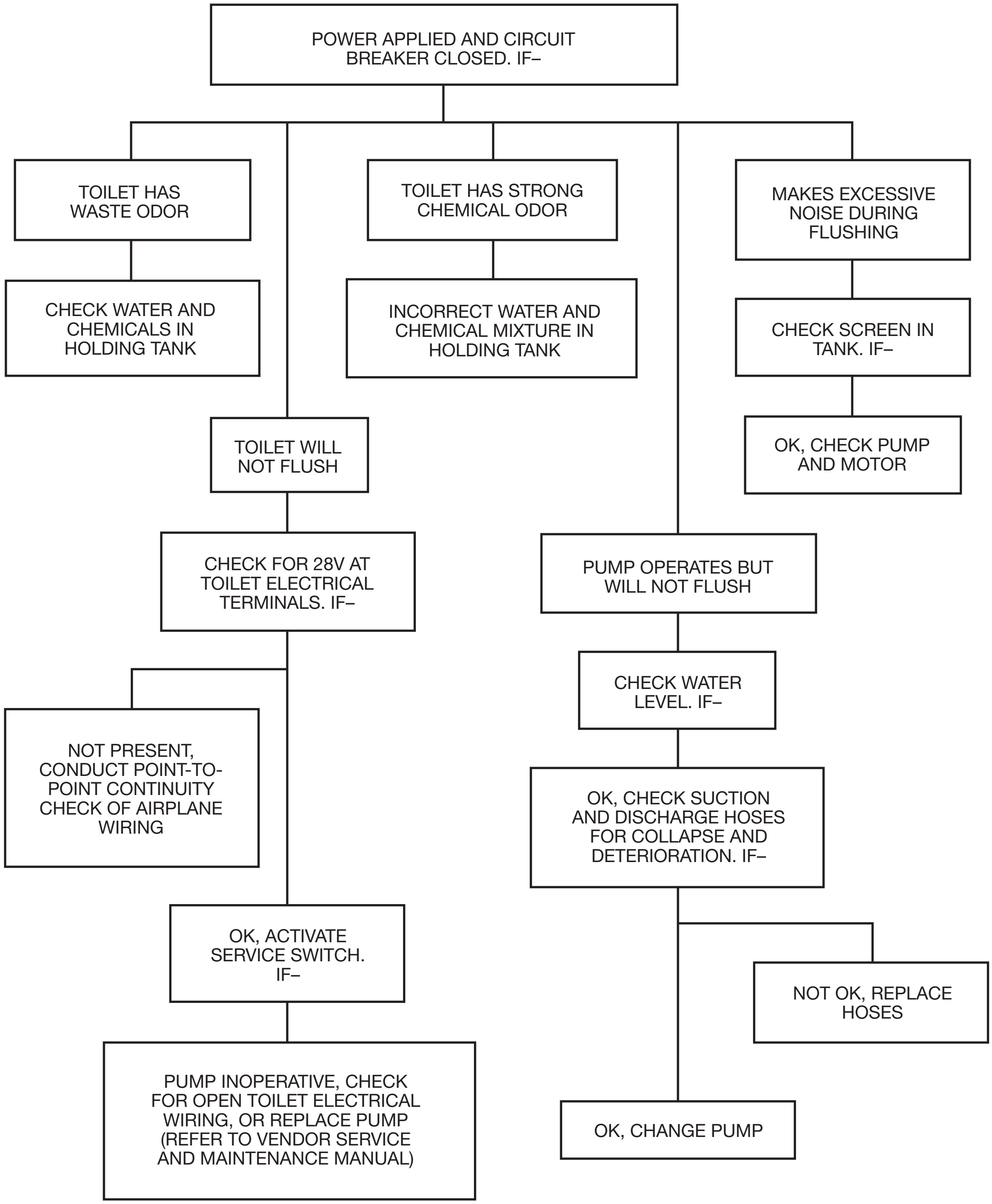


Figure 38-7. Troubleshooting Chart

DIAGNOSTICS

Drain Line Pressure Test

Externally Serviceable Toilet Drain Line Pressure Test

Refer to Figure 38-7.

1. Remove the toilet.
2. Fill the drain line with water to a level within 4 ± 1 inches (114 ± 25 mm) from the top edge of the drain line opening.
3. Use screws to attach the CJMD338-101 toilet drain line plug and plate to the structure.
4. Attach a pressure regulator and pressure gage to the threaded hole in the drain line plug.
5. Use 9.3 ± 0.5 psi regulated air to pressurize the drain line.
6. Hold the pressure for 5 minutes.
7. Make sure that there are no leaks at any point along the drain line.
8. Release the air pressure from the drain line.
9. Remove all the test equipment.
10. If leaks are found, replace the unserviceable component before the toilet is used again.
11. Install the toilet.

Alternate Drain Line Pressure Test:

NOTE

It is not necessary to remove the toilet assembly to complete this procedure. It is permitted to complete this procedure at the same time as Task 21-30-00-710.

1. Remove all the contents of the toilet tank.
2. Rinse the toilet several times.
3. Close the inner flapper valve on the drain.
4. If connected, disconnect the ground service unit and close the outer toilet drain valve.
5. Pull the yellow T-handle and leave the toilet plug open.
6. Fill the toilet drain line with water.
7. Complete Task 21-30-00-710, Pressurization System Operational Test.
8. Complete an inspection of the drain line assembly and make sure there are no leaks from the plug to the service panel.
9. Perform the servicing of the toilet. Refer to Externally Serviceable Flush Toilet–Servicing.

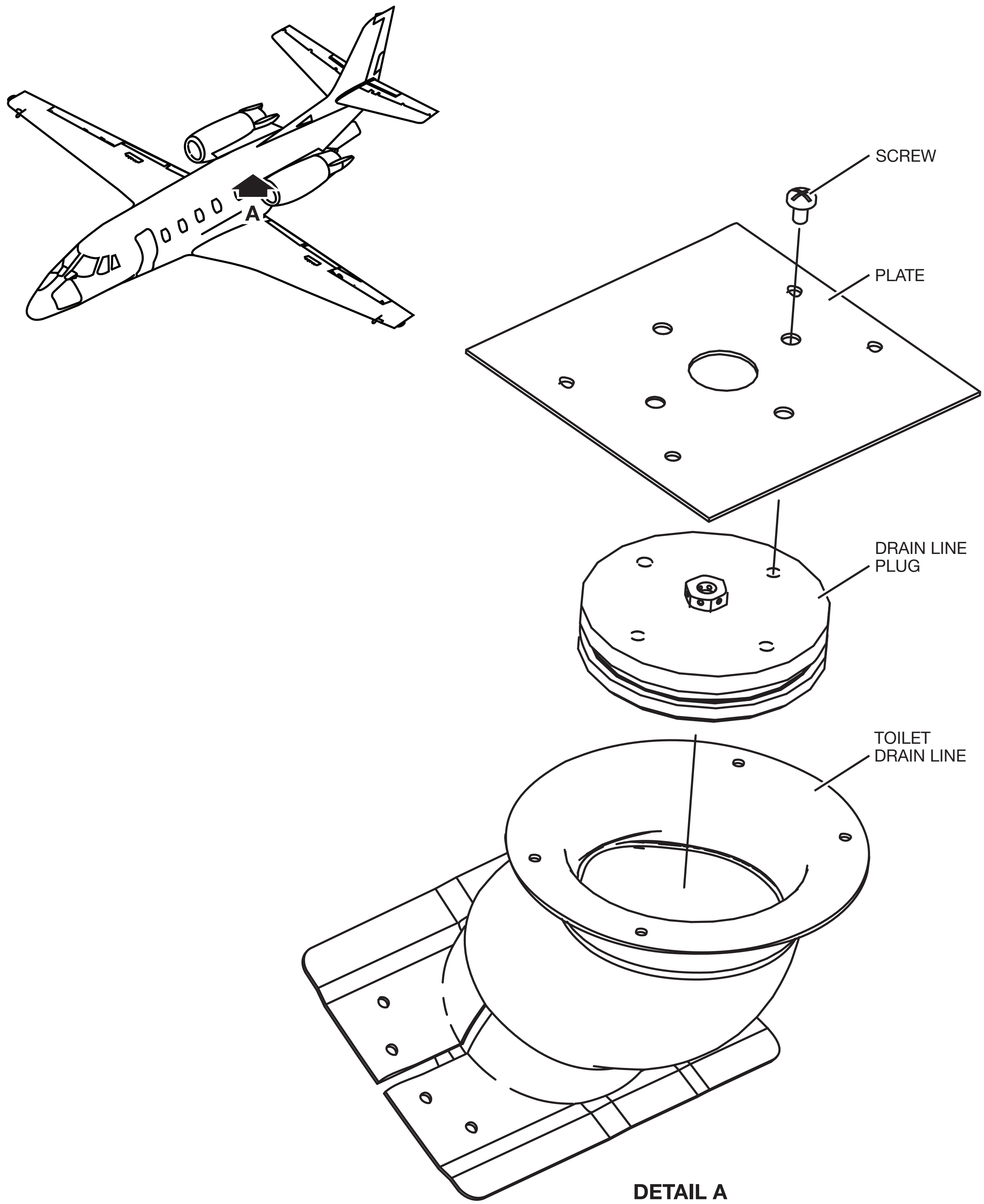


Figure 38-8. Toilet Drain Line Assembly

Drain Line Operational Check

Refer to Figure 38-8.

- Connect external power to the aircraft.
- Fill the refreshment center potable water dispenser and ice chest drip pans with clean water.
- Press the DRAIN button and verify that the water drains overboard.
- Fill the vanity sink with clean water.
- Press the DRAIN button and verify that the water drains overboard.
- Remove external power from the aircraft.

NOTES

CHAPTER 45
CENTRAL MAINTENANCE SYSTEM

CONTENTS

ILLUSTRATIONS

CHAPTER 45
CENTRAL MAINTENANCE SYSTEM

INTRODUCTION

This chapter describes the central maintenance system which is an integrated part of the Collins Pro Line 21 Avionics Suite on the XLS+ aircraft . Information is provided on the procedures to look at the maintenance messages available through the Collins Pro Line 21 system.

GENERAL

The Collins Pro Line 21 system messages are shown directly on the multifunction displays (MFDs). For information on access and use of the maintenance diagnostic computer (MDC) refer to "Collins Maintenance Diagnostic Computer System—Description and Operation" in the *Aircraft Maintenance Manual (AMM)*.

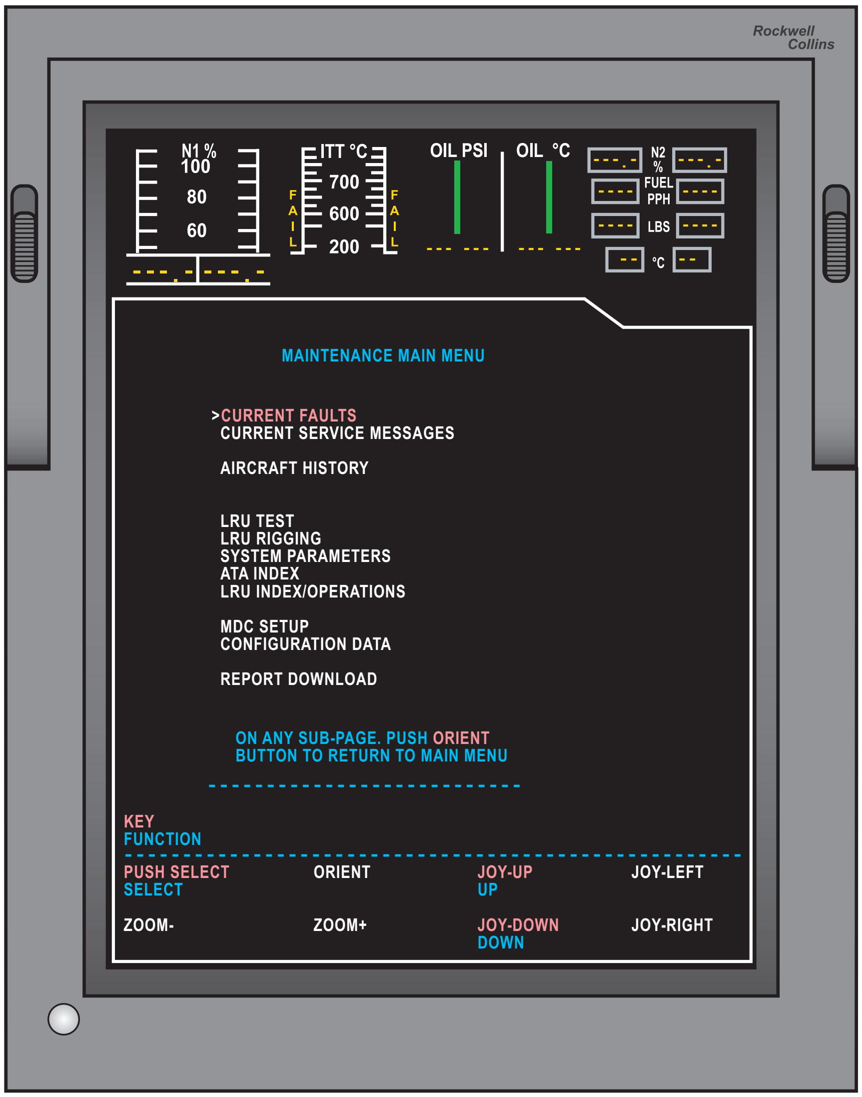


Figure 45-1. Maintenance Main Menu

DESCRIPTION

The Collins MDC system monitors many of the aircraft components to find failures, isolate faults, and give you historical fault data. The MDC monitors the following aircraft components:

- Air data computers (ADC)
- Automatic direction finders (ADF)
- Attitude heading computers (AHC)
- Control display units (CDU)
- Display control panels (DCP)
- Distance measuring equipment (DME)
- Full authority digital engine controls (FADEC)
- Enhanced ground proximity warning system (EGPWS)
- Traffic and collision avoidance system (TCAS 11)
- Flight guidance computers (FGC)
- Flight management computers (FMC)
- Global positioning system (GPS)
- High frequency (HF) and very high frequency (VHF) communications systems
- Integrated avionics processor system (IAPS)
- Primary flight displays (PFD)
- MFD
- Radio altimeter
- Navigation systems (NAV), weather radar (WXR)
- Stormscope
- Radio tuning units (RTU)
- Radio interface units (RIU)
- Audio control panels (ACP)

The Collins MDC system monitors the data that it receives from the left and right IOC-4110 input/output concentrator (IOC) Modules in the IAPS card cage. The diagnostic data is sent from the IOCs to the MDC on the left and right IOC-4 high-speed ARINC 429 buses.

The MDC sends the diagnostic data through the MDC-2 high-speed ARINC 429 bus to be shown on the MFD (Figure 45-1). You can see this diagnostic data and download diagnostic reports from the file server unit (FSU).

The cursor control panel (CCP) gives you the controls to get access to the Maintenance Main Menu for the MDC. The Maintenance Main Menu contains the following pages:

- Current Faults
- Aircraft History
- System Parameters
- ATA Index
- LRU Index
- LRU Test
- LRU Rigging
- MDC Setup
- Configuration Data
- Report Download

There is also a live data view capability that displays airframe PCB I/0 data to indicate current state, FADEC operating parameters, and switch/sensor values. The switch/sensor values include flap position as well as control surface trim position in support of auto rig functionality for FDR control surface position sensors.

The DBU-5000 is a data loader and interface for the LRUs on the aircraft and is located in the pedestal below the right CCP (Figure 45-2). The DBU is used to load the FMS, MDC, FSU databases and software through the Ethernet port on the FSU-5010. Removable storage media (e.g. USB memory device) is used to transfer files to the LRUs.

The MDC monitors the data that it receives from the left and right IOC-4110 IOC modules in the IAPS card

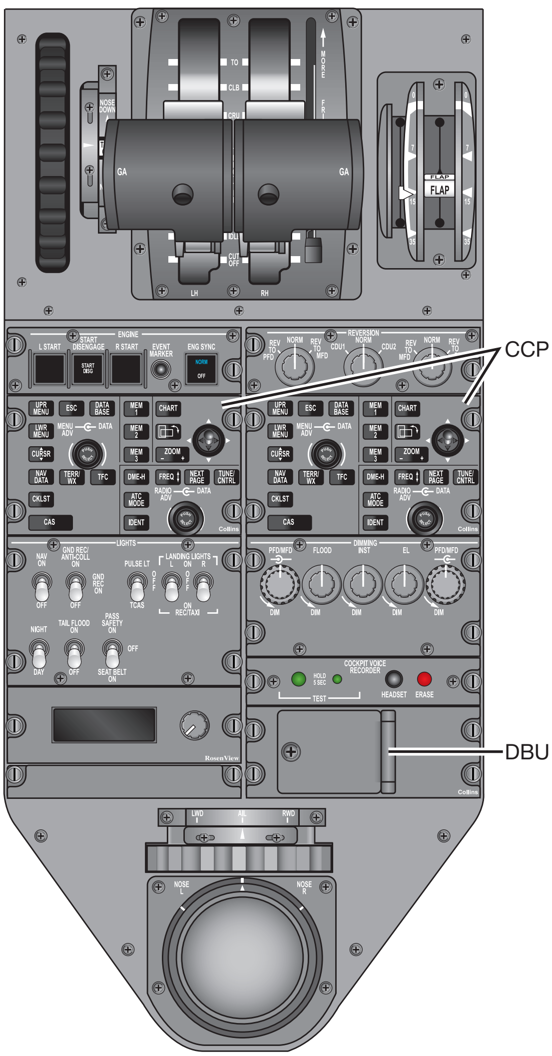


Figure 45-2. Center Pedestal

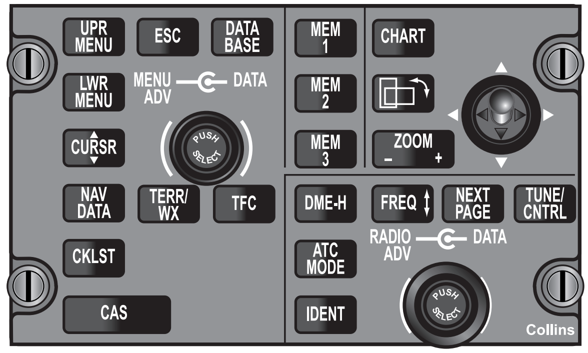


Figure 45-3. CCP Panel

OPERATION

To access the maintenance menu apply external power to the aircraft and turn on aircraft power and the avionics. Allow approximately 5 minutes for the system to stabilize before proceeding.

Push the DATABASE button on the CCP followed by the UPR MENU or LWR MENU button on the CCP to show the DATABASE MENU (Figure 45-3). Turn the MENU ADV (outer) knob on the CCP to highlight the MAINTENANCE MAIN MENU. Push the PUSH SELECT (inner) button on the CCP to get access to the MAINTENANCE MAIN MENU page.

The MAINTENANCE MAIN MENU contains the following pages :

CURRENT FAULTS—Gives advanced diagnostics of the component with the fault. The information contained in the current faults page includes:

- DETAILED DIAGNOSTIC DATA—Displays active data words and status conditions, as well as the diagnostic words that caused the fault message to be recorded. The DATA USED TO DETERMINE MESSAGE display shows all the diagnostic words that were used in the MDC logic equation.
- The LRU DIAGNOSTIC DATA fields show the octal label and diagnostic word received from the effected LRU and a maximum of ten diagnostic words can be shown. The field shows dashed lines if the associated word is not received. Each diagnostic word contains useful information that may simplify troubleshooting. The diagnostic words can be shown in either binary or hexadecimal format.

CURRENT SERVICE MESSAGES—Shows events or conditions which may not be faults but could be of interest to maintenance.

AIRCRAFT HISTORY—Gives the fault history and flight leg summary that includes fault history, service message history, engine trend history, and flight leg summary.

LRU TEST—Provides discrete testing capability for individual LRUs.

LRU RIGGING—Accesses specific rigging information for selected LRU.

SYSTEM PARAMETERS—Shows maintenance or information aircraft system parameters on a real-time basis.

ATA INDEX—Shows a list of the ATA chapters for the LRUs on the aircraft.

LRU INDEX/OPERATIONS—List of aircraft LRUs.

MDC SETUP—Provides setup control of some MDC parameters.

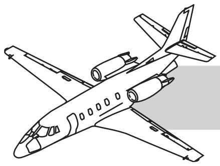
Examples include the aircraft identification code and the aircraft clock, configuration control, and the file load function that allows files from a database, including user checklists, to be loaded from a disk.

CONFIGURATION DATA—Shows the configuration strapping unit configuration.

REPORT DOWNLOAD—Provides the ability to download MDC report files via the DBU.

There is also a separate flight control system (FCS) DIAGNOSTICS page that allows troubleshooting of autopilot and yaw damper failures. Access to the FCS DIAGNOSTICS page is obtained from the DATABASE MENU.

NOTES



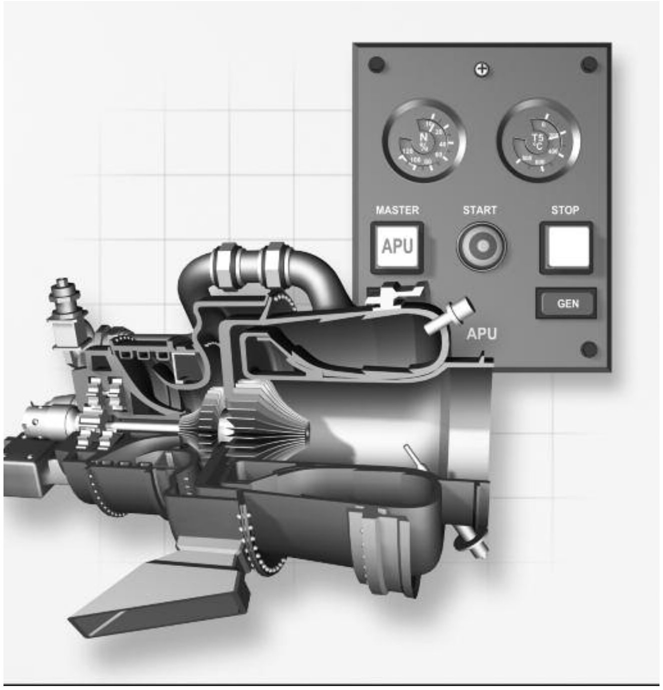
CHAPTER 49
AUXILIARY POWER UNIT

CONTENTS

ILLUSTRATIONS

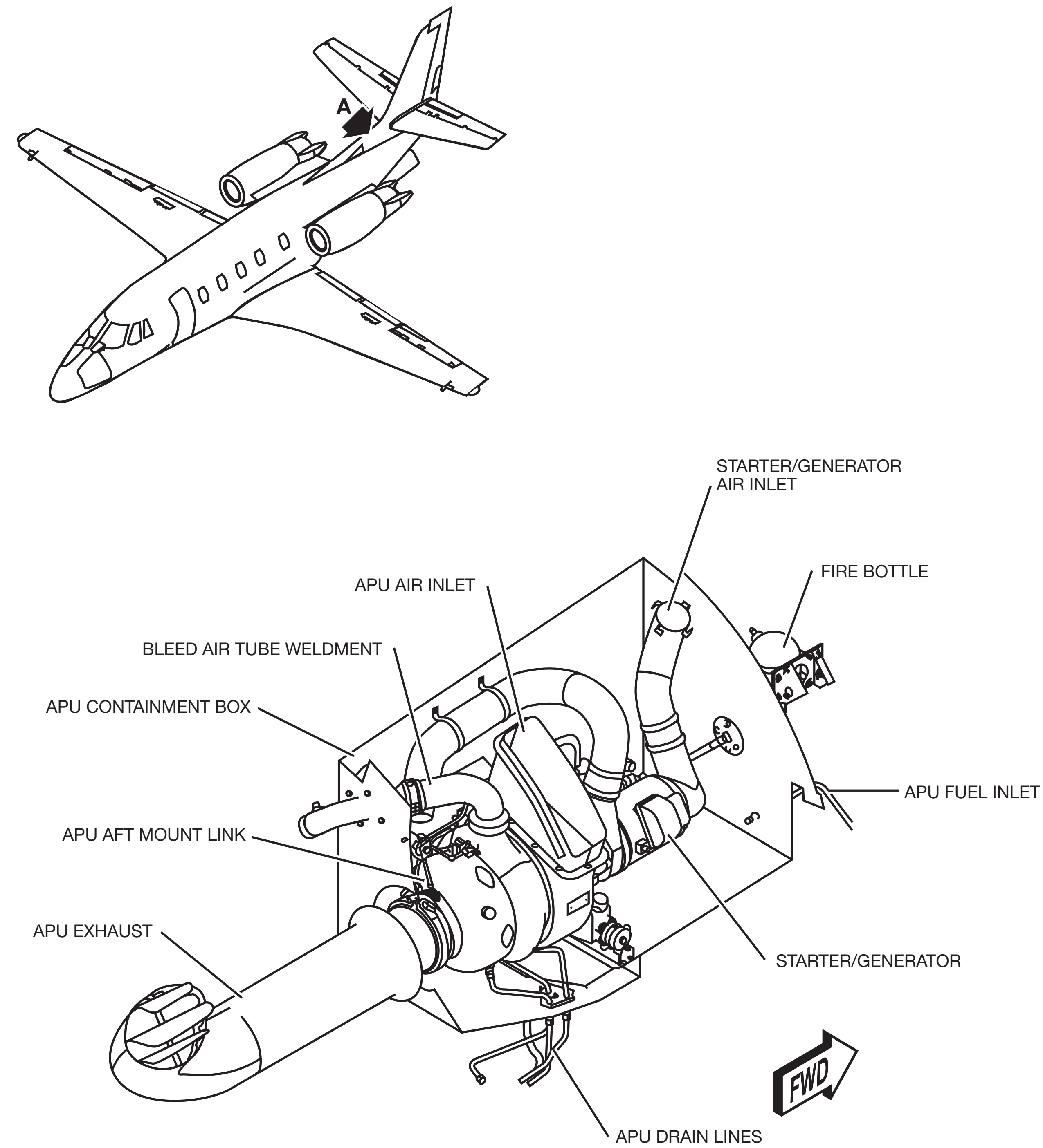
TABLES

CHAPTER 49
AUXILIARY POWER UNIT



INTRODUCTION

This chapter describes the onboard auxiliary power unit (APU) for the 560XL/XLS/XLS+, which generates to provide auxiliary bleed air and auxiliary electrical power. Information is provided for the major APU sections, major systems, fire detection and extinguishing. References for this chapter and further specific information can be found in Chapter 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," and Chapter 49—"Airborne Auxiliary Power," of the *Aircraft Maintenance Manual (AMM)*.



DETAIL A

Figure 49-1. RE100 (XL) APU

GENERAL

The APU is an AlliedSignal Aerospace Inc./Honeywell model RE100 (XL). It is in a containment box in the tail cone of the aircraft (Figure 49-1). The air inlet for the APU is on the upper right corner of the access door. Exhaust exits through a vent on the right side of the tail cone, forward and slightly below the horizontal stabilizer leading edge. It is a fully-automatic, constant-speed, gas turbine engine, that provides both electrical and pneumatic (bleed air) power while on the ground or in the air.

The APU is certified for ground and in-flight use. Maximum altitude is 20,000 feet for starting and 30,000 feet during operation. An amber APU ON crew alerting system (CAS) message appears if the APU is on above 30,000 feet (Figure 49-2).

APU ON			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard

This message indicates the APU is on above 30,000 feet.
APU operation is not approved above 30,000 feet.

Figure 49-2. APU ON CAS Message

Subsystems include:

- Lubrication
- Fuel
- Ignition
- Control/indicating
- Electrical
- Pneumatics

A fire detection and extinguishing system (attached to the APU) is monitored and controlled in the cockpit. Electrical power is supplied (using shaft power) by the auxiliary generator mounted to the accessory gearbox.

Pneumatic and shaft power may be supplied simultaneously or independently. When both types of power are demanded, the shaft power has priority. Selector switches in the flight crew compartment initiate all load requirements. APU is said to be at idle when running at 100% speed with no power being extracted. When loads are exerted on APU, fuel flow increases maintaining constant speed of 100%.

Protective devices within the APU engine control and monitor lubricating oil pressure and temperature, turbine exhaust gas temperature, powerplant overspeed and underspeed, and sequence failure.

NOTES

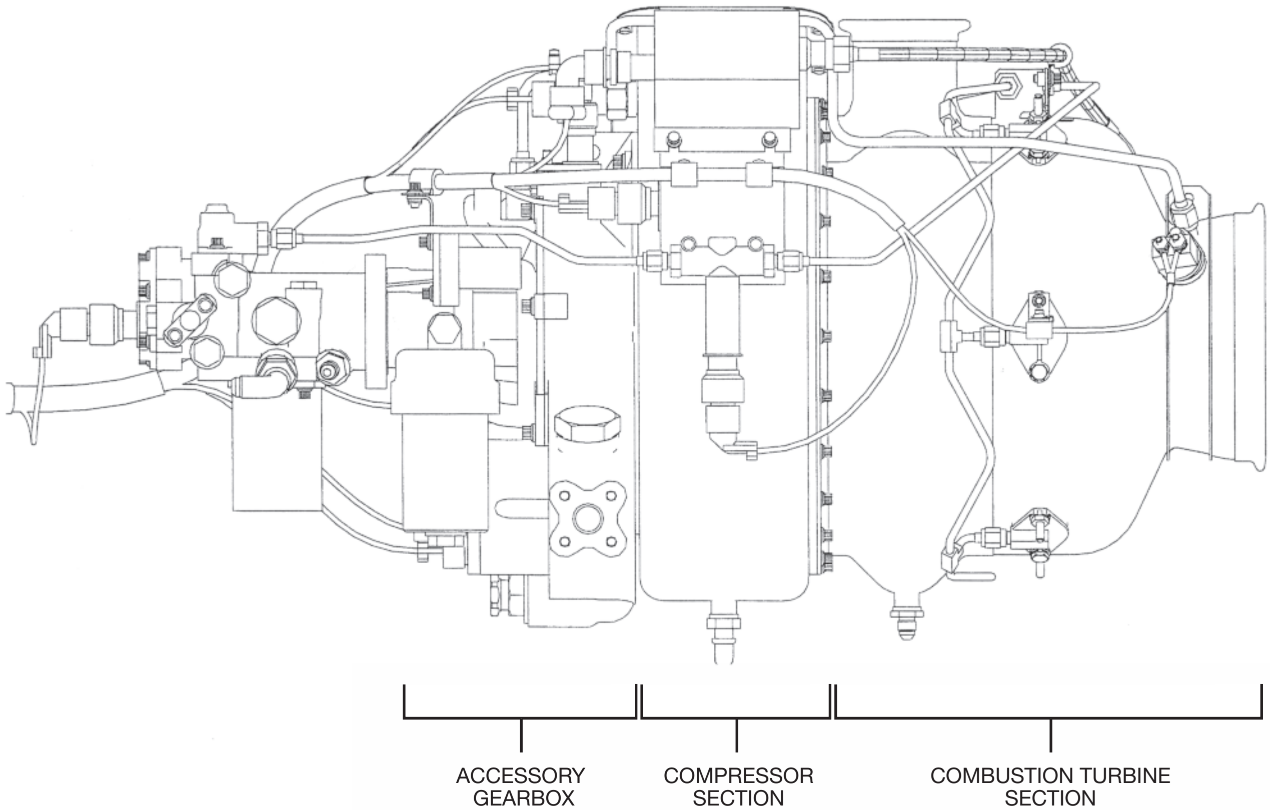


Figure 49-3. APU Major Sections

APU MAJOR SECTIONS

DESCRIPTION

The APU is divided into three main sections (Figure 49-3):

- Compressor
- Turbine
- Accessory gearbox

APU engine power is developed through compression of ambient air by the centrifugal compressor. The compressed air, when mixed with fuel and ignited, drives the single-stage radial inflow turbine. The rotating shaft power of the turbine rotor drives the compressor and the output drive shaft, which in turn powers the accessory gearbox.

COMPONENTS

Compressor Section

The compressor section includes:

- Air inlet duct
- Inlet housing
- Inlet screen
- Single-stage centrifugal compressor impeller
- Diffuser
- Deswirl deflector

Turbine Section

The turbine section consists of:

- Turbine housing
- Annular reverse flow combustor
- Turbine nozzle
- Single stage radial inflow turbine rotor

Accessory Gearbox

The accessory gearbox reduces high-speed low-torque shaft power from the power section, to low-speed high-torque power as required to drive the APU accessories. This gearbox section includes:

- The starter/generator
- Lubrication module
- Fuel control unit

NOTES

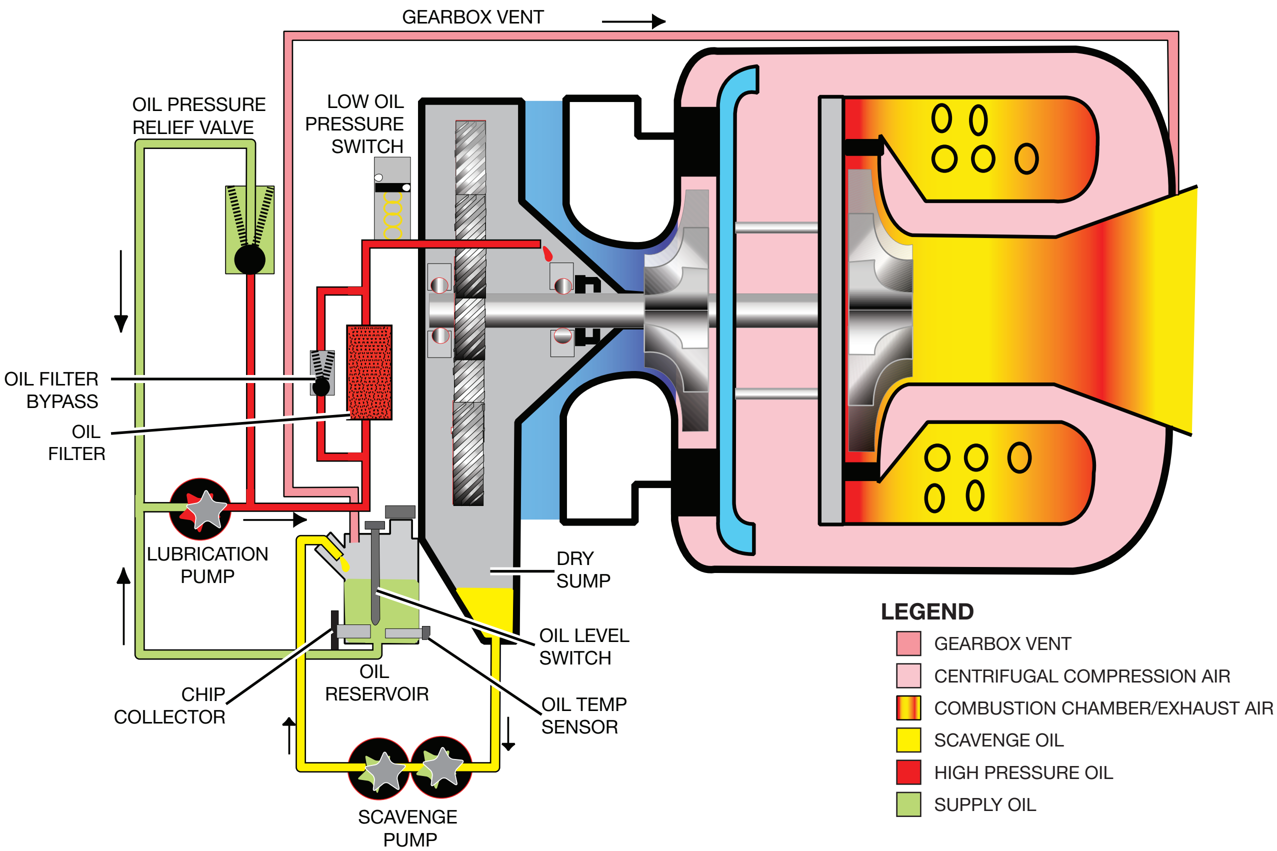


Figure 49-4. Lubrication System Schematic

APU SYSTEMS

The APU systems include:

- Lubrication system
- Fuel system
- Ignition system
- Control/indicating system
- Electrical system
- Pneumatic system

LUBRICATION SYSTEM

Description

Lubrication is necessary to clean, cool, and reduce friction between moving parts (Figure 49-4). The lower portion of the gearbox housing is cast to form a 2-quart oil reservoir. The critical points requiring lubrication are served by drilled passages and a transfer manifold. Remaining areas are lubricated by splash oil.

Oil flow starts with the lubrication pump drawing oil directly from the sump and directing it through a 10-micron filter, which has bypass provisions.

Oil travels through internal passageways and manifolds to:

- Main shaft bearings
- Planetary gear system bearings
- Fuel control unit drive shaft

Oil is also directed to the low oil pressure (LOP) switch, which monitors oil pressure.

The oil gravity feeds along the sides of the gear case housing, from the various lubricated areas and back to the sump area. It is here that the oil is cooled. The oil is then returned to the reservoir by the lubrication scavenge pump. An oil temperature bulb in the oil reservoir always monitors the oil temperature. Air is separated from the oil by an air/oil separator and is then routed to the tail pipe through the vent line.

NOTES

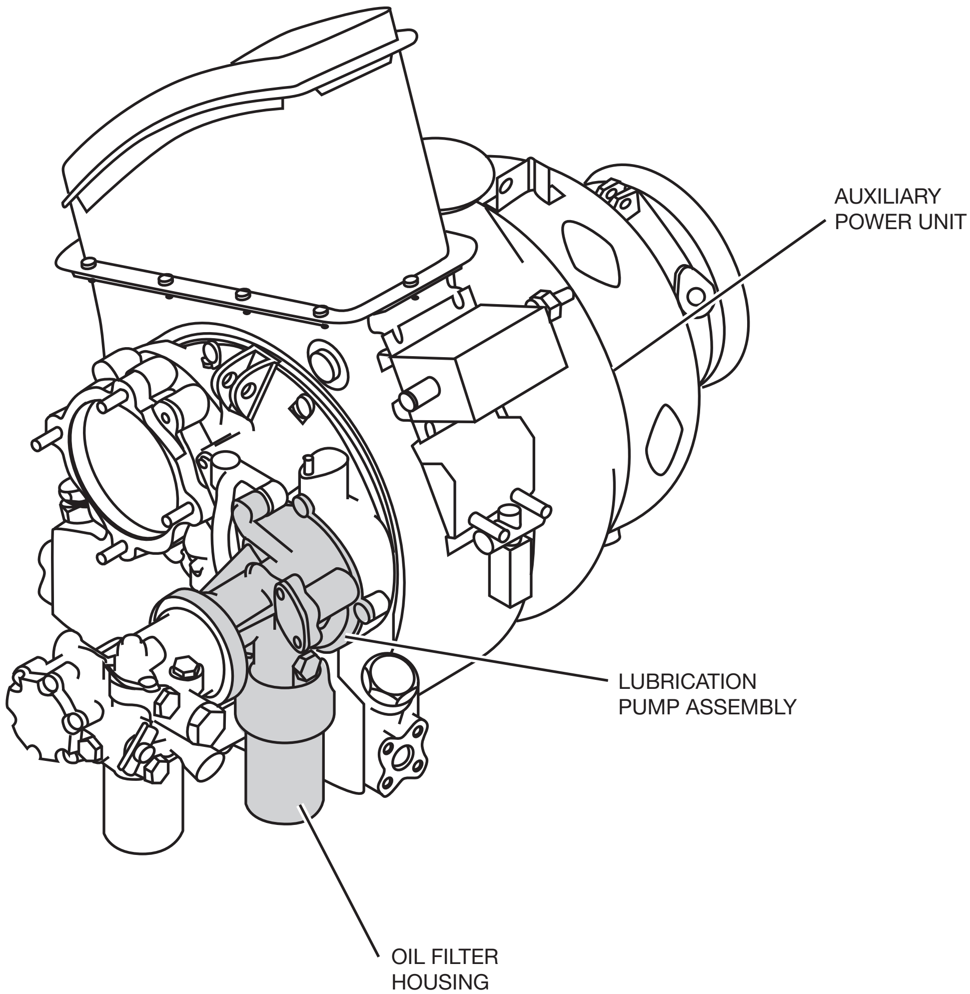


Figure 49-5. Lubrication Pump Assembly

Components

Lubrication Pump Assembly

The lubrication pump assembly is on the gearbox housing (Figure 49-5). The pump assembly contains both a pressure and a scavenge pump. There is no pressure regulator in the system. The pressure pump provides a supply of oil with enough volume to obtain 60 to 80 psig. An ultimate relief valve set at 200 psig prevents system overpressurization.

The pressure pump is a single-gear rotor-type pump. The scavenge pump is a dual-gear rotor-type pump. Both are part of the lubrication pump assembly, which is a line replaceable unit (LRU).

Oil Filter Assembly

The oil filter assembly is on the lubrication pump assembly. It includes: the filter housing, filter element, and a bypass indicator. The filter housing threads into the pump assembly and seals with an O-ring. The filter is a disposable pleated cellulose element. The bypass indicator, on the bottom of the filter housing, indicates an impending bypass condition of the filter element. At 45 psi differential across the filter element, the indicator extends out the bottom of the housing. At 60 psi differential, oil begins bypassing the filter element.

NOTES

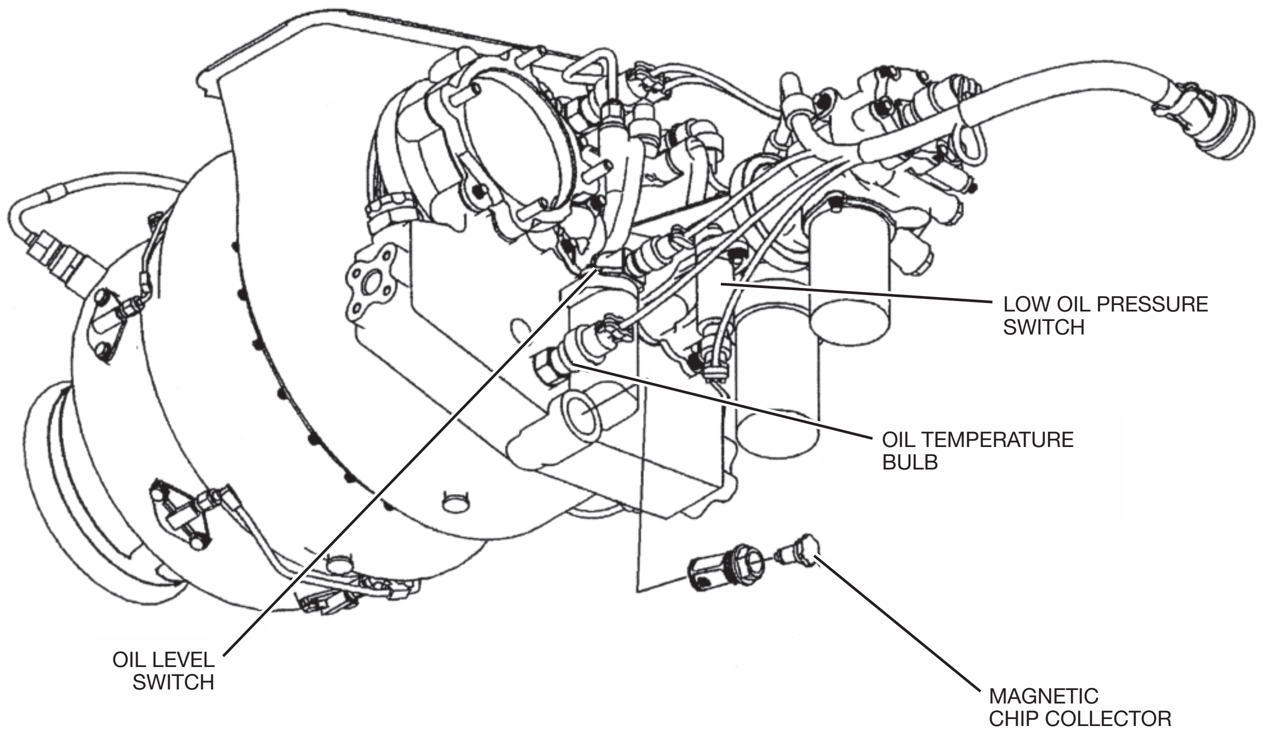


Figure 49-6. Oil System Components

Low Oil Pressure Switch

The low oil pressure (LOP) switch is on the bottom of the lubrication pump assembly (Figure 49-6). The switch position is checked before start by the electronic control unit (ECU). The switch position must be closed. If it is found open, the APU will not start and a fault is logged into memory. During start, oil pressure begins to increase, and the LOP switch opens by approximately 40 psig. Once the APU reaches 95% rpm, if the oil pressure drops below 26 psig for 10 seconds, the ECU shuts down the APU.

Oil Temperature Bulb

The oil temperature bulb is below the starter generator on the lower portion of the gearbox. It is a resistance-type sensor that allows the ECU to monitor oil temperature. During prestart, the ECU checks for a resistance value that would indicate a temperature between –73°C (–100°F) and 260°C (500°F). If the resistance is out of this range, the ECU determines that the oil temperature bulb is faulty and does not allow the APU to start. Once the APU reaches 95% speed and above, the ECU initiates a protective shutdown if the oil temperature exceeds 149°C (300°F) for 10 seconds.

Oil Level Switch

The oil level switch is on the forward side of the gearbox just below the starter/generator. The oil level switch monitors the oil level inside the gearbox and indicates whether the level is low, or if it needs to be serviced on the maintenance panel.

Magnetic Chip Collector

The chip collector is threaded into the oil drain plug, which is at the lowest point in the reservoir on the forward side. The collector provides for a visual inspection for ferrous metal in the oil. A self-closing check valve behind the chip collector allows the collector to be removed and inspected periodically for ferrous metal without draining the reservoir.

Diagnostics

To Replace Oil Filter Element:

1. Gain access to the APU by removing the APU access door.
2. Carefully twist the oil filter body until it is free of the housing.
3. Remove filter body with element.
4. Separate the filter element from the oil filter body. Dispose of the filter element.
5. With a new O-ring, install the filter element and the oil filter body in the receptacle.
6. Secure the oil filter body by hand turning until body is tight.

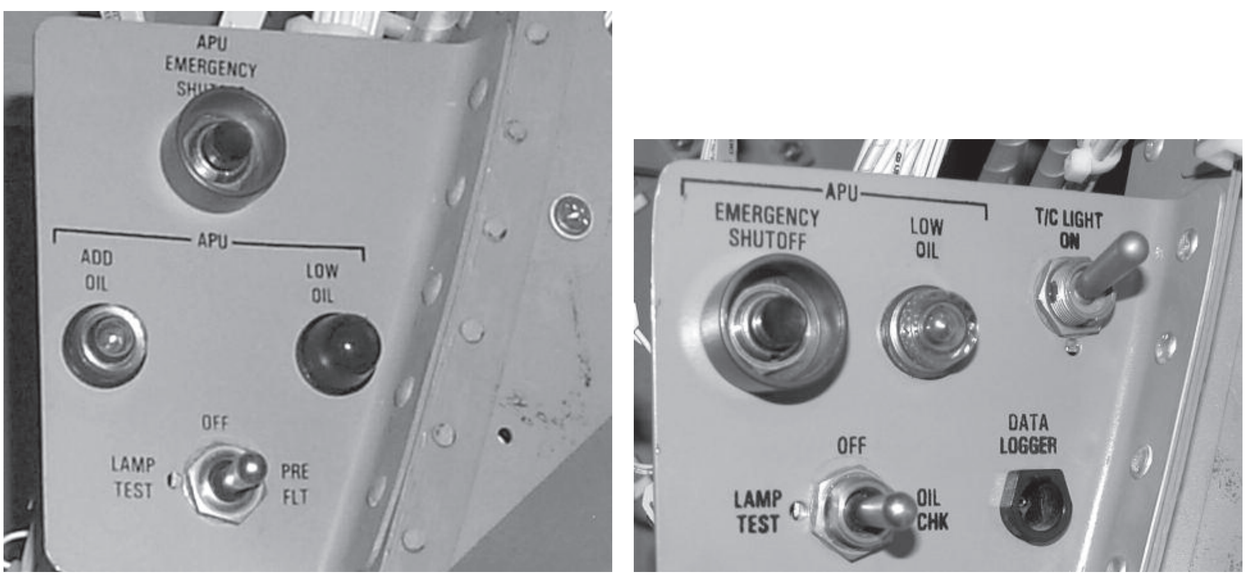
CAUTION

Do not use a wrench to tighten oil filter body. Over-tightening can cause damage to the packing and filter housing.

7. Clean any oil spills from inside containment box.
8. Install the APU access door.

OIL FILLER CAP

OIL LEVEL SWITCH



XL/XLS **XLS+**

APU MANTENANCE PANELS

Figure 49-7. APU Oil Servicing

Oil Servicing

Servicing the onboard APU oil system consists of periodic oil changes and normal between-oil-change servicing (adding oil). The engine operates on oils conforming to MIL-L-23699 specifications, the same oil utilized in the main engines.

There is an APU maintenance panel on the forward doorframe of the tail cone access door (Figure 49-7). There are two lights on the panel: an amber ADD OIL light, and a red LOW OIL light. The lights indicate the level of the oil while the switch on the maintenance panel is held to PRE FLT. The ADD OIL light indicates that the oil level has dropped to approximately 300cc below full. The LOW OIL light indicates that the oil level has dropped to approximately 550cc below full. If the LOW OIL light illuminates, oil must be added before APU operation.

NOTE

Oil level must be checked before the first start each day the APU is operated.

To Add Oil:

1. Gain access to the oil filler plug on the top right portion of the APU gearbox by opening the APU access door.
2. Cut safety wire, press down and rotate cap a quarter-turn counterclockwise.

CAUTION

Never allow the oil level above the full mark when engine is cold. An overfull oil sump results in oil foaming, low oil pressure, and abnormal gear wear.

3. Using the proper type of oil, add oil to the sump until the correct level is reached.
4. Install cap by pressing down into filler neck and rotating a quarter-turn clockwise. Safety wire the cap.
5. Clean up any oil spills. Close the APU access door.

To Drain Oil:

1. Gain access to the APU oil sump by removing the panel from APU containment box floor.
2. Place a container with approximately 4-quart capacity under the containment box.

NOTE

The chip collector is threaded into the oil drain plug. Remove the chip collector and oil drain plug as one assembly.

3. Remove the O-ring from the drain plug. Separate the magnetic plug from the drain plug. Remove the O-ring from the magnetic plug.

CAUTION

If metallic particles are found on the magnetic plug, refer to the *APU Component Maintenance Manual.*

4. Check for metallic chips on the chip detector of the magnetic plug.
5. With a new O-ring, install magnetic plug in the oil drain plug.
6. With a new O-ring, install the drain plug (and chip collector) in the oil sump. Tighten the drain plug and safety wire.
7. Fill the oil sump with three quarts of lubricating oil.
8. Check the O-ring on the cap. Replace O-ring if damaged. Install cap into filler neck by pressing down and turning a quarter-turn clockwise.
9. Operate the APU for approximately two minutes, then shut down the unit.

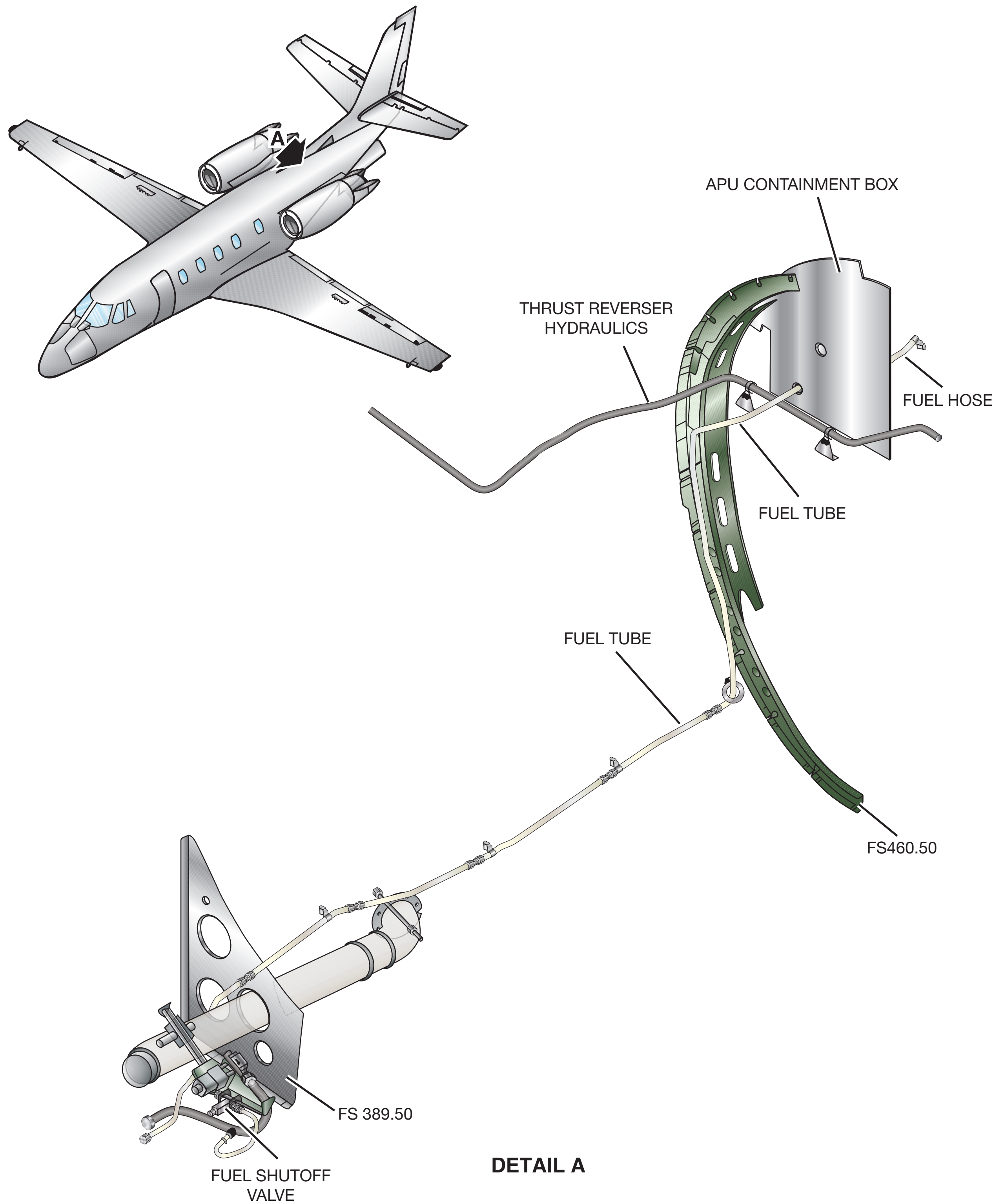


Figure 49-8. Aircraft Fuel System

AIRCRAFT FUEL SYSTEM

Description

Fuel supplied to the APU originates at the right wing fuel hopper (Figure 49-8). Fuel pressure is supplied to APU fuel control unit (FCU) by the right electric fuel boost pump. This is automatically activated, if the right fuel boost pump switch is in NORM when the APU start sequence is initiated; and is discontinued when the APU stop sequence is initiated. The fuel boost pump operates the entire time the APU is in operation.

NOTE

XL/XLS—When the right boost pump is commanded to operate due to only APU operation, the FUEL BOOST R annunciator does not illuminate. However, if the APU is running and the pump is commanded ON for any other reason (ie: crossfeed, low fuel pressure, main engine start, or right boost pump switch on) the FUEL BOOST R annunciator illuminates.

XLS+—The FUEL PUMP BOOST ON R white CAS message is displayed when commanded to operate due to APU operation only. The FUEL BOOST PUMP ON R amber CAS message displays when the fuel boost pump is on, fuel pressure is low, and the throttle is not in cutoff (Figure 4-9).

FUEL BOOST PUMP ON R			
Color	**Inhibited By**		**Debounce**
White	SIPI		

The white message is displayed when the fuel boost pump is selected on, APU running, or not turned on by low fuel pressure.

Figure 49-9. FUEL PUMP BOOST R CAS Message

NOTE

The boost pump does not operate, (to supply the APU with fuel) if the boost pump switch is in OFF (XL/XLS).

There is a solenoid-type fuel shutoff valve behind the aft wing spar below the fuselage on the right side. The shutoff valve is energized open as soon as START is initiated. It remains open while the APU is in operation. It closes when APU operation is terminated, through either a commanded shutdown or a protective shutdown.

Operation

If crossfeed from the left wing tank is desired while the APU is operating, the right fuel boost pump automatically shuts off to allow fuel transfer. When the fuel crossfeed is turned OFF, the right fuel boost pump automatically reactivates once the system cycles OFF.

When crossfeed is selected from the right wing tank while the APU is operating, the right boost pump continues to operate, but the FUEL BOOST R annunciator illuminates. When crossfeed is terminated, the FUEL BOOST R light extinguishes, but the boost pump continues to operate.

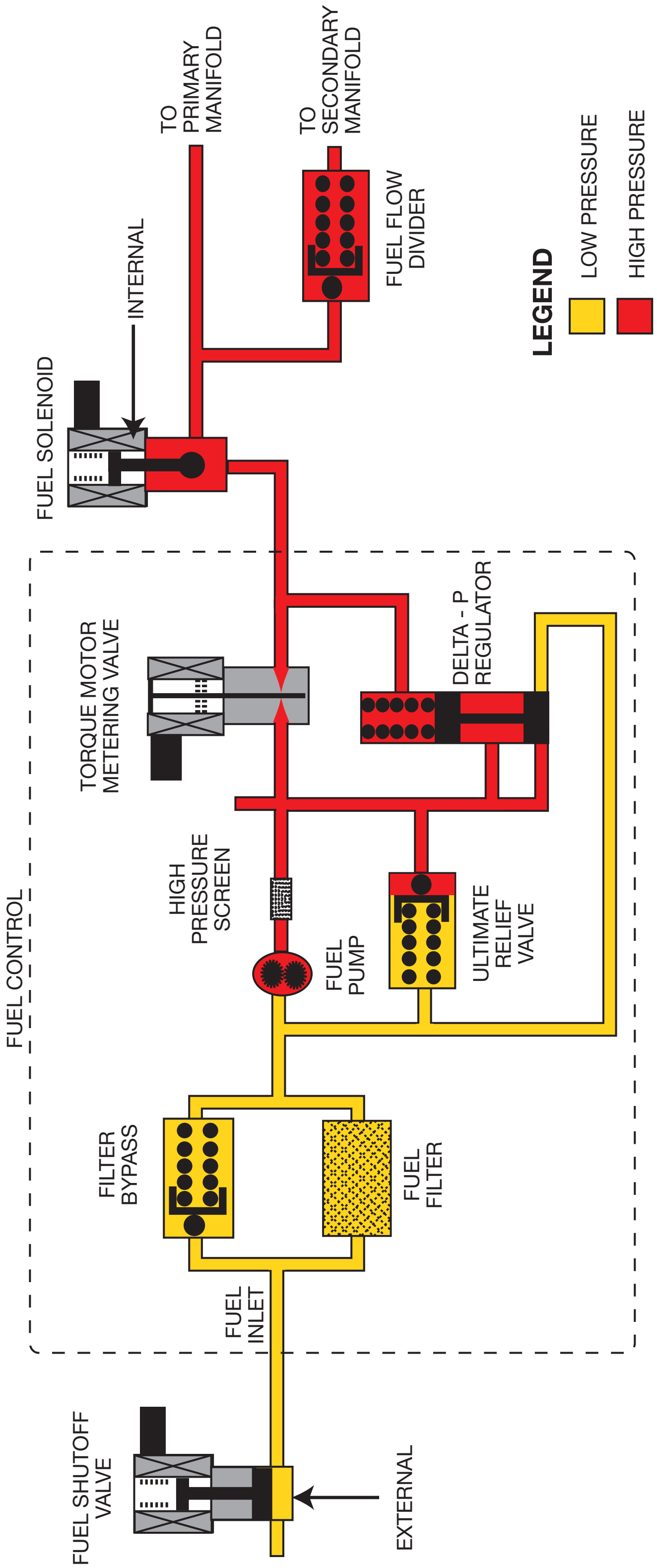


Figure 49-10. Fuel System Schematic

APU FUEL SYSTEM

Description

The APU fuel system is a fully automatic electronically controlled system (Figure 49-10). During start-up the fuel system provides the correct amount of fuel to support combustion to governed speed. Once governed speed is reached, fuel flow is controlled as needed to meet the demands of varying pneumatic and electrical loads while maintaining a constant speed.

Fuel for the APU powerplant is drawn from the right fuel sump. A solenoid-actuated fuel shutoff valve, controlled by either the APU master switch or APU fire signal, isolates the APU fuel system from the airplane fuel system. This APU fuel supply is generated and sustained by the airplane right fuel boost pump. If crossfeed is selected, then the right boost pump shuts down and the left boost pump supplies fuel for APU operation.

At the APU, fuel enters the FCU section, which contains a disposable fuel filter element. Fuel flows through the filter element then on to a high-pressure pump. Fuel discharged from the high-pressure pump, passes through a 70-micron, non-accessible screen before going to the torque motor (metering valve) and the Delta-P (DP) regulator. An ultimate relief valve, set at 800 psig, is incorporated, allowing fuel to flow back to the pump inlet during APU shutdown. This prevents to the components downstream of the pump.

When more fuel is demanded by the APU, the metering valve is commanded toward a more open position by the ECU. As the metering valve moves to a more open position, increased pressure across the metering valve forces the DP bypass valve to a more “closed” position. This DP valve action increases fuel flow to the fuel nozzles and maintains APU rpm at governed speed.

Beyond the metering valve, fuel flows out of the FCU in a line which leads to a fuel shut-off valve, between the FCU and fuel nozzles. The ECU controls this shut-off valve, and energizes it open at APU speeds above 5%, allowing the fuel to pass on to the flow divider. The flow divider then divides the fuel between the primary and secondary manifolds.

NOTES

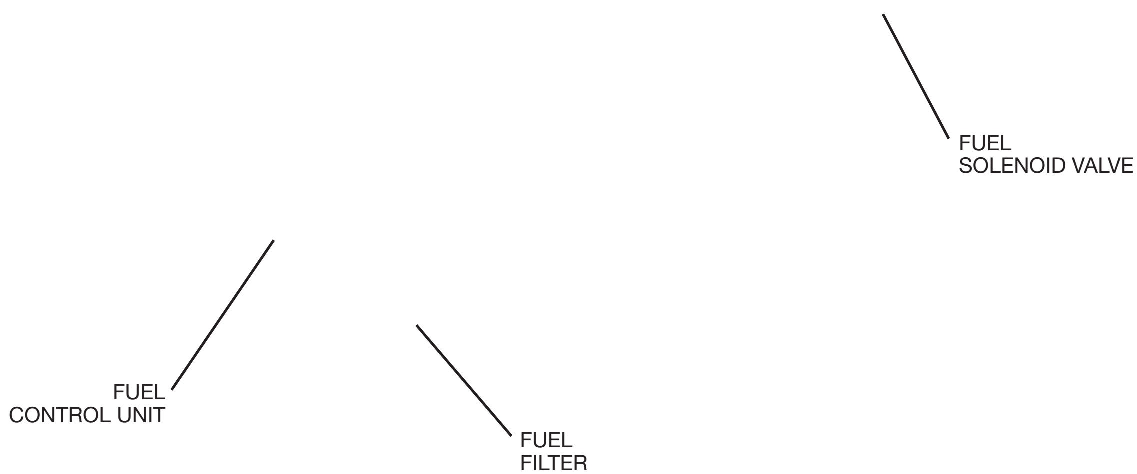


Figure 49-11. APU Fuel System

Components

Fuel Control Unit

The FCU is in a V-band clamp on the front side of the lubrication pump assembly (Figure 49-11). It receives electrical signals from the ECU to schedule the correct amount of fuel for the desired engine operation.

NOTE

When removing the FCU, it is normal to see oil dripping from the fuel control/lubrication pump split-line area. Do not forget to install the O-ring when reassembling.

Fuel Filter

The fuel filter is on the bottom of the FCU. It is a 40-micron disposable-type element that is periodically replaced. The FCU allows fuel to bypass the filter element at a 7.5 psid across the filter without any indication.

Fuel Solenoid Valve

There is a fuel solenoid valve downstream of the FCU. It is a normally “closed” valve that is energized “open” during APU operation. During the start sequence, the fuel solenoid valve receives 28 volts from the ECU when the APU reaches 5% speed. The valve continues to receive 28 volts from the ECU until an APU shutdown is initiated.

NOTE

There is one pair of carbon discs and lapped seats in the fuel solenoid valve. Extreme care must be taken when installing the fittings into the fuel solenoid valve to prevent damage to the carbon discs.

NOTES

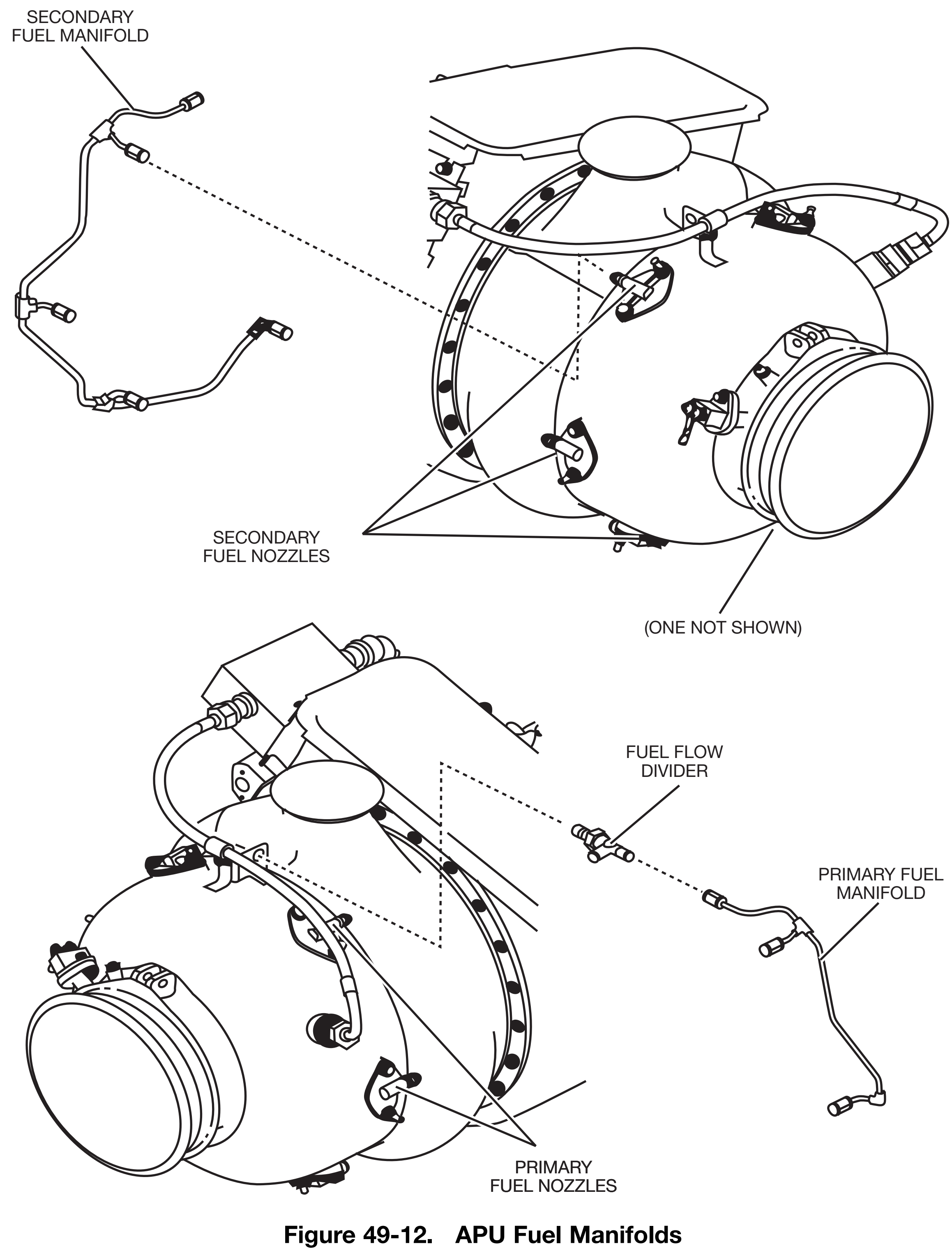


Figure 49-12. APU Fuel Manifolds

Fuel Flow Divider

The flow divider is in the aft turbine plenum section. Its purpose is to sequence and distribute fuel between the primary and secondary manifolds (Figure 49-12). Initially during the start sequence, fuel is sent out to the primary manifold only. As fuel pressure increases to 75 psi, the flow divider valve opens and allows fuel to flow to the secondary manifold also.

Fuel Nozzles

The APU contains a total of six fuel nozzles, around the turbine plenum and extending into the combustion chamber. Two are primary nozzles, on either side of the igniter plug. The other four are secondary nozzles, which are then evenly distributed around the remainder of the turbine plenum.

The fuel nozzles provide proper fuel atomization for initial combustion and maximum energy extraction. An air shroud directs compressor air across the nozzle surface surrounding the tip of the nozzle. Cooler air from the compressor ensures proper nozzle tip temperature. The primary and secondary nozzles are not interchangeable and have different part numbers.

NOTES

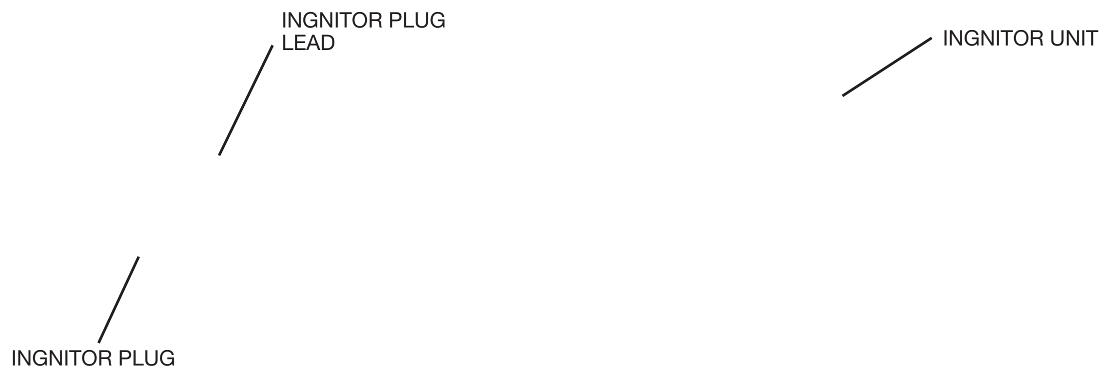


Figure 49-13. Ignition System

IGNITION SYSTEM

Description

The ignition system includes:

- Ignition unit
- Igniter plug lead
- Igniter plug

The ignition unit is in a bracket on the left side of the APU inlet housing (Figure 49-13). The igniter plug is on the upper, right side of the turbine plenum where it protrudes into the combustion chamber. The igniter plug lead connects the ignition unit to the igniter plug.

The ignition system is a fully-automatic system, controlled by the ECU. At 5% APU rpm, the ignition unit is supplied 28 volts, to begin firing the igniter plug at rate of 3 sparks per second. Four seconds after reaching 95% APU rpm, the ignition unit is deenergized, terminating ignition. If APU rpm drops below 94% after the ignition has terminated, the ECU turns the ignition system back on. This is the “auto relight” function of the ECU.

WARNING

Output voltage of the ignition unit is dangerous and could be lethal. Make sure it is deenergized and grounded before input and output leads are disconnected. Do not touch center contact of output terminal.

NOTES

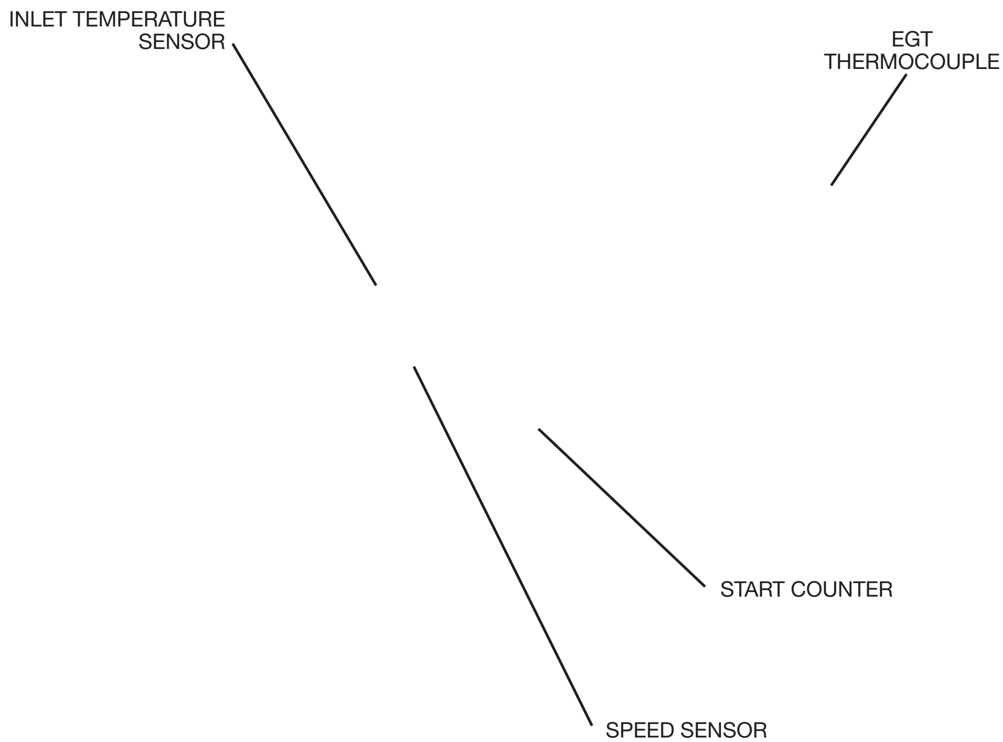


Figure 49-14. Control/Indicating Components

CONTROL/INDICATING SYSTEM

Description

The engine uses control components that automatically control the engine and hold required engine speed and safe operating temperatures, from start initiation to full load. Some of the control components require connection with remote switches and relays for proper operation. Two PC boards: APU monitor and APU control, are inside a logic module box next to the copilot seat in the right side console. These provide an interface between the APU and the aircraft.

Speed Sensor

The speed sensor is on top of the gearbox above the lubrication pump (Figure 49-14). It provides the ECU with the APU speed in order for the ECU to control the operation of the APU.

EGT Thermocouple

The exhaust gas temperature (EGT) thermocouple (T5 or immersion thermocouple) is at approximately the 9 o'clock position on the aft section of the turbine plenum. It extends into the exhaust gas stream. The information from the EGT thermocouple is supplied to the ECU for:

- Fuel scheduling
- Indicating in the cockpit
- Control the bleed-load control valve (BLCV)
- To trigger an overtemperature protective shutdown

Inlet Temperature Sensor

The inlet temperature sensor (T2) is a resistive thermal device threaded into the APU inlet air plenum (Figure 49-12). It monitors the APU inlet temperature and supplies its information to the ECU for fuel scheduling. When the APU is operating with the bleed- load control valve open, the ECU checks the T2 sensor to verify that it is indicating a temperature range between –73°C (–100°F) and 93°C (200°F). If it is not within the range, the ECU will initiate a protective shutdown.

Start Counter

The start counter on the left side of the APU between the ignition unit and fuel solenoid valve, retains the total APU start cycles. During a start cycle, when the APU reaches 95% speed plus 4 seconds, the start counter adds one start to its internal memory.

NOTES

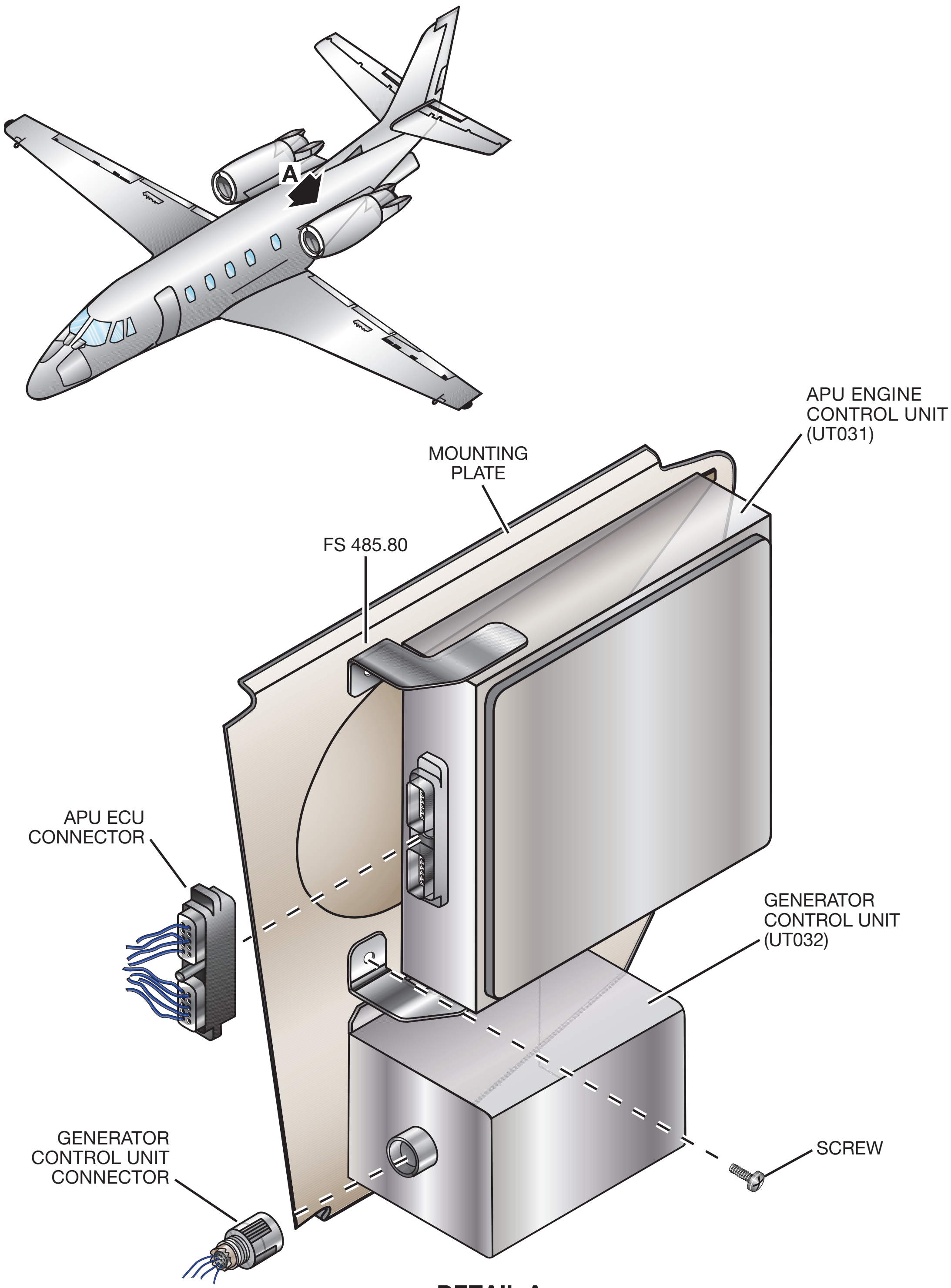


Figure 49-15. Electronic Control Unit

Electronic Control Unit

The ECU is a fully-automatic digital unit in the tail cone, on the inboard side of the containment box (Figure 49-15). The ECU is the interface for receiving inputs and giving the required outputs to safely operate the APU and to shut it down when necessary.

The following are controlled by the ECU:

- Start sequence
- Acceleration timing
- On-speed governing
- Normal shutdown
- Fault detection and the logging of faults
- Connection to other aircraft subsystems

The following are conditions that cause the ECU to automatically shutdown the APU:

- High oil temperature (HOT)
- Loss of HOT protection
- Low oil pressure (LOP)
- EGT overtemperature
- Loss of EGT signal
- Loss of inlet temperature (T2)
- Overspeed *
- Open speed monitoring circuit
- Slow start
- Reverse flow
- No flame 20 seconds after start
- Loss of DC power
- ECU failure
- Fire loop failure
- Fire bottle low
- Fire indication*
- Pressing APU FIRE or APU EMERGENCY SHUTOFF button*
- Acceleration not reached

* Requires APU MASTER to be cycled before APU will restart.

The following conditions will cause a start inhibit of the APU:

- ECU internal failure
- Emergency discrete signal (APU panel or maintenance panel)
- Fire discrete input from the copilot control panel
- Oil temperature sensor out of range
- T2 sensor out of range
- EGT thermocouple out of range

ECU Inputs

The ECU receives several inputs from both the APU and the aircraft.

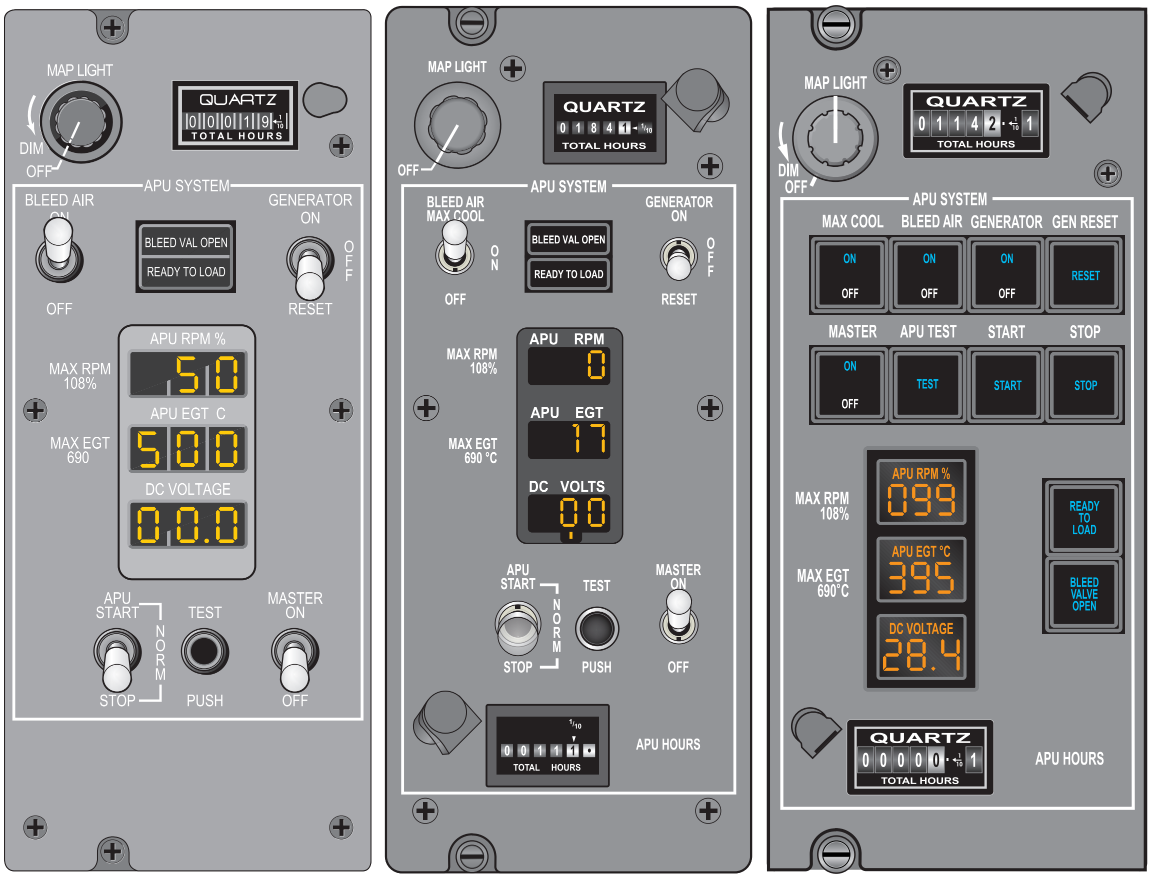
The following are ECU inputs:

APU to ECU Inputs

- Oil pressure from the LOP switch
- Oil temperature from the oil temperature bulb
- % rpm speed from the speed sensor
- Inlet air temperature from the T2 temperature sensor
- Exhaust gas temperature from the T5 temperature sensor (or immersion thermocouple)
- BLCV or LCV position

Aircraft to ECU Inputs

- Master On/Off
- Bleed On/Off
- Start/Stop switch
- Squat switch GND/AIR
- APU fire loop
- Emergency shutdown switch
- Generator relay open/closed
- Start relay open/closed



XL XLS XLS+

Figure 49-16. APU Control Panel

ECU Outputs

With the APU and aircraft inputs, the ECU gives several outputs depending upon which inputs are received. The following are outputs given by the ECU:

- EGT indication
- % RPM speed indication
- READY TO LOAD light illumination
- BLEED VAL OPEN light illumination
- APU FAIL light illumination
- Input to FCU torque motor
- Input to fuel shut-off solenoid
- Input to ignition unit
- Input to start counter
- Input to BLCV

APU Control Panel

Operator control of the APU is achieved from the APU control panel forward of the copilot CB panel (Figure 49-16).

The APU control panel incorporates digital RPM, exhaust gas temperature (EGT) and DC VOLTAGE indicators. A READY TO LOAD and a BLEED VAL OPEN annunciator light indicates when the APU is ready to be loaded and when the bleed-air valve is open (respectively).

Switches (XL/XLS)

BLEED AIR ON/OFF (XL)—A two-position switch that sends a GND input into the ECU when in the ON position (as long as the ACM has not overheated.) In the OFF position, the input to the ECU is open.

BLEED MAX COOL ON/OFF (XLS)—A three-position switch that sends a GND input into the ECU when in the ON position, commanding the BLCV to the 50% open position (as long as the ACM has not overheated.) In the OFF position , the input to the ECU is open. In the MAX COOL position, a GND is sent to the ECU commanding the BLCV to the 100% open position

GENERATOR ON/OFF/RESET—A three-position switch in which the RESET position is a momentary contact. In the ON position, power out from the GCU on pin D is then supplied into the APU Monitor PC board for a generator ON input.

APU START/NORM/STOP—A three-position switch, normally in the center position. It has momentary contacts in the START and STOP positions. When either START or STOP are selected, a ground input to the APU CONTROL PCB is momentarily removed from either pin 27 or pin 28 (respectively).

MASTER ON/OFF—A two-position switch that supplies power to both APU PC boards on pin 2 and to bus bar XF002 while it is in the ON position.

TEST (PUSH)—A momentary push-button that initiates a test by providing a 28 VDC input to the APU monitor PC board and also into the ECU. Push and hold the test button, and the following lights illuminate:

- APU RELAY ENGAGE
- APU FAIL
- FIRE
- BLD VLV OPEN
- READY TO LOAD

After 10 seconds the APU display in the right CB subpanel displays the following information:

- EGT.. 500 ± 10
- RPM .... 50 ± 5
- VDC 0

Table 49-1. XLS+ APU CAS Messages

APU FIRE

Color	Inhibited By		Debounce
Red	LOPI	TOPI	Standard

This message is displayed when a fire is detected in the APU by a fire loop. 28 Volts on the input to EICAS means a fire has been detected, which causes the message to be displayed. Open circuit means a fire has not been detected, which causes the message to be removed. A voice aural is also triggered with this message.

APU FAIL

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message is displayed when the APU has failed. An APU failure indicates either the APU ECU has reported a failure or the APU fire bottle is low. The APU fail light on the RH panel will also come on simultaneously with the CAS message. A 28 Volt input to EICAS means the APU has failed, which causes the message to be displayed. Open circuit means the APU has not failed, which causes the message to be removed.

APU GENERATOR OFF

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

This message is displayed when the APU is on and the APU generator relay is not closed. The message is amber if the APU generator switch is selected on, and it is white if the APU generator is not selected on. 28 Volts on the input means that the APU is on, the APU generator relay is closed, and the APU generator switch is selected on, respectively. Open means the APU is not on, the APU generator relay is open, and the APU generator switch is selected off, respectively. If the APU generator is reset while the APU generator is on-line, the APU generator reset switch will turn off the generator relay, and the white message shall appear.

APU ON

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard

This message indicates the APU is on above 30,000 feet. APU operation is not approved above 30,000 feet.

APU FIRE BOTTLE LOW

Color	Inhibited By		Debounce
White	LOPI	TOPI	Standard

This message is displayed when the APU fire bottle is low, as measured by a pressure switch on the bottle. When the bottle is low, it sends a ground signal to the EICAS system, which posts the message. When the bottle is filled, it sends an open signal which removes the message. The APU FAIL message will be display with this message.

APU GENERATOR OFF

Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
White			

This message is displayed when the APU is on and the APU generator relay is not closed. The message is white if the APU generator is not selected on. Refer to amber EICAS message for details.

Switches (XLS+)

MAX COOL—Allows a greater amount of bleed air produced by the APU into the aircraft duct system. MAX COOL should not be on for engine starts. The switch illuminates in cyan when ON and white when OFF.

BLEED AIR—Allows bleed air produced by the APU to be diverted into the aircraft duct system. The switch illuminates in cyan when ON and white when OFF.

GENERATOR—Allows the APU generator to be connected to the crossfeed bus after the APU is running. The switch illuminates in cyan when ON and white when OFF.

GEN RESET—If the APU generator trips offline, the APU GEN RESET switch provides the capability to reset the generator field relay if the problem that caused the trip is no longer a factor. The switch illuminates in cyan when pushed and extinguishes when released.

MASTER—Powers up the ECU and provides power for the control panel gauges, switches, and annunciators. The switch illuminates in cyan when ON and white when OFF.

APU TEST—Performs a lamp test of all the annunciators on the APU SYSTEM panel and the copilot instrument panel. The TEST function also tests the digital indicators and the fire-detection system. The switch illuminates in cyan when pushed and extinguishes when released.

Annunciators (XL/XLS)

READY TO LOAD—An amber annunciator that illuminates four seconds after the APU reaches 95% speed to indicate that electrical and bleed air are now available. Illumination of the annunciator is a function of the ECU. The ground for the annunciator comes from the APU MONITOR PC board so that the DAY/NIGHT switch is able to dim the annunciator.

BLEED VAL OPEN—An amber annunciator that illuminates to indicate that the BLCV or LCV has opened. The illumination is a function of the ECU. When the ECU receives a signal back from the BLCV that it has opened, then the ECU sends 28 VDC to the annunciator and to the APU CONTROL PC board. The ground for the annunciator comes from the APU MONITOR PC board so that the DAY/NIGHT switch can dim the annunciator.

CAS Messages (XLS+)

Table 49-1 lists the APU CAS messages for the XLS+ aircraft.

APU Display

The APU display is made up of three LED indications: APU RPM, APU EGT, and DC VOLTS (see Figure 49-14). These displays are dimmed using the right panel dimming rheostat.

APU RPM—An indication of the APU engine speed as sensed by the speed sensor, the display input is a direct output of the ECU. MAX APU RPM: 108%.

APU EGT—An indication of the APU exhaust gas temperature as referenced by the EGT (T5) thermocouple, the display input is a direct output of the ECU. MAX APU EGT: 690°C.

DC VOLTS—An indication of the APU starter/generator output voltage, the display input comes from a 1-amp HT031 circuit breaker in the main power J-box .

Figure 49-17. Right Instrument Subpanel

Copilot Subpanel

The APU generator ammeter, APU FIRE switchlight, APU RELAY ENGAGED and APU FAIL indicators are on the copilot instrument panel (Figure 49-17).

AMMETER—The ammeter is an analog instrument with markings from 0–400 amps in increments of 50 (Figure 49-18). A red triangle is placed at 200 amps to indicate the max on-ground continuous rating; and a red line is placed at 230 amps for max continuous in flight. The indication comes from a shunt in the main power J-box that is sensing the current flow from the APU GEN BUS to the X-FEED BUS.

APU FIRE switchlight—The switchlight is combined with a switch and has a cover guard that must be raised in order to activate the switch. The switchlight illuminates when an overheat condition is sensed within the APU powerplant enclosure. Illumination is a function of the APU monitor PC board. Power is always supplied to the switch, allowing the extinguishing agent to be discharged by the crew. Inadvertent pressing of the switch causes the APU fire bottle to be discharged.

"APU RELAY ENGAGED" ANNUNCIATOR—The annunciator illuminates to indicate that the start relay is energized closed. An indicator contact within the start relay provides the ECU with this information. The ECU then provides power for the APU RELAY ENGAGED annunciator to illuminate. The ground for the annunciator comes from the APU monitor PC board, so that the annunciator dims with the panel lights.

"APU FAIL" ANNUNCIATOR—The annunciator illuminates to indicate that the ECU has detected a fault, or the APU fire bottle pressure is low. If the ECU detects a fault, it supplies a ground into the APU monitor PC board on pin 33. The PC board supplies power and a ground to the annunciator for light illumination. If the APU fire bottle is low, a ground is supplied to the APU monitor PC board on pin 34. This input also causes the PC board to provide power and ground to the APU FAIL light.

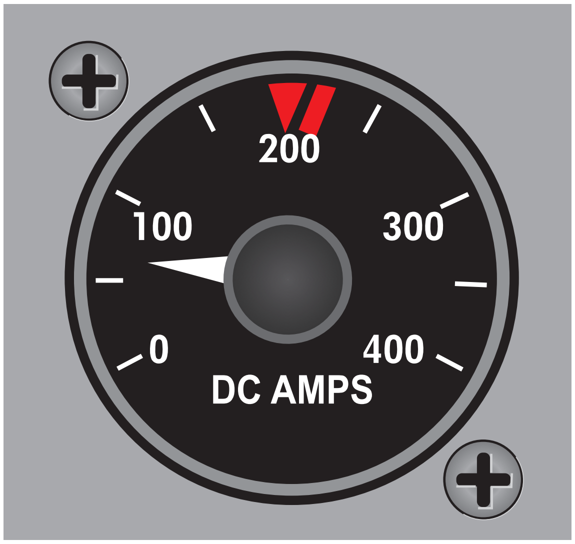
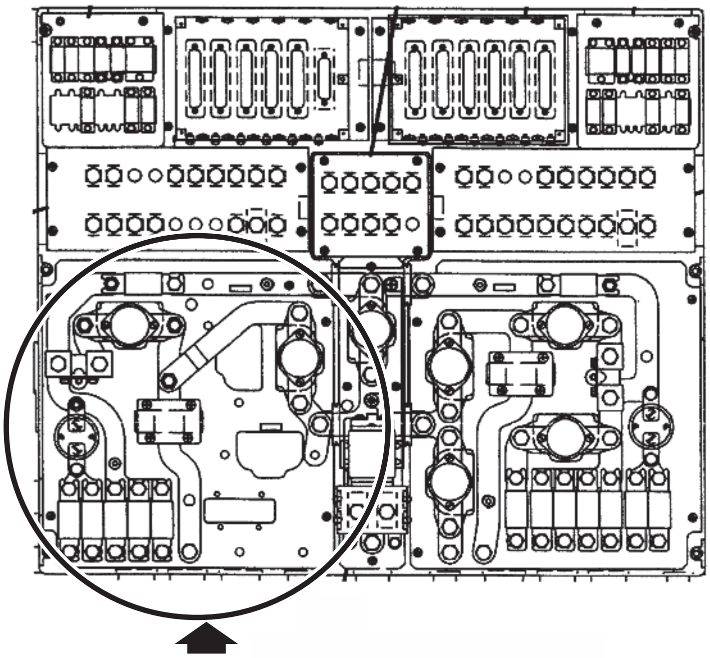
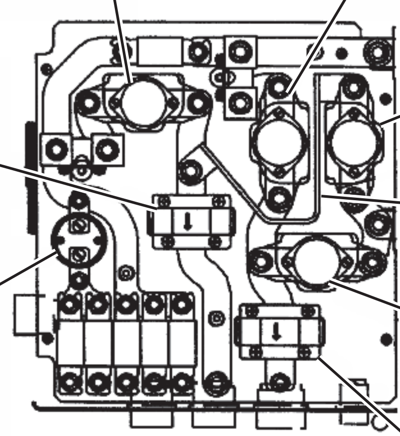


Figure 49-18. APU Ammeter





DETAIL A
LEFT POWER JUNCTION BOX RELAY PANEL ASSEMBLY (WITH APU)

Figure 49-19. Electrical Power Distribution

APU ELECTRICAL SYSTEM

General

The APU provides two types of power for aircraft use. Shaft horsepower is provided via the gearbox mount pad to drive the auxiliary generator for electrical power. The APU also provides pneumatic power (bleed air) to operate the aircraft environmental and service air systems. In general, these loads may be applied independently or simultaneously. In cases where both types of power are demanded, shaft power has priority. Selector switches in the cockpit initiate all load requirements. When the APU is running at 100% speed with no power being extracted, the APU is said to be at idle.

Electrical Power Distribution

Electrical power to start the APU is provided by the hot battery bus. Power is supplied out to the APU starter/generator through the APU start relay in the left power junction box (Figure 49-19). DC power provided by the APU is connected to the crossfeed bus by actuating the APU generator contactor (in the left power junction box). It is distributed through the rest of the normal aircraft DC distribution system.

NOTES

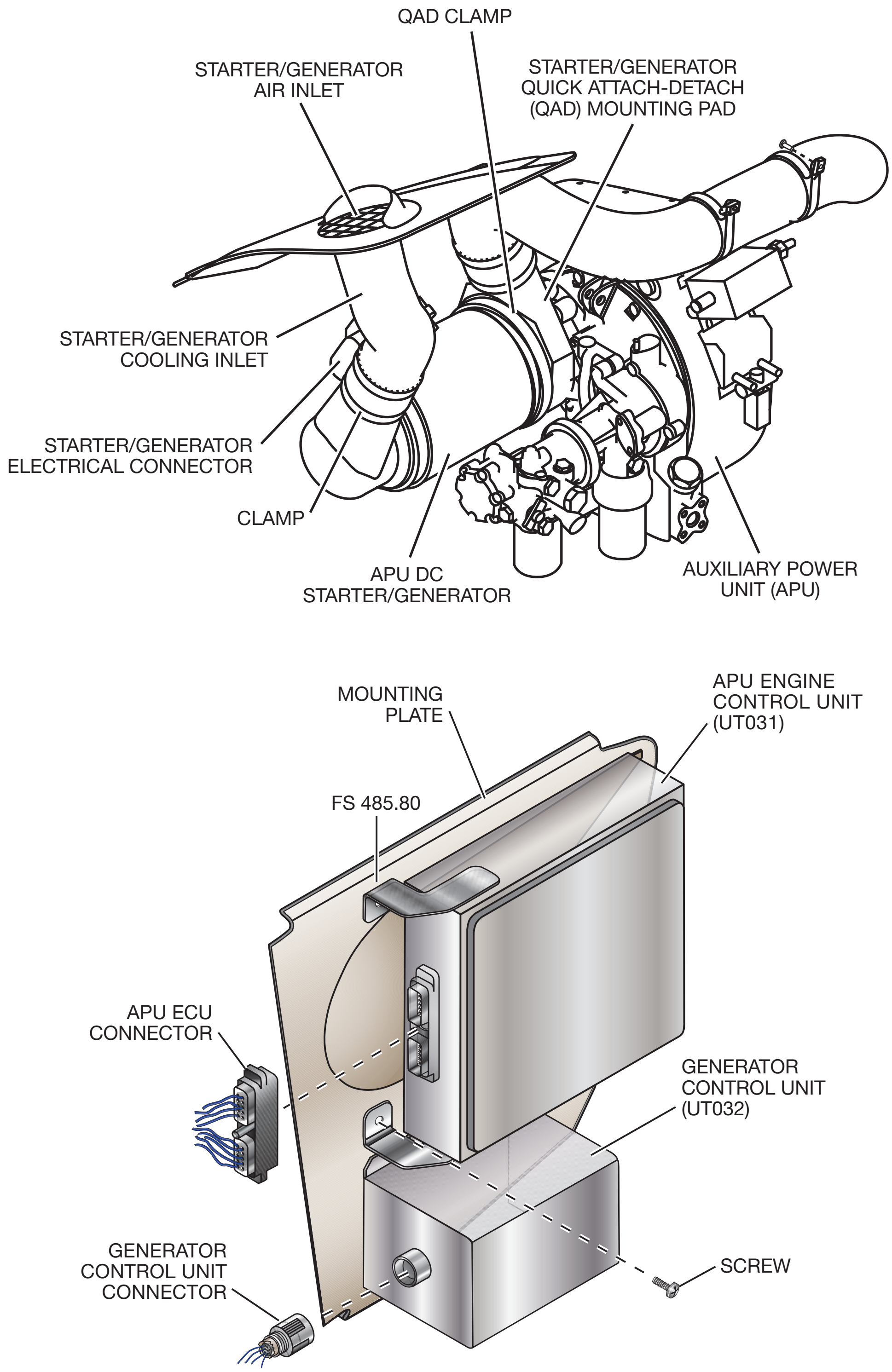


Figure 49-20. APU Electrical Power

Components

The components of the APU DC starter/generator system include (Figure 49-20):

- 300 amp DC starter/generator
- Quick attach/detach (QAD) mount
- Generator control unit (GCU)
- Ground fault protection circuits
- Various relays and switches

STARTER/GENERATOR—The starter/generator is on the front of the APU gearbox. It provides initial rotation of the APU and supplies DC electrical power to the aircraft. With the APU running and the generator online, electrical power is available for main engine starting and also for backup DC bus power. The APU DC starter/generator can be operated in parallel with the airplane DC generators through the crossfeed bus.

GENERATOR CONTROL UNIT—The GCU is in the DC generating system. The GCU utilizes solid-state integrated circuits and amplifiers to provide lightweight controls. The GCU consists of:

- Voltage regulation with automatic high accuracy load division
- Overvoltage monitor system
- Overexcitation protection
- Automatic line contactor control
- Reverse current protection
- Starter cut-off
- Field weakening
- Ground fault protection

The GCU is controlled by a microprocessor and utilizes built-in test (BIT) and nonvolatile memory for fault detection and isolation during start-up and in the running mode. The GCU has the capability of recording and displaying a "no fault" or 14 possible fault codes when the reset button is pressed and held. The four LEDs on the front of the GCU are used to indicate faults. The LED fault indicators are identified as:

- Generator Control Unit (GCU)
- Overvoltage (OV)
- Ground Fault (GF)
- System (SYS).

NOTES

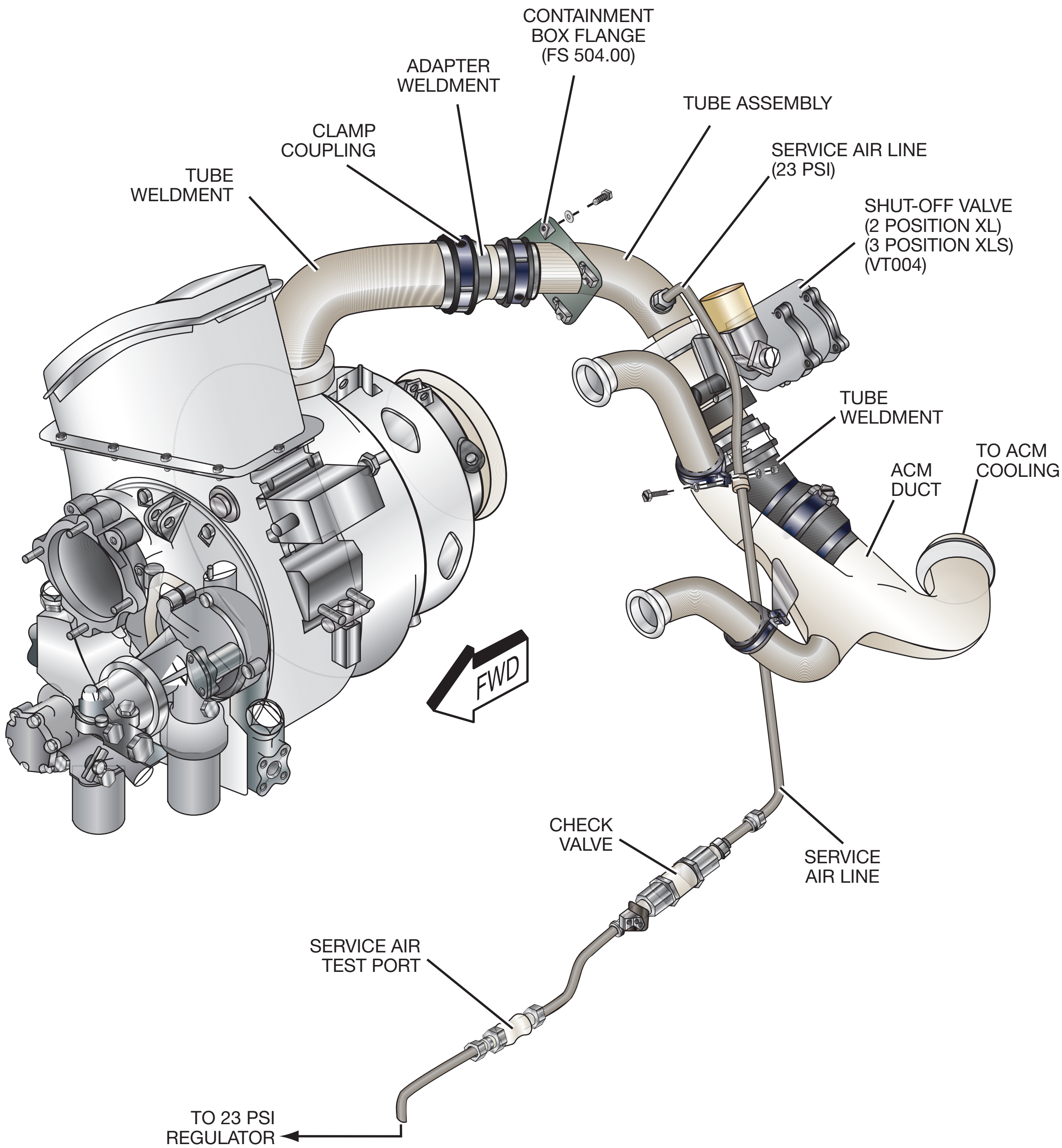


Figure 49-21. APU Pneumatic System

APU PNEUMATIC SYSTEM

Description

The APU provides compressed air (bleed air) for the aircraft ECU and the 23 psi service air system (Figure 49-21).

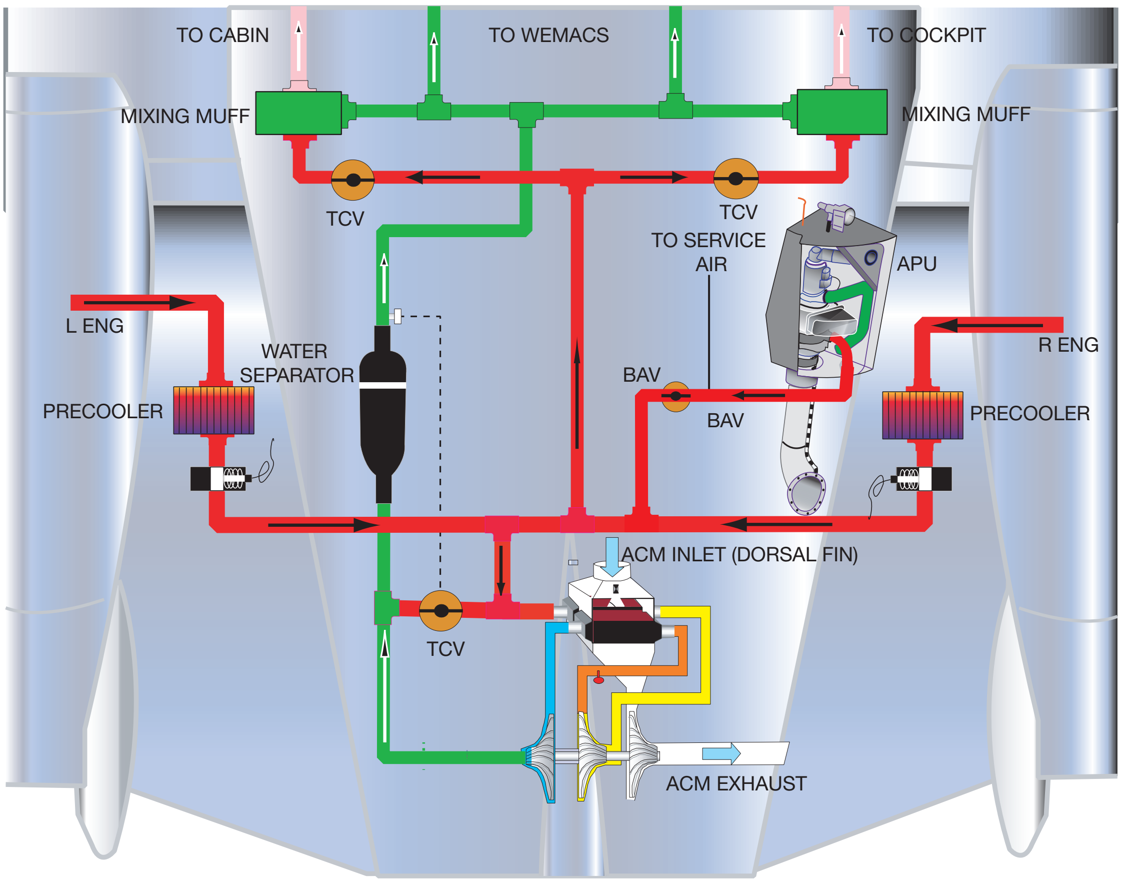
Bleed Air

Airflow passes from the APU bleed-air port to the aircraft ECU through the BLCV or LCV. Bleed air extraction from the APU is available once the APU reaches 95% speed and above, for 4 seconds. The BLCV is energized open by the ECU to either a half-open or to a full-open position, depending upon which pin the ECU supplies 28 VDC. Upon initial selection of the BLCV open, the ECU opens it to the half-open position for 20 seconds before progressing on to the full-open position. This allows the FCU to increase fuel flow stabilizing the APU at 100% rpm again, before the valve proceeds to the full-open position. The BLEED VAL OPEN annunciator illuminates when the BLCV has opened approximately 11–13% and remains illuminated as long as bleed air is being extracted. Since the electrical system is given priority over the pneumatic system, if the APU is not able to maintain 100% rpm, the BLCV closes (even though the BLEED AIR switch is still selected to the ON position). The BLEED AIR ON input is removed from the ECU anytime the ECU overheats, thus closing the BLCV and shutting down the air supply to the ECU.

NOTE

Due to the high flow rates of conditioned air available from the APU, selecting the APU bleed valve open with the cabin door closed may cause a small pressure "bump" in the cabin. Similarly, closing the cabin door with the APU bleed valve already open causes a pressure "bump" in the cabin.

NOTES



LEGEND

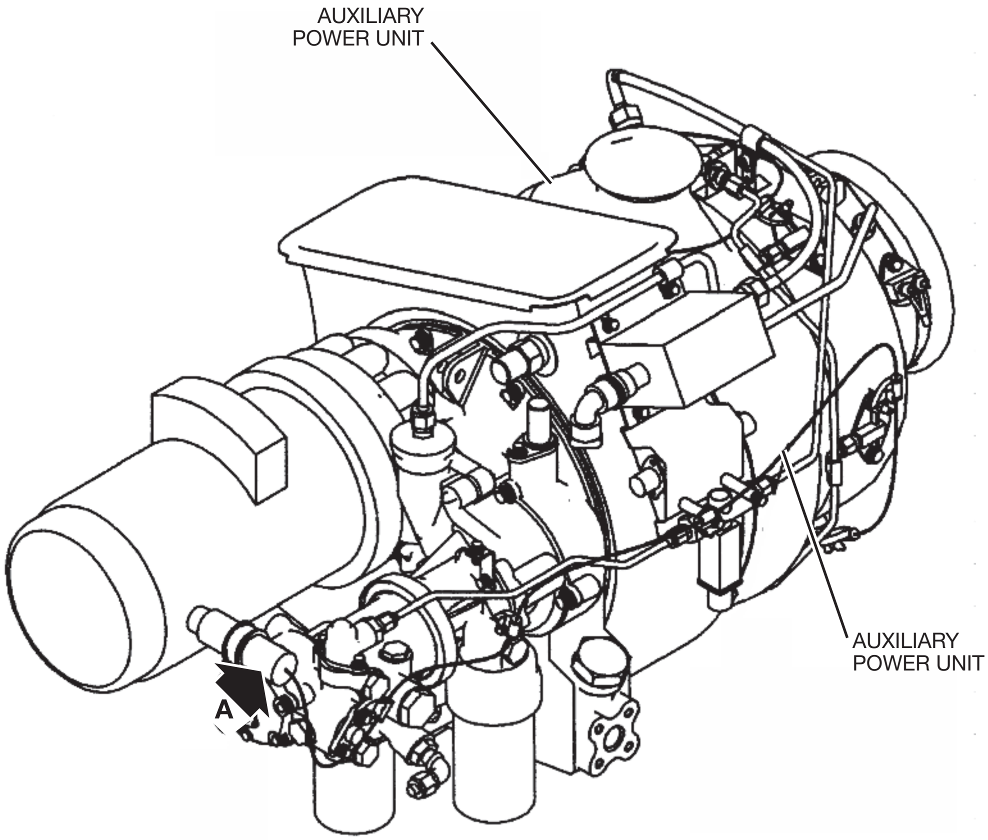
- CABIN/COCKPIT UNDER-FLOOR DUCTING
- COLD ACM AIR
- PRECOOLED BLEED AIR
- PRIMARY HEAT EXCHANGE AIR
- COMPRESSOR DISCHARGE AIR
- SECONDARY HEAT EXCHANGE AIR
- RAM AIR

Figure 49-22. APU Bleed-Air Schematic

Service Air

Bleed air is supplied into the 23 psi service air system anytime that the APU is operating (Figure 49-22). The bleed air is picked up before the BLCV, then routed through a check valve and supplied into the 23 psi air test port line T fitting. The test port T fitting is below the hydraulic reservoir in the tail cone.

NOTES



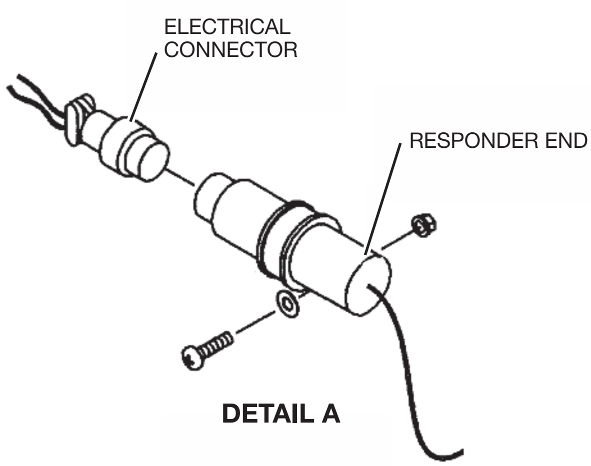


Figure 49-23. APU Fire Detector Assembly

APU FIRE DETECTION

DESCRIPTION

The fire detector assembly is routed around the APU powerplant at strategic points and includes an integral sensor element and a responder unit (Figure 49-23). The integral sensor element is constructed of stainless steel. The center core is charged with and retains a fixed volume of inert gas. An increase in temperature on any area of the sensor element, (which is routed around the APU) causes the inert gas to expand. The expansion of the gas actuates a switch in the responder unit. The responder unit incorporates two pressure switches that are permanently joined to a common sensor. The switches function as an alarm and integrity responder. When a fire is detected, this responder unit supplies 28 VDC to the ECU and to the APU monitor PC board on pin 22. With this input on pin 22, the APU monitor PC board supplies power out on pin 13 for APU FIRE switchlight illumination.

NOTES

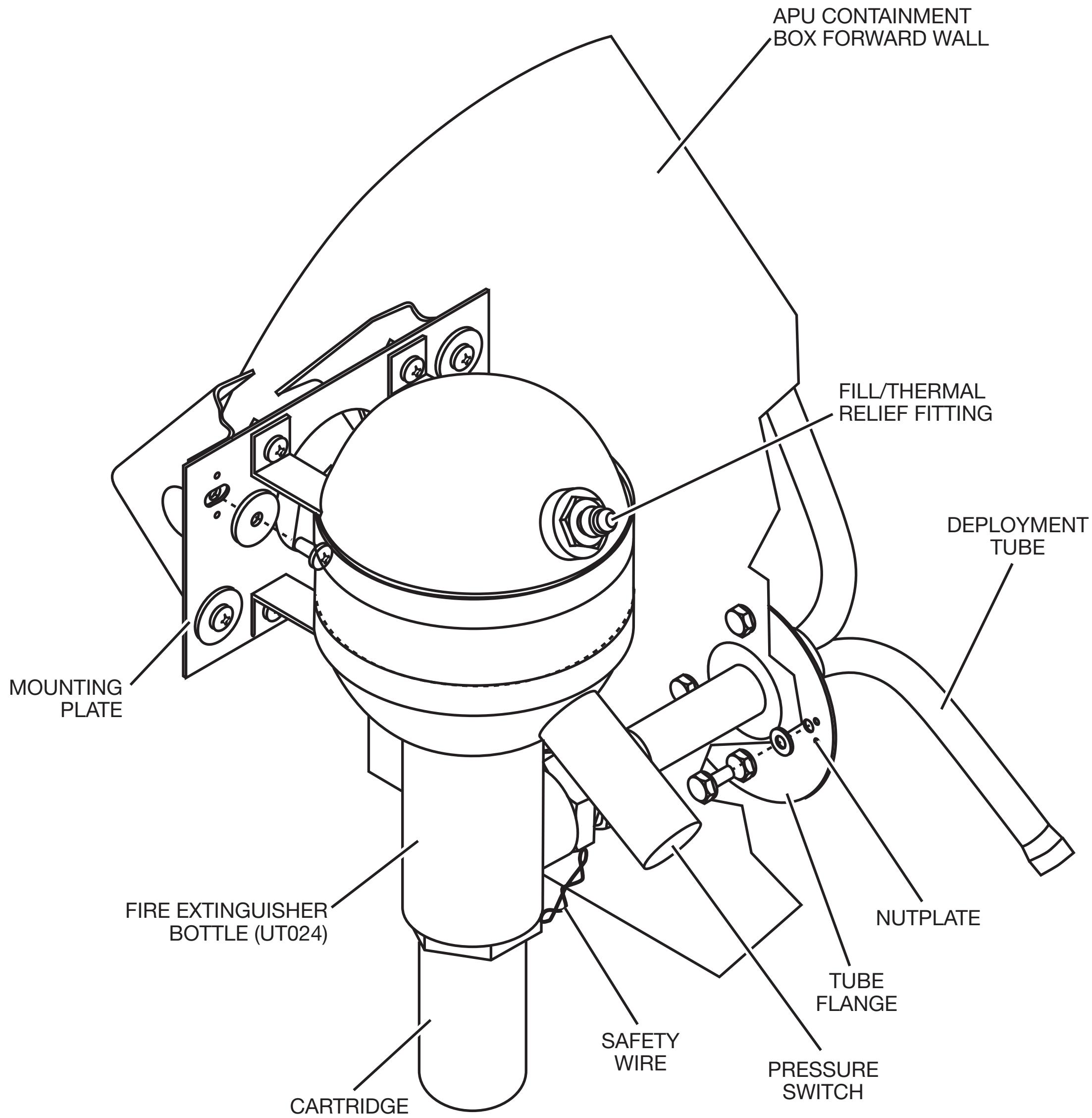


Figure 49-24. APU Fire Bottle

APU FIRE EXTINGUISHING

DESCRIPTION

The APU is completely enclosed in a fire containment box made of titanium and stainless steel. Access to the APU is gained through a door on the right side of the fuselage. The fire extinguishing system deploys extinguishing agent from a single fire extinguisher bottle into the APU fire containment box, in the event a fire is detected by the associated fire detection system. This bottle is below the firewall fairing and dispenses extinguishing agent via a single deployment tube. The deployment tube is routed through the firewall fairing and terminates at a "T" fitting, which disperses the fire-extinguishing agent within the APU enclosure.

COMPONENTS

Fire Bottle

The fire bottle assembly consists of a steel cylinder, fill/thermal relief port, aluminum discharge outlet, and pressure switch (Figure 49-24). The bottle contains 1.0 pound (0.45kg) of Halon 1301 which is pressurized by dry nitrogen at 600 psig, +25 or –0 psig.

FILL/THERMAL RELIEF—The fill/thermal relief port is on the upper portion of the bottle. This port also incorporates a thermal relief valve which ruptures if internal bottle temperature exceeds between 205°F to 226°F at a pressure of between 1520 and 1710 psi.

DISCHARGE OUTLET—The discharge outlet is on the bottom of the bottle and contains a pyrotechnic device, which is fired manually by the cockpit APU FIRE switchlight or automatically 8 seconds after a fire is detected. A 28VDC signal is sent to the pyrotechnic device when either the APU FIRE switchlight is depressed or by the APU monitor PC board 8 seconds after it receives an input from the fire detection system. The resulting explosion ruptures a diaphragm inside the discharge outlet. This rupture allows rapid expulsion of the pressurized Halon through the discharge outlet and into the discharge tube.

PRESSURE SWITCH—There is a pressure switch on the lower portion of the bottle. This switch is wired into the APU monitor PC board and supplies a signal to the PCB when bottle pressure drops below 500 psig, ± 30 psig at 70°F.

WARNING

The fire extinguisher bottle cartridge is a pyrotechnic device. Inadvertent detonation can cause personal injury. Always remove electrical power from the airplane, disconnect electrical connector from the cartridge and immediately install shunt plug/wire over cartridge electrical connector pins prior to removing/handling the fire bottle. Also avoid maintaining the fire extinguisher bottle near active radio broadcasting equipment, radar equipment, high voltage lines or during electrical storms.

WARNING

The fire extinguisher bottle discharges 8 seconds after receiving a signal from the fire detection loop.

Diagnostics

The following is verified by depressing the TEST button on the APU control panel:

- The integrity of the entire fire detector assembly
- The condition of the sensor
- Fire extinguisher bottle for adequate extinguishing agent/pressure

Activation of the test circuit illuminates the APU FIRE switchlight.

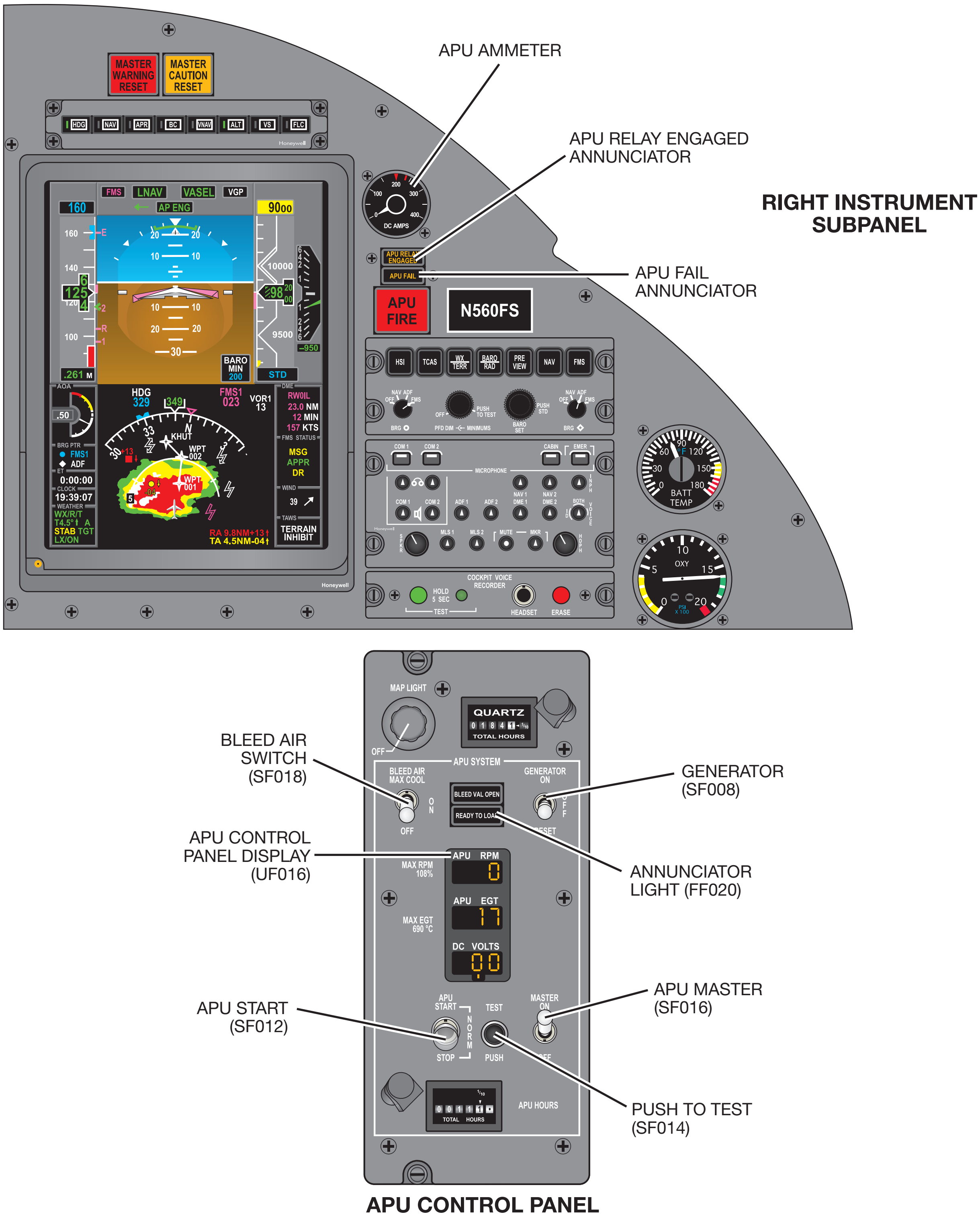


Figure 49-25. APU Control and Indication (XLS)

APU OPERATION

To begin an APU start sequence, momentarily place the START/STOP switch in the START position. Three start signals are sent from the APU control PC board:

- One signal to the GCU
- One signal to the ECU
- One signal to the main engine start PC boards

An OK TO START signal is returned to the APU control PC board from the GCU and the ECU. The signal to the main engine start PCB is for main generator start power when assisting the start of the APU with a main engine generator online (Figure 49-25).

After the OK TO START signals are received the APU control PC board closes the start relay, which supplies current to the starter/generator. The start sequence can be terminated either manually (by momentarily placing the START/STOP switch in the STOP position) or automatically (by the removal of either OK TO START signal from the PC board). If the start sequence is not terminated manually, the GCU continues to supply power through the start relay until the APU reaches 60% speed, at which point the ECU discontinues the start sequence and the APU accelerates under its own power. When the speed of the APU reaches 95% speed (+4 seconds) the READY TO LOAD light illuminates, indicating that bleed air and/or electrical power is available.

NORMAL PROCEDURES

Exterior Inspection

1. APU engine and generator cooling inlet—CLEAR
2. APU exhaust—CLEAR
3. APU drain—CLEAR
4. APU oil level—CHECK
    a. APU "ADD OIL" light (amber) illuminated—APU may be operated. Service the oil at the next available opportunity.
    b. APU "LOW OIL" light (red) illuminated—APU may not be operated until oil is serviced.

APU Starting

1. APU master switch—ON
2. APU FAIL light—VERIFY OFF

NOTE

If APU FAIL light is illuminated, do not attempt APU start.

3. APU TEST button—Push, verify the following and release:
    a. Fire warning light illuminated
    b. APU FAIL light illuminated
    c. APU RELAY ENGAGED light illuminated
    d. BLEED VAL OPEN light illuminated
    e. READY TO LOAD light illuminated
4. APU GENERATOR switch—OFF
5. APU BLEED AIR switch—OFF
6. APU START/STOP switch—START (momentarily)
7. APU RELAY ENGAGED light—ILLUMINATED, then EXTINGUISHED prior to the READY TO LOAD light illuminating

NOTE

If the main generators are on and operating, the main engine start lights illuminate when starting the APU on the ground. With the aircraft in air mode, only the battery provides power for start.

8. READY TO LOAD light—ILLUMINATED (start is complete)
9. APU generator—AS DESIRED
10. APU ammeter—CHECK (200 amps maximum)
11. APU BLEED AIR Switch—AS DESIRED

APU Shutdown

1. APU START/STOP Switch—STOP POSITION (momentarily)
2. APU READY TO LOAD Annunciator—EXTINGUISHED

NOTE

Hot section service life will be maximized when the APU is shut down in its existing load configuration. If the APU is operating in the loaded condition, shutdown is accomplished directly from the loaded condition.

3. APU BLEED AIR Switch—OFF
4. APU GENERATOR Switch—OFF
5. APU MASTER Switch—OFF, after RPM indicates 0% RPM

COLD WEATHER PROCEDURES

APU cold weather starting has been demonstrated to –30°C using aircraft battery. Starts may be attempted below this temperature, but repeated attempts drain the battery to the point where a main engine start using the battery is not possible. A GPU is then required.

After the APU start has been completed (READY TO LOAD light illuminated) the APU generator and bleed valve may be operated in a normal manner as desired.

For starting the APU below –40°C, it is recommended that Type I (MIL-L-7808) lubricating oil be utilized.

NOTES

APU LIMITATIONS

- APU operation is prohibited until a satisfactory APU test has been accomplished.
- Starting the APU is prohibited when the APU fail light is illuminated.
- APU start attempt is prohibited after a dual generator failure.
- Following shutdown for any reason, APU restart must not be attempted until 30 seconds after the rpm indicator reads 0%.
- Applying deice (anti-ice fluid of any type) is prohibited with APU operating.
- Deployment of the thrust reversers for more than 30 seconds with the APU running is prohibited.
- The APU is not approved for unattended operation.
- The following limits (Table 49-2) apply to APU starting and operation:

Table 49-2. APU LIMITATIONS

OPERATING CONDITION	START	RUN
MAX ALT FT	20,000	30,000
MAX EGT (°C) (NOTE 2)	690	690
N_1%	---	108
FUEL TEMP °C	ACFT FUEL LIMITATIONS	ACFT FUEL LIMITATIONS
MAX GEN AMPS (NOTE 1)	---	200 GND 230 FLT
AMB TEMP °C	± 54	± 54

NOTE

Transient current greater than 200 amperes is approved for APU cross generator start of the main engines. APU ammeter instrument markings: Red triangle = 200 amps, Red line = 230 amps.

NOTE

The APU automatically shuts down if ITT limits are exceeded.

Starter Limitation—Three APU start cycles per 30 minutes. Three cycles of operation with a 90-second rest period between cycles is permitted.

Battery Limitation—Nine APU start cycles per hour. (An APU battery start counts as 1/3 of a normal engine battery start).

NOTE

On the ground, no battery cycle is counted when starting the main engines with a cross generator start from the APU or from a GPU.

Use of an external power source with voltage in excess of 28 VDC or current in excess of 1000 amps may damage the starter. Minimum amps for start: 800.

If battery limitation is exceeded, a deep cycle including a capacity check must be accomplished to detect possible cell damage.

QUESTIONS

1. The APU generator limits are:
   A. 200 amps ground/300 amps flight
   B. 230 amps ground/300 amps flight
   C. 200 amps ground/230 amps flight
   D. 230 amps ground/230 amps flight

2. The maximum altitude is:
   A. 45,000 feet for APU operations
   B. 30,000 feet for APU operations
   C. 30,000 feet for APU starting
   D. 25,000 feet for APU starting

3. Select the false statement:
   A. The APU fire bottle discharges automatically 8 seconds after a fire is detected
   B. The APU fire bottle discharges at any time if selected by the pilot
   C. With a dual generator failure inflight, the APU should be started to supply electrical power
   D. Low fire bottle pressure illuminates the APU FAIL annunciator

4. Select the correct limitation:
   A. APU start attempt is prohibited after a dual generator failure
   B. Deployment of thrust reversers is prohibited with the APU operating
   C. The APU is approved for unattended operation
   D. The aircraft battery is limited to three APU starts per hour

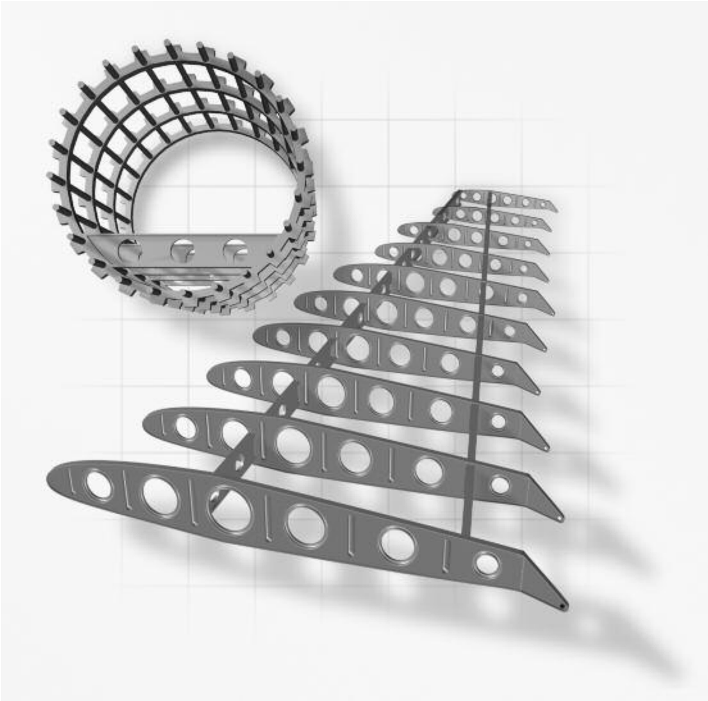
5. Select the correct statement:
   A. The APU bleed air valve does not close in response to an ECU overheat
   B. APU fuel is normally taken from the left tank (crossfeed off)
   C. APU fuel can be taken from the left tank (crossfeed open)
   D. The only method of manual shutdown of the APU is the START/STOP switch on the APU SYSTEM panel

CHAPTER 51-57
STRUCTURES

CONTENTS

ILLUSTRATIONS

CHAPTER 51-57 STRUCTURES



INTRODUCTION

This chapter provides a description of the aircraft structures, doors, fuselage, nacelles, pylons, stabilizers, windows, and wings on the Citation XL/XLS/XLS+ aircraft.

GENERAL

This chapter is divided into seven sections briefly described below. For locating subjects within the sections, refer to the Table of Contents at the beginning of this chapter.

This chapter provides an overall description of the nose, cabin, fuselage, tail cone, empennage, and wing.

The Doors section provides a general description of the cabin entry door, emergency exit door, baggage door, nose compartment and service doors, auxiliary power unit door, and the door warning system.

The Fuselage section describes the structural compartments for equipment, flight crew, passengers, and baggage. Included are the main frame, nose radome, plates/ skin, auxiliary structure, and aerodynamic fairings.

The nacelles/pylon section describes the construction of the pylons, and the engine attach fittings.

The Stabilizers section includes information on the horizontal stabilizers, elevator, vertical stabilizer, rudder, and attach fittings.

The Windows section provides information on the windshields and flight compartment windows, and cabin windows.

The Wings section includes information on the main frame, auxiliary structures, and flight surfaces.

DESCRIPTION

The Citation XL/XLS/XLS+ is a pressurized executive aircraft employing a straight wing and powered by two turbofan engines. It has comfortable seating for 6 to 8 passengers with suitable allowance for luggage and optional equipment.

COMPONENTS

Powerplant

The aircraft has two engines on the sides of the tail cone and a low wing (slightly below the bottom of the fuselage) to provide maximum usable cabin space and maximum wing fuel reservoir volumes. Keel beams on the lower surface of the carry-thru beams provide protection in the event of a gear-up landing.

Landing Gear

The main landing gear retracts inboard into the wing and fuselage. The landing gear well envelope is under the fuselage, and in the wing inboard of the wing planform break. The nose gear retracts forward into the nose section of the fuselage.

Doors

Airplane entry is through a one-piece door slightly aft of the cockpit (on the left side of the fuselage). The over-wing emergency exit is on the right side of the fuselage.

Fuselage

The nose section of the fuselage is unpressurized. A bonded skin and stringer construction achieves aerodynamic and aesthetic smoothness as well as weight reduction. The nose compartment provides space for avionics equipment, crew baggage, and the nose wheel well. Avionics equipment is accessed through doors on the left and right sides of the nose. The radome is removable for access to the radar antenna. The nose gear retracts forward to allow gear loads to feed directly into the structure of the forward pressure bulkhead.

The cabin size meets requirements for low drag and passenger comfort. The aircraft has a flat cabin floor except for a dropped center aisle. The dropped aisle extends from the cockpit divider aft to the aft pressure bulkhead for extra stand up height.

The nonpressurized tail cone contains a baggage compartment (aft of the aft pressure bulkhead and below the engine carry-thru beams). The baggage compartment is accessed through a hinged door on the left side of the tail cone.

Windows

There are six windows on the right side of the cabin and five on the left. The cabin and cockpit areas are pressurized compartments. A forward pressure bulkhead separates the nose section and cockpit; and an aft pressure bulkhead separates the cabin and tail cone. These sections form the pressurized area of the fuselage.

Stabilizers

The empennage structure is based on semimonocoque design consisting of spars, stringers, ribs, and skin. The vertical fin has a sweep back of 32°, 54 minutes along the quarter chord. The horizontal stabilizer has 9° dihedral and a sweep of 0° along the 68% chord line. The horizontal stabilizer is positioned either 1° noseup or 2° nosedown by a hydraulic actuator.

Wings

The wing structure is based on semi-monocoque design consisting of spars, stringers, ribs, and skin. The main spar is at 35% chord and the rear spar is at 65% chord. Both spars are of similar construction and consist of cap extrusions, stiffeners, and webs. The outboard wing is attached to a center carry-thru structure that passes beneath the pressure vessel. The entire wing is attached to the fuselage by five attach fittings.

NOTES

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CHAPTER 52
DOORS

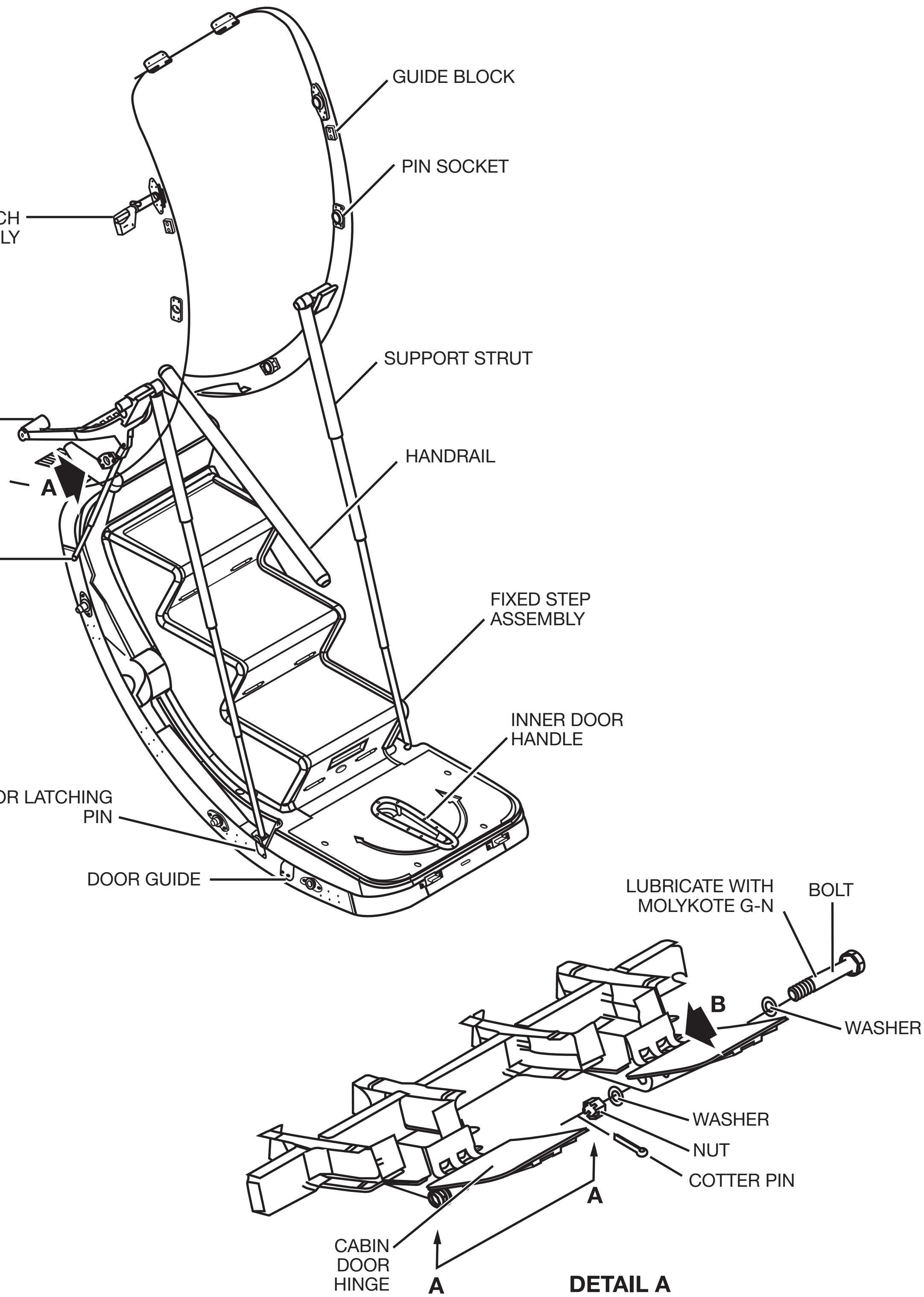


Figure 52-1. Cabin Entry Door Installation (Sheet 1 of 2)

CABIN ENTRY DOOR

Description

The cabin entry door is on the left forward side of the aircraft (between FS 190.20 and FS 213.80) (Figure 52-1). Stairs incorporated with the door provide entry into the cabin and cockpit when the door is opened.

The major components of the main cabin entry door are:

- Fixed steps
- Folding handrail
- Door actuating mechanism
- Locking mechanism
- Primary (inflatable) and secondary pressure seals
- Cabin door vent
- External key lock
- Cabin door monitor warning system

NOTES

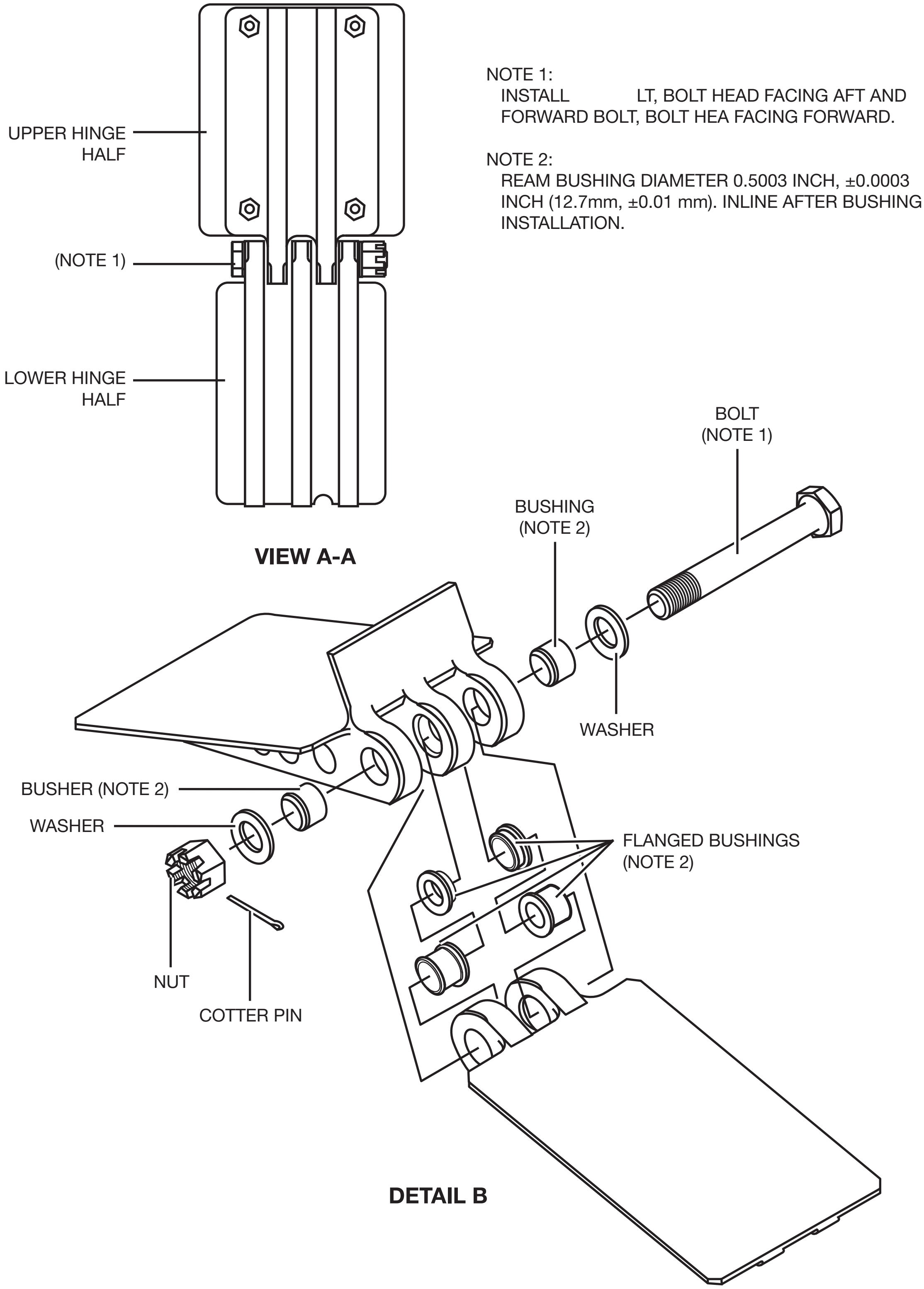


Figure 52-1. Cabin Entry Door Installation (Sheet 2 of 2)

Components

The cabin door is hinged at the bottom. It opens outward (Figure 52-1). As the door opens, steps and a handrail provide access to the cabin. Eight locking pins are mechanically linked to internal and external lock/unlock handles. There are three locking pins on each side of the door and two pins at the top, which extend into the door frame (sockets) to secure the door in the closed position.

The lower aft locking pin actuates the door seal valve, which controls pneumatic air to the primary (inflatable) door seal. In addition to the inflatable seal, a secondary pressure door seal compresses outward against the door, to preserve cabin pressure if the primary (inflatable) seal fails. The door is centered in the door frame by nylon door guides (two on each side). The nylon guides mate with corresponding nylon guide blocks on the door frame. No maintenance is necessary for the guides or blocks, except for replacing worn or damaged parts.

The cabin door incorporates a counterbalance mechanism, which assists in lowering and raising the door. The counterbalance consists of two coiled springs on separate shafts connected to cable reels. Left and right cables are connected to the cable reels. They wind/unwind on the cable reels. The other end of the cables connect to fuselage door frame fittings. When the door is lowered, the tension of the coiled springs increases as the cables unwind from the reels. The spring tension counterbalances the weight of the door. When the door is raised, the tension of the coiled springs decreases, and counterbalances the weight of the door as the cables wind onto the reels.

Controls and Indications

Six switches monitor the door closed-and-locked position. The door locking pins (upper and lower forward and upper and lower aft) actuate four switches. The fifth switch is a proximity switch at the door vent. The sixth switch at the bellcrank linkage, monitors door-lock condition. The switches are electrically connected to the warning lights.

Operation of the inflatable door seal is monitored by a differential pressure switch on the bottom of the cabin door channel (at FS 202.00). When air pressure within the seal drops below 5 psi, an electrical signal is sent to the DOOR SEAL annunciator (XL/XLS) or the amber CABIN DOOR SEAL CAS message (XLS+) (Figure 52-2).

DOOR SEAL
Annunciator steady on ground, flashes in flight, if the door seal pressure drops below 5.5 psi, activates MASTER CAUTION lights. Annunciator will extinguish if door seal pressure increases to approximately 8.5 psi.

XL/XLS ANNUNCATOR

CABIN DOOR SEAL			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the pressure in the cabin door seal is less than 5 PSI.			

XLS+ CAS MESSAGE

Figure 52-2. Cabin Door Seal Indications

Nine indicating windows (inspection holes) in the door upholstery allow a visual check of the indicator assemblies for the cabin door bolting system, and the flag assembly for the outside cabin door handle linkage. With the cabin entry door locked, the indicator assemblies are directly under their respective indicating window. A phosphorescent green disc is fully and clearly visible through the indicating window when the outside cabin door handle linkage is in the door-locked position.

Door Warning System

The door warning system signals the flight crew if any of the following doors are not locked in the stowed-open position:

- Main entrance door
- Emergency exit door
- Nose compartment doors
- Tail cone baggage door
- Forward tail cone maintenance access door
- Lavatory doors

This section covers warning switches and their components.

The door warning system consists of door switches that provide door position information to the annunciators.

The cabin entrance door warning system is comprised of:

- Four door pin switches (SC014, SC016, SC048, and SC049)
- Door handle switch (SC026)
- Vent door proximity switch (SC046)
- Door seal switch (SC028)
- Cabin door monitor switch (SC066
- Cabin door monitor printed circuit board (NZ007)
- CABIN DOOR annunciation
- DOOR SEAL annunciator

The cabin-door monitor printed circuit board (PCB) monitors the status of:

- Four door pin switches
- Cabin door handle switch
- Vent door proximity switch

When the cabin door is closed, the cabin door monitor switch supplies emergency power to the PCB for 30 minutes (± 5 minutes) allowing enough time for normal power up without resequencing the door closure procedure. The CABIN DOOR annunciator (XL/XLS) or the amber CABIN DOOR CAS message (XLS+) illuminates if either of the following occurs (a ground is provided by the PCB) (Figure 52-3):

- Any of the four door pin switches or the door handle switch does not indicate closed
- The cabin vent door target has moved away from the proximity switch

CABIN DOOR
Annunciator flashes to indicate the cabin door is not properly locked (possible disengagement of a cabin door pin or vent door not closed). Check visual indicators on door frame. Annunciator remains steady on ground prior to engine start.

Annunciator flashes after engine starts and in flight, activates the MASTER CAUTION lights.

XL/XLS ANNUNCATOR

CABIN DOOR			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
This message is displayed when the cabin door is open.			

XLS+ CAS MESSAGE

Figure 52-3. Cabin Door Indications

The CABIN DOOR annunciator (XL/XLS) or the amber CABIN DOOR CAS message (XLS+) extinguishes only after the proper closure sequence occurs. The proper sequence is as follows:

1. Door is open (including all pins, handle and vent door).
2. All pins and handle locked.
3. Vent door is open at least 0.5 seconds after the handle is locked.
4. Vent is door closed.

The door seal switch provides an electrical ground, causing the DOOR SEAL annunci-

ator to illuminate when door seal pressure is less than 5 psi (34.5 kPa).

Lavatory Door Warning System

The lavatory door warning system is comprised of four switches (S11F and S13F for the left divider and S12 and S14F for the right divider) and a LAV DOOR annunciator (XL/XLS) or the amber LAVATORY DOOR CAS message (XLS+) (Figure 52-4).

LAV DOOR
Annunciator flashes if the lavatory door(s) is/are not latched open prior to takeoff, or in flight with the flaps not up (zero degrees). Activates MASTER CAUTION lights.

XL/XLS ANNUNCATOR

LAVATORY DOOR			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the lavatory door is closed and the aircraft is on the ground or flaps out of 0° position.			

XLS+ CAS MESSAGE

Figure 52-4. Lavatory Door Indications

The lavatory doors are monitored by switches that provide a ground to illuminate the LAV DOOR annunciator (XL/XLS) or the amber LAVATORY DOOR CAS message (XLS+) anytime the aircraft is on the ground or in flight with the aircraft flaps not up, and lavatory doors are not in the latched-open position. The LAV DOOR annunciator does not illuminate when the flaps are up and the aircraft is in flight.

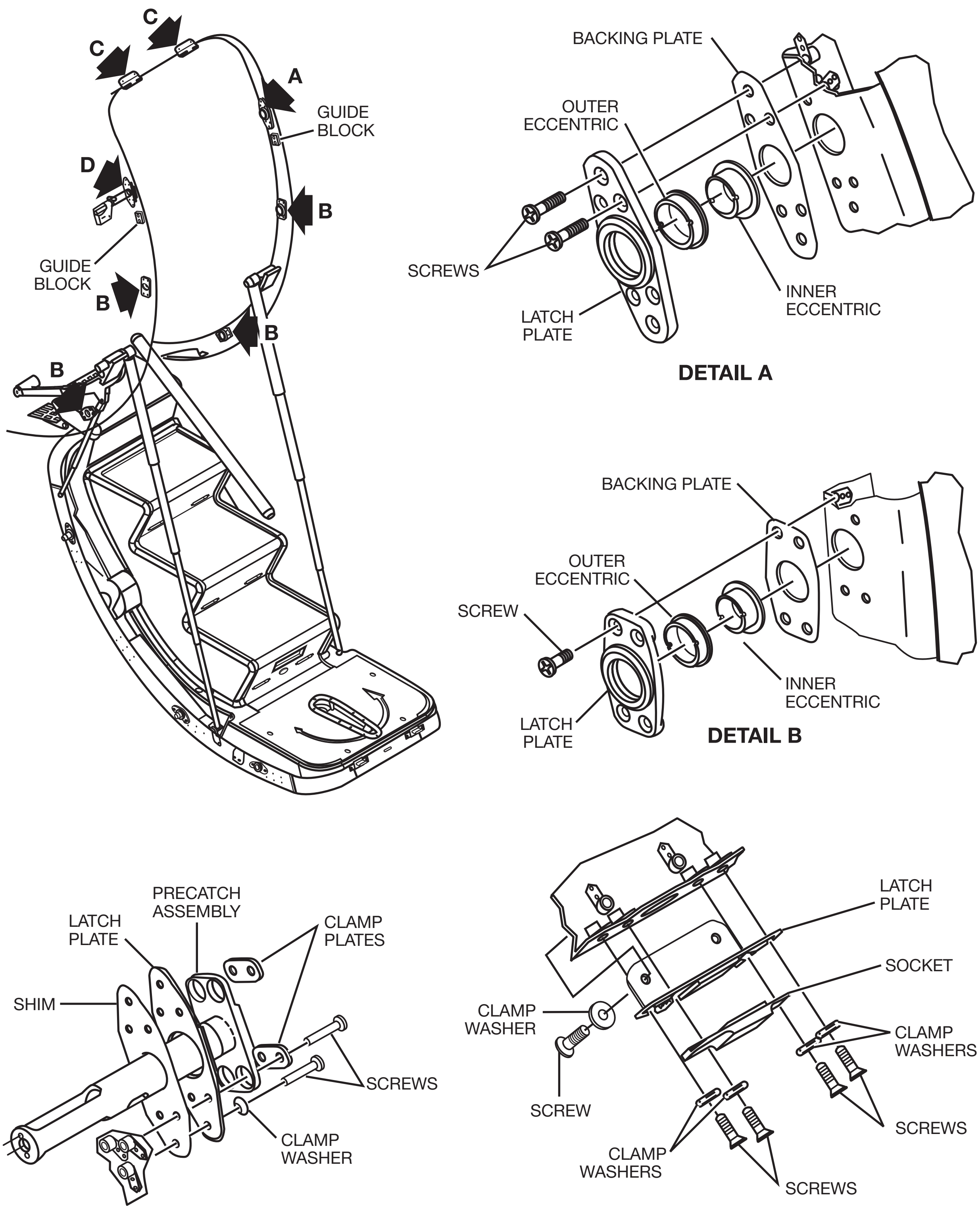


Figure 52-5. Cabin Entry Door Sockets Adjustment

Operation

Opening the Cabin Door From Inside

The flush inside cabin door handle is in the lower door step. The lower step is at the top of the cabin entry door when the door is closed. To open the cabin entry door from the inside:

1. Push a button to displace the spring-loaded handle.
2. Then pull out on the small end of the handle.
3. Rotate the handle 90° counterclockwise to unlatch the door.

The locking mechanism retracts the lockpins from the door frame sockets (Figure 52-5).

If electrical power is available when the lockpins retract, the warning switches actuate to complete an electrical ground. This illuminates the CABIN DOOR annunciator (XL/XLS) or the amber CABIN DOOR CAS message (XLS+). The annunciator illuminates momentarily in the flashing mode. This is a normal check of the flasher circuit. At the same time, the lower aft lockpin permits the door seal valve to dump pneumatic pressure from the inflatable door seal (allow 3 to 5 seconds for seal deflation). When the DOOR SEAL annunciator (XL/XLS) or the amber CABIN DOOR SEAL CAS message (XLS+) illuminates, push on the upper part of the door to start the door outward and down, and move the door handle downward. The counterbalance mechanism prevents free fall of the door. An overcenter locking linkage of the door handle and two telescoping support struts provide solid footing when entering and exiting the cabin.

If the cabin door is held closed only by the precatch, or if the precatch engages when door is unlatched, release the precatch by pulling the red handle, just forward of the door.

NOTE

Precatch should not engage when door is unlatched. If precatch engages when door is unlatched, check for improper adjustment or binding.

NOTES

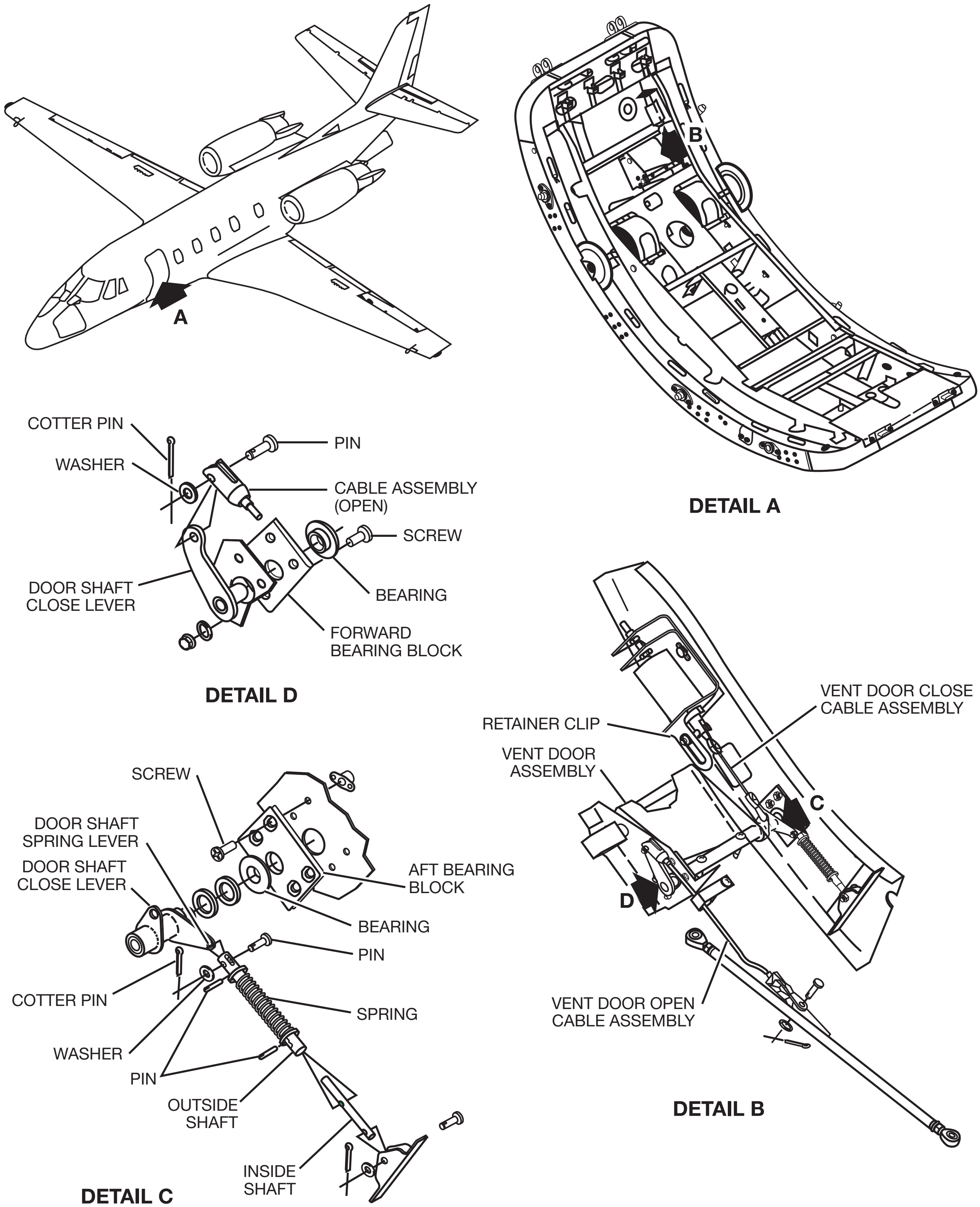


Figure 52-6. Vent Door Installation

Closing the Cabin Door From Inside

1. Lift the door handle (actuating lever) from the overcenter position to close the cabin door from inside the cabin.
2. Continue to lift the actuating lever to raise the door to the closed position. As the door is raised, the counterbalance mechanism assists the closing effort, and the handrail folds toward the door.
3. Then pull on the pull handle (on the aft side of the lower step) with sufficient force to engage the precatch
4. Push the button to displace the inside cabin door handle, and rotate the handle 90° clockwise to latch the door.

The locking mechanism extends the lockpins into the door frame sockets. If electrical power is available when the lockpins extend, the pin switches close, extinguishing the CABIN DOOR annunciator (XL/XLS) or the amber CABIN DOOR CAS message (XLS+). At the same time, the lower aft lockpin opens the door seal valve. This permits inflation of the primary pressure door seal if pneumatic air (engine-bleed air) is available. When the bolting pins are fully engaged, indicating windows provide a visual indication to verify that the door is locked. The door latching mechanism provides a mechanically prevents cabin pressurization until the cabin entry door is fully latched and locked. A small vent door (Figure 52-6), behind the fairing (above the spade door), prevents aircraft pressurization until the bolting system is fully locked.

Opening the Cabin Door From Outside

If the cabin door is key locked:

1. Insert the key in the key lock below the outside cabin door handle and rotate the key 90° counterclockwise.
2. Pull the flush-mounted handle outward by the finger hole (in the small end) and rotate the handle 90° clockwise to unlatch the door.
3. Pull the door outward and down.
4. As the door lowers, the handrail moves into proper position.
4. When the door reaches its lowest point, place a foot on the lower step; and with pressure on the step, press the handrail down until the inside actuating lever linkage is overcenter.

If the cabin door is held closed only by precatch, or if precatch engages when the door is unlatched, release precatch by pressing the PRECATCH RELEASE button (just forward of the door).

NOTE

Precatch should not engage when door is unlatched. If precatch engages when door is unlatched, check for improper adjustment or binding.

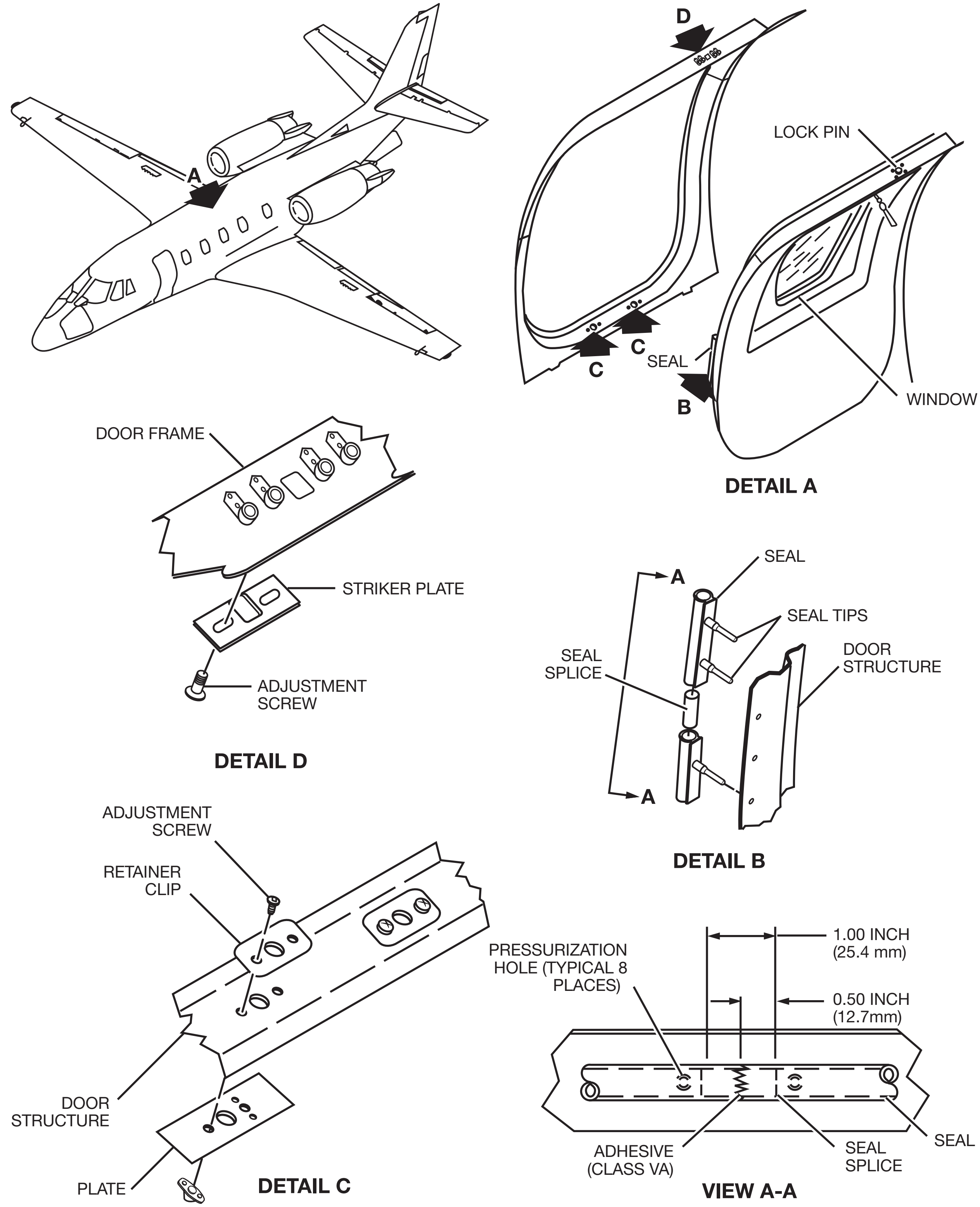


Figure 52-7. Emergency Exit Door Installation

Closing the Cabin Door From Outside

WARNING

Do not stand directly in front of the lower step when releasing the cabin entry door from the locked-down position. The door may spring upward, striking the legs.

To close the cabin door from the outside:

1. Stand to the side of the door (near the handrail) and lift up on the handrail. This releases the actuating lever linkage from overcenter.
2. With the aid of the counterbalance mechanism, lift the door to the closed position, using enough force to engage the precatch.
3. Pull the outside cabin door handle outward by the finger hole and rotate the handle 90° counterclockwise to latch and lock the door.
4. If desired, key lock the cabin door, by rotating the key 90° clockwise with the handle in the stowed position.

EMERGENCY EXIT DOOR

The emergency exit door is on the right side of the fuselage (between FS 346.146 and FS 370.038) (Figure 52-7). The door installs from the inside of the cabin and incorporates a passenger compartment window. Refer to Chapter 56—"Cabin Windows" for window removal/installation.

Description

The emergency exit door is constructed of:

- Frames
- Stiffeners
- Doublers
- Skin panel

There is a rubber bulb-type pressure seal around the outer periphery of the door.

The emergency exit door is held in place by a lockpin (at the top center of the door) and two adjustable alignment pins (at the bottom of the door) which align with retainer clips in the door frame. The lockpin is actuated by a latching mechanism, and inside and outside handles.

The outer door handle housing contains a moisture drain. The inside surface of the outer handle/housing is Teflon-coated to prevent ice adhesion.

Controls and Indications

The emergency exit door is monitored by a proximity switch (SC056). When the emergency exit door is unlocked and the target on the door is moved away from the proximity switch, the switch closes, providing a ground for EMER EXIT annunciator (XL/XLS) or the amber EMERGENCY EXIT CAS message (XLS+) illumination (Figure 52-8).

EMER EXIT
Annunciator flashes if the emergency exit door is not properly secured and locked. Activates MASTER CAUTION lights.

XL/XLS ANNUNCATOR

EMERGENCY EXIT			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
This message is displayed when the emergency exit is open.			

XLS+ CAS MESSAGE

Figure 52-8. Emergency Exit Indications

Operation

To open the emergency exit door rotate either the inside or outside handle. This disengages the lockpin. When opening the door from the

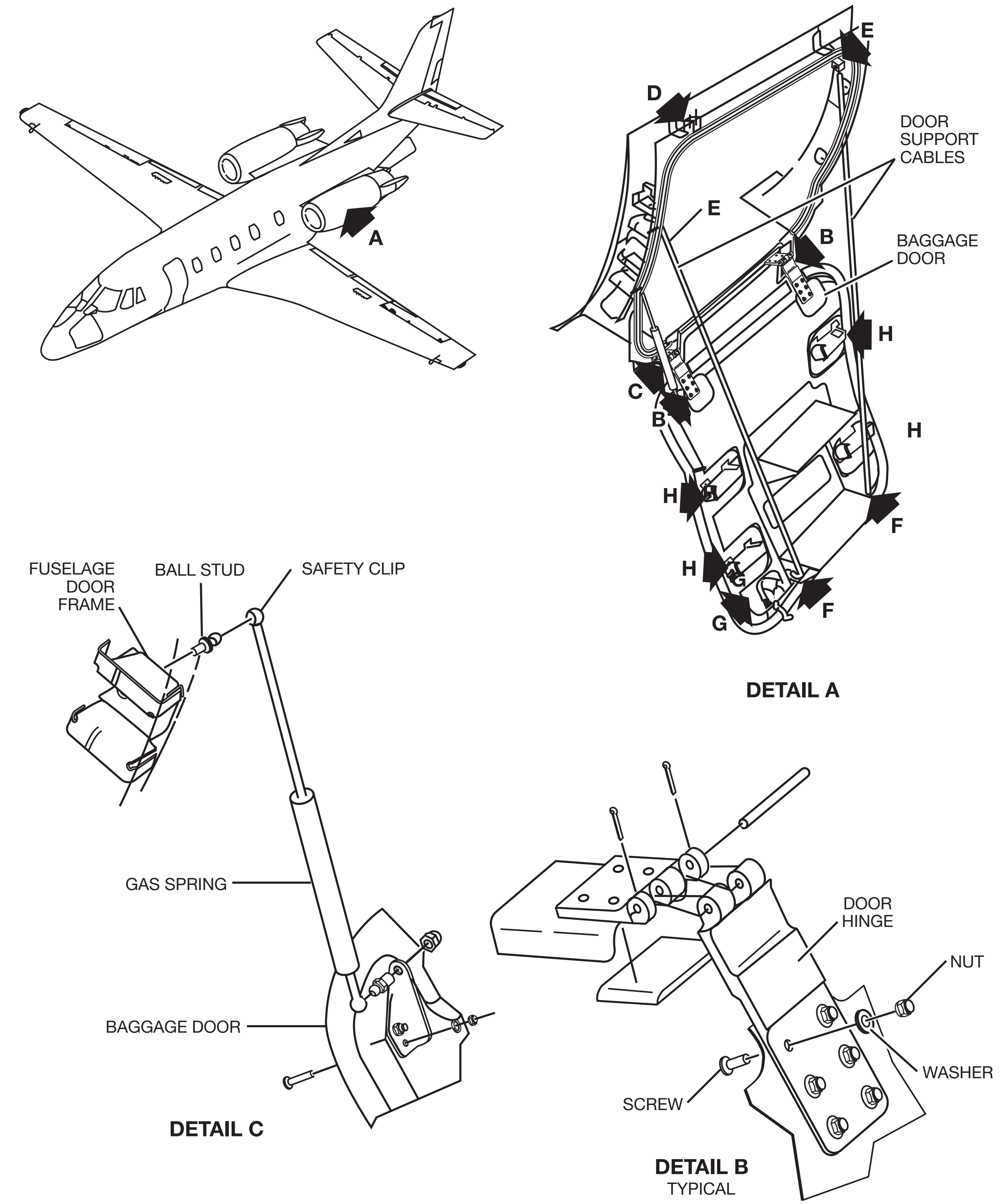


Figure 52-9. Baggage Door Installation

inside for maintenance, inspection or seal replacement, remove the ground-locking pin and the plastic guard over the inside handle. For security purposes, the door cannot be opened from the outside before the locking pin is removed. Allow the top of the door to extend inside the cabin until it clears the door frame. Raise the door to clear the retaining clips and remove the door.

Emergency Exit Door Reinstallation

The emergency exit door must be reinstalled from the inside the cabin.

1. Rotate the inside handle to the unlatched position.
2. Position the lower edge of the door in the door frame opening, behind the retainer clips.
3. When the door is behind the retainer clips, push the top of the door into the frame opening.
4. When door is properly aligned, firmly push the door shut and rotate the handle to latch the lockpin.

BAGGAGE DOOR

Description

The baggage door is on the left side of the tail cone (Figure 52-9). It is attached at the bottom by two hinges and supported in the open position by two support cable assemblies. A gas spring, attached to door and fuselage door frame (at the forward side), assists in opening the door and holds the door in the open position. The door is secured in the closed position by four latches. One switch monitors the door closed-and-latched position. The switch is electrically connected to the ACC DOOR UNLOCKED TAIL annunciator (XL/XLS) or the amber AFT BAGGAGE DOOR CAS message (XLS+) (Figure 52-10). A key lock is provided for baggage security.

NOSE/TAIL ACC DOOR UNLOCKED

NOSE
Flashes to indicate one of the nose avionics doors is not properly latched. The two bottom latches on each door are monitored (four total).

TAIL
The baggage or tailcone door is not properly latched. SNs 5188 and on or otherwise modified, the battery door is also monitored and will illuminate.

XL/XLS ANNUNCATORS

AFT BAGGAGE DOOR			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the baggage door is open.			

XLS+ CAS MESSAGE

Figure 52-10. Aft Baggage Door Indications

Controls and Indications

The tail cone baggage door and forward tail cone maintenance access door are monitored by the tail cone baggage door switch and the tail cone light door switch. The switches are double-pole single-throw. One side of each switch controls the interior lights for each compartment. The other side of each switch is wired in parallel, providing a ground to illuminate the ACC DOOR UNLOCKED TAIL annunciator (XL/XLS) or the amber TAILCONE ACC DOOR CAS message (XLS+) when the door is open (when either switch is closed) (Figure 52-11). The ACC DOOR UNLOCKED TAIL annunciator extinguishes only when both switches indicate open (when the door is closed).

TAILCONE ACC DOOR			
Color	**Inhibited By**		**Debounce**
Amber	LOPI	TOPI	Standard
This message is displayed when the tailcone access door is open.			

XLS+ CAS MESSAGE

Figure 52-11. TAILCONE ACC DOOR CAS Message

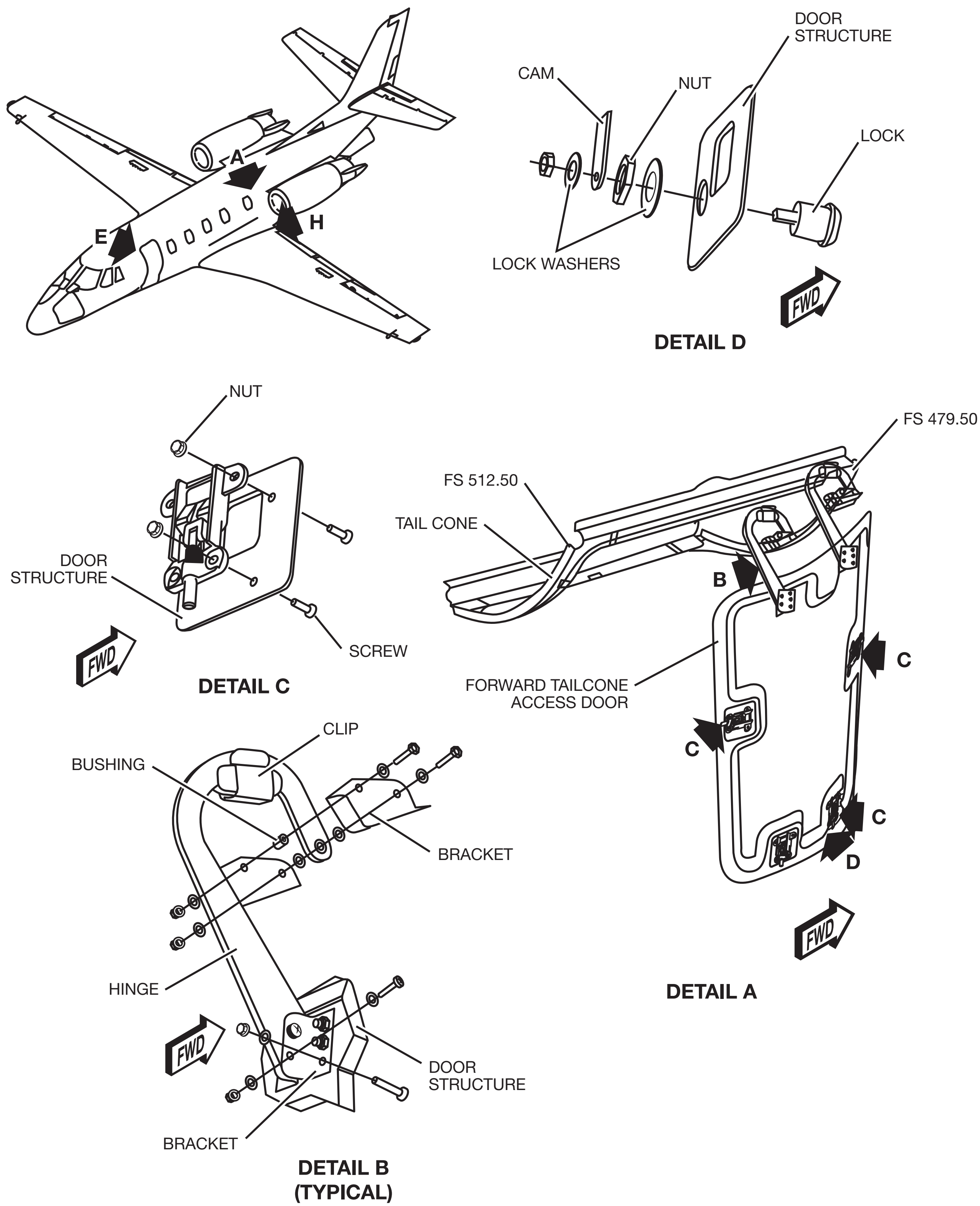


Figure 52-12. Lockable Service Door Installation (Sheet 1 of 3)

51-57 STRUCTURES

The accessory doors consist of:

- Nose compartment doors
- Baggage door
- Forward tail cone maintenance access door

The tail cone baggage door and forward tail cone maintenance access door warning system is comprised of:

- Two tail cone baggage door switches (forward ST011 and aft ST013)
- Tail cone light door switch (ST008)
- ACC DOOR UNLOCKED TAIL annunciation

Operation

Open Baggage Door

The baggage door is opened by unlocking key lock, releasing the latches and lowering the door.

Close Baggage Door

The door is closed and secured by lifting the door to the closed position, closing the latches and locking the key lock.

NOSE COMPARTMENT AND SERVICE DOORS

Service/access doors on the outside of the aircraft allow access to enclosed components for removal/installation, adjustment and maintenance purposes.

There are 11 service/access doors covered in this section (Figures 52-6 and 52-7):

- Aft tail cone access door
- Forward tail cone access door
- Hydraulic service door
- Toilet service door
- Battery access door
- Brake service door
- External electrical power receptacle door
- Single point refuel door
- Tail cone controls access door
- Nose compartment access doors

Description

The nose compartment doors are right and left of the nose section. The doors are attached at the top by two hinges. Two hook-type latches at the bottom of the doors secure them in the closed position. A gas spring (at the upper aft hinge of each door) assists door opening and holds the door in the open position. Both doors are equipped with key locks.

The service/access doors are typically attached with hinges, and are held in place by latches or mechanical fasteners. The forward tail cone access door, battery access door and the single point refueling door are equipped with key locks for added security.

The aft tail cone access door is on the lower side of the tail cone (between FS 528.50 and FS 544.50). This door provides access to the components aft of the canted bulkhead.

The forward tail cone access door is on the right side of the tail cone (between FS 479.50 and FS 512.50). This door provides access to the components aft of the baggage compartment.

The hydraulic service door is on the right of the aircraft (at FS 424.50). It provides access to the hydraulic connections for service and maintenance.

The toilet service door is on the right side of the aircraft (between FS 408.45 and FS 422.03). This door provides access to externally service the toilet.

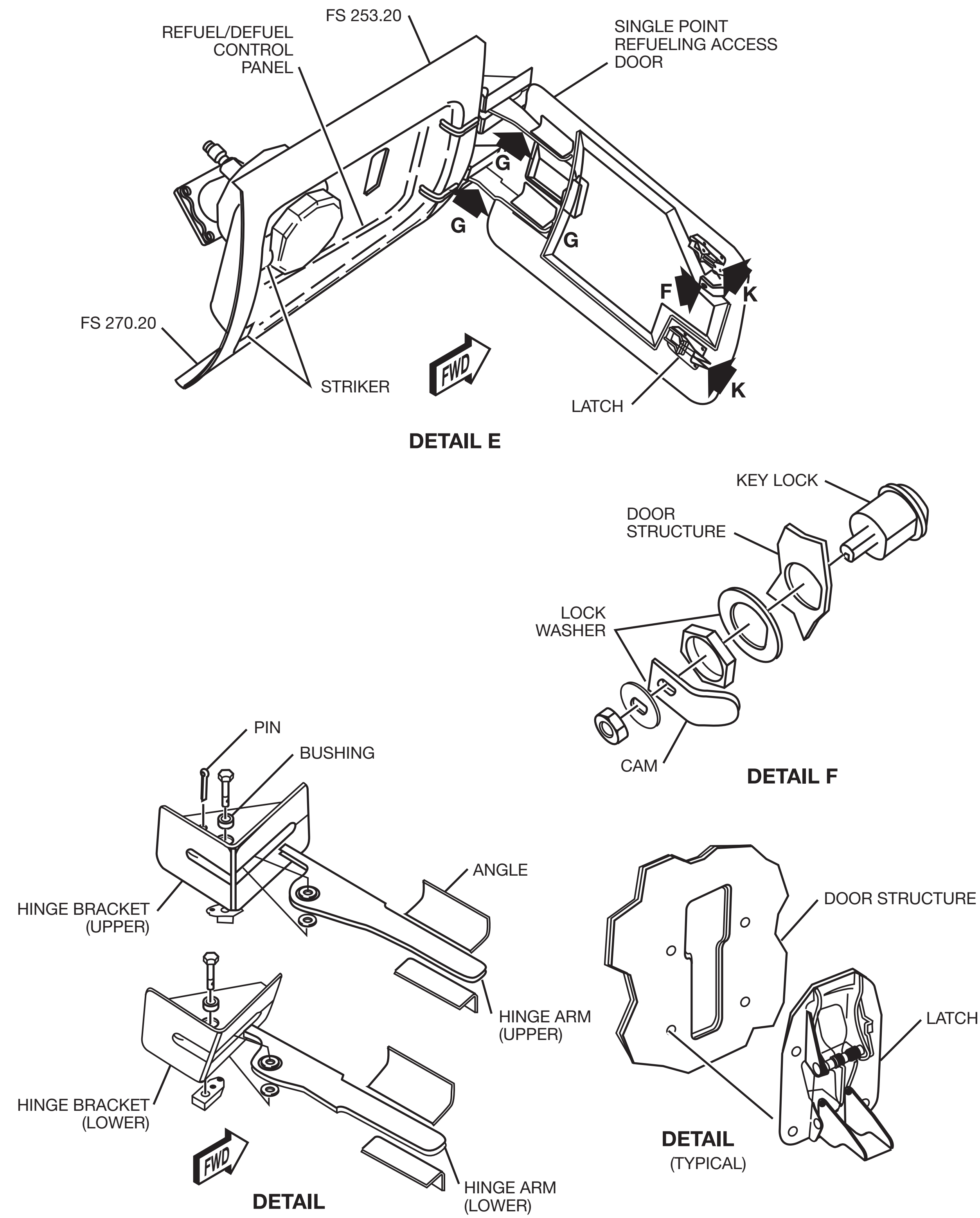


Figure 52-12. Lockable Service Door Installation (Sheet 2 of 3)

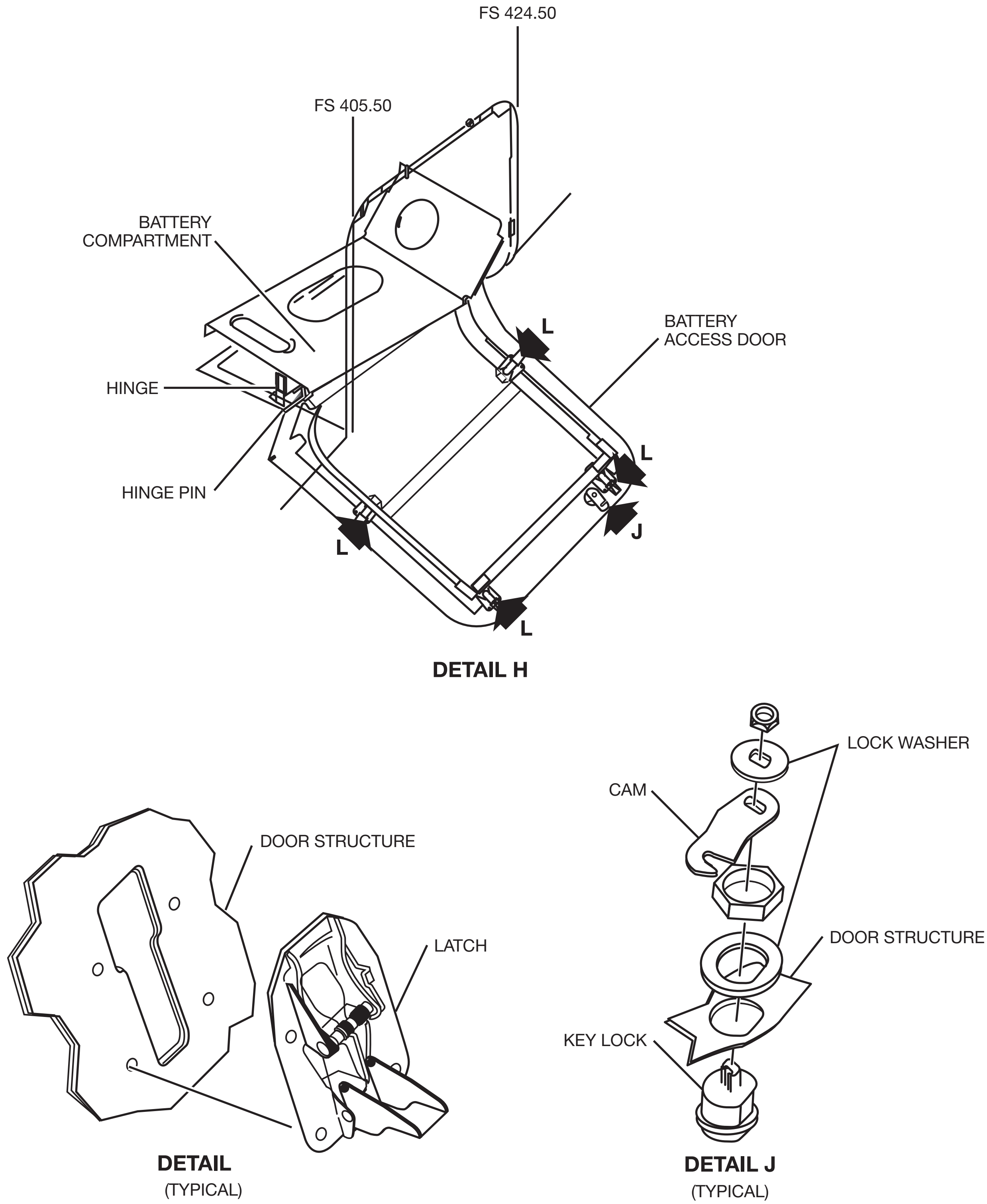
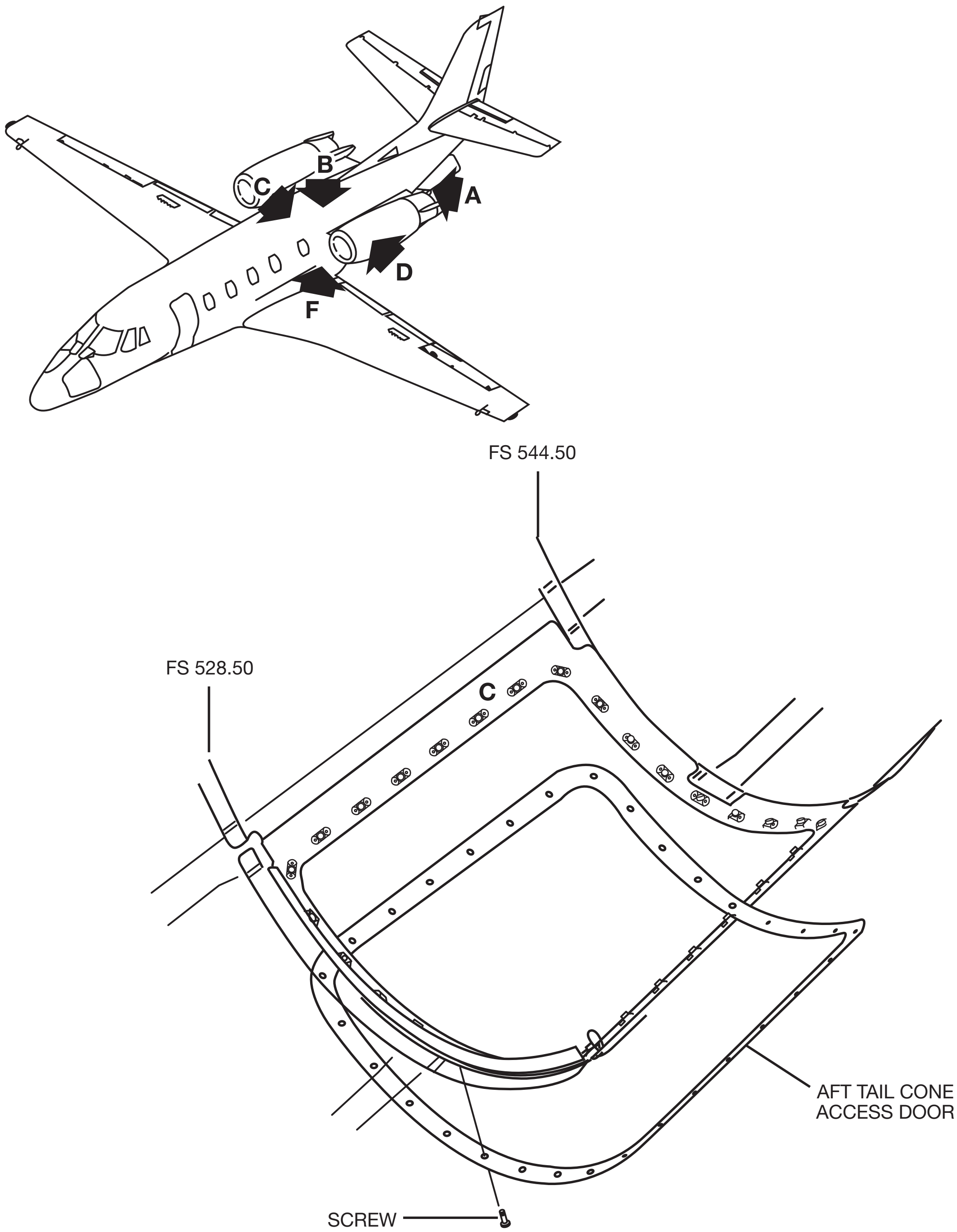


Figure 52-12. Lockable Service Door Installation (Sheet 3 of 3)



DETAIL A

Figure 52-13. Service Door Installation (Sheet 1 of 5)

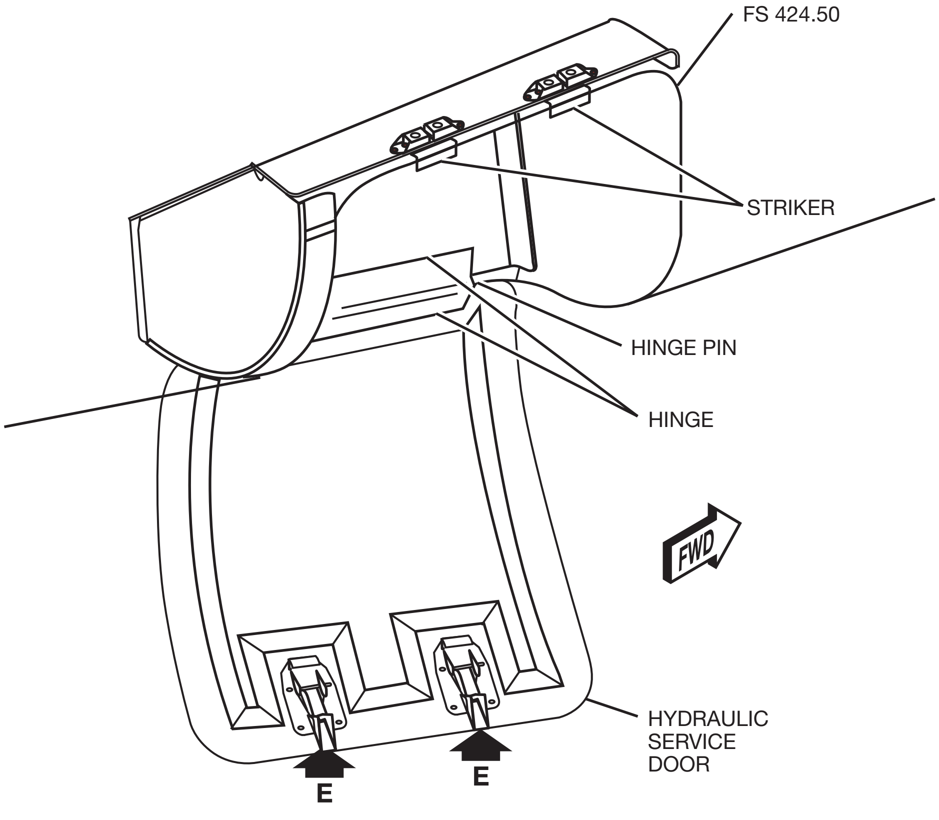
The battery access door is on the left side of the aircraft (between FS 405.50 and FS 424.50). This door provides access to the battery and battery disconnect.

The battery access door is found on aircraft SNs 5501 and subsequent. It is on the left side of the aircraft (between FS 389.50 and FS 405.50). This door provides access to the brake reservoir and accumulator service port.

The external electrical power receptacle door is on the left side of the tail cone (at FS 491.50). The door provides access to the external electrical supply connection.

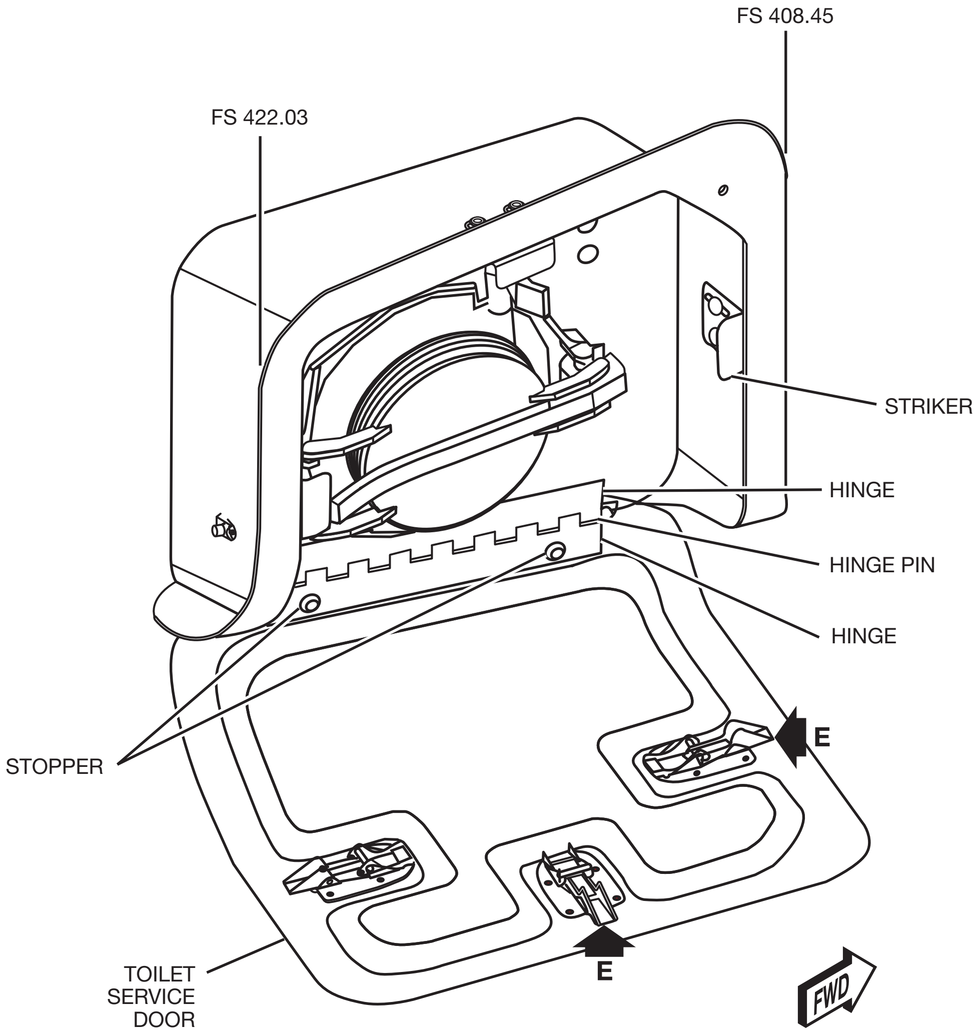
The single-point refueling door is on the right side of the aircraft (between FS 253.20 and FS 270.20). This door provides access to the refueling/defueling control panel and the single-point fueling location.

The tail cone controls access door is on the left side of the tail cone (at FS 573.50). This door provides access to tail cone controls.



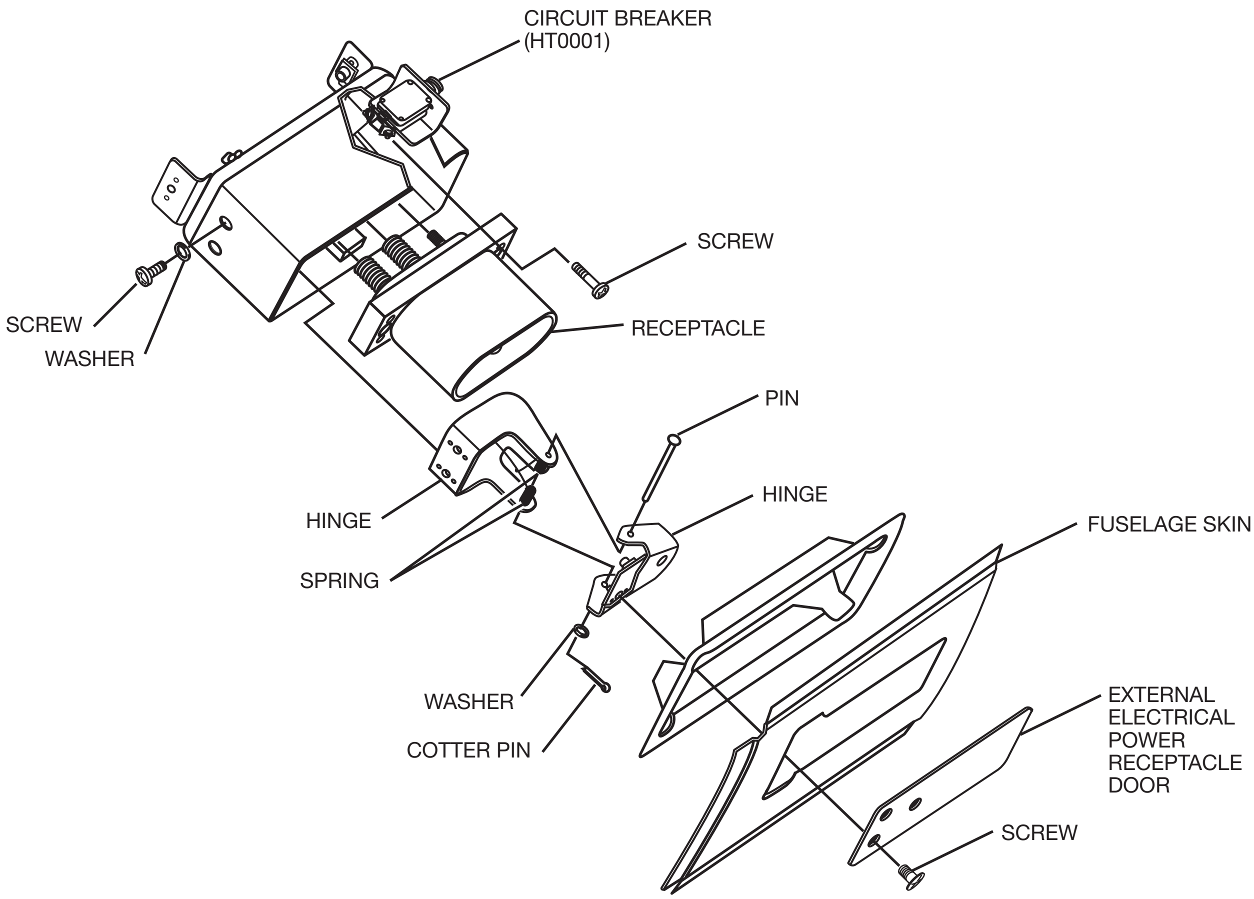
DETAIL B

Figure 52-13. Service Door Installation (Sheet 2 of 5)

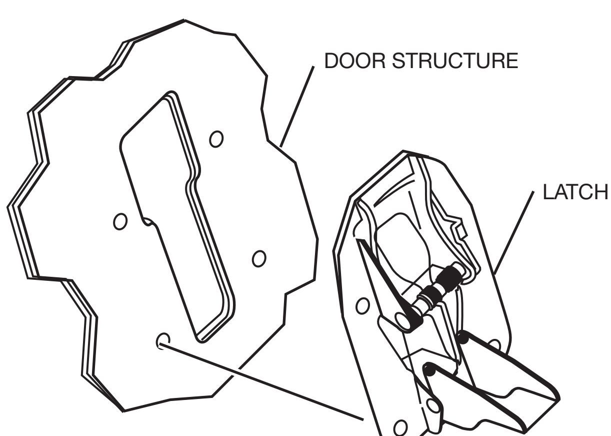


DETAIL C

Figure 52-13. Service Door Installation (Sheet 3 of 5)



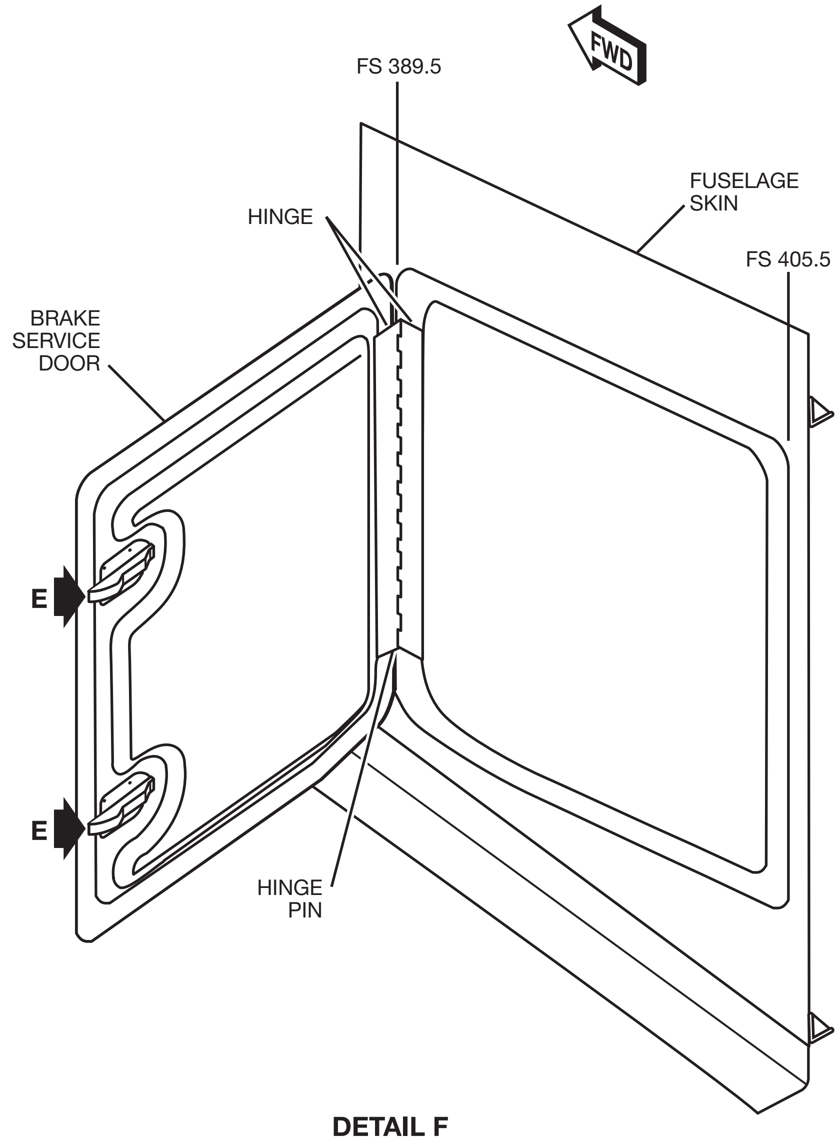
DETAIL



DETAIL

Figure 52-13. Service Door Installation (Sheet 4 of 5)

51-57 STRUCTURES



DETAIL F

(AIRPLANES 5501 AND ON)

Figure 52-13. Service Door Installation (Sheet 5 of 5)

Controls and Indications

Each nose compartment door is monitored by two switches. The left and right door warning circuits form a parallel circuit, with each set of switches are wired in parallel. This provides a ground to illuminate the ACC DOOR UNLOCKED NOSE annunciator (XL/XLS) or the amber NOSE DOOR CAS message (XLS+), when any door is open (switch is closed). The ACC DOOR UNLOCKED NOSE annunciator (XL/XLS) or the amber NOSE DOOR CAS message (XLS+) extinguishes only when all four switches indicate the door is closed (switches open) (Figure 52-14).

NOSE DOOR			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
This message is displayed when either nose door is open.			

XLS+ CAS MESSAGE

Figure 52-14. NOSE DOOR CAS Message

The nose compartment door warning system is comprised of:

- Four switches:
  - SN003 and SN023 (left)
  - SN008 and SN018 (right)
- An ACC DOOR UNLOCKED NOSE annunciation

Battery Access Door

The battery access door is located on the left side of the aircraft between FS 405.50 and FS 424.50. This door provides access to the battery and battery disconnect. The battery access door is monitored by a proximity switch theat provides a ground to illuminate the ACC DOOR UNLOCK annunciator (XL/XLS) or the amber BATTERY DOOR CAS message (XLS+) (Figure 52-15).

BATTERY DOOR			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
This message is displayed when the battery door is open.			

XLS+ CAS MESSAGE

Figure 52-15. BATTERY DOOR CAS Message

AUXILIARY POWER UNIT DOOR

The auxiliary power unit (APU) door is on the right side of the fuselage (between FS 467.00 and FS 504.00). The door installs from the outside of the fuselage and incorporates:

- APU inlet
- APU service door assembly
- A removable APU starter/generator cooling inlet

The APU door components referenced in this section are the APU service door assembly and the APU starter/generator cooling inlet.

The APU service door is in the right-center portion of the APU door. It consists of a hinge on the top and two latches on the bottom edge of the door.

The APU starter/generator cooling inlet is on the upper right corner of APU door. The APU starter/generator cooling inlet section of the APU door is removable to allow access to the APU without removing the starter/generator inlet.

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51-57 STRUCTURES

CHAPTER 53
FUSELAGE

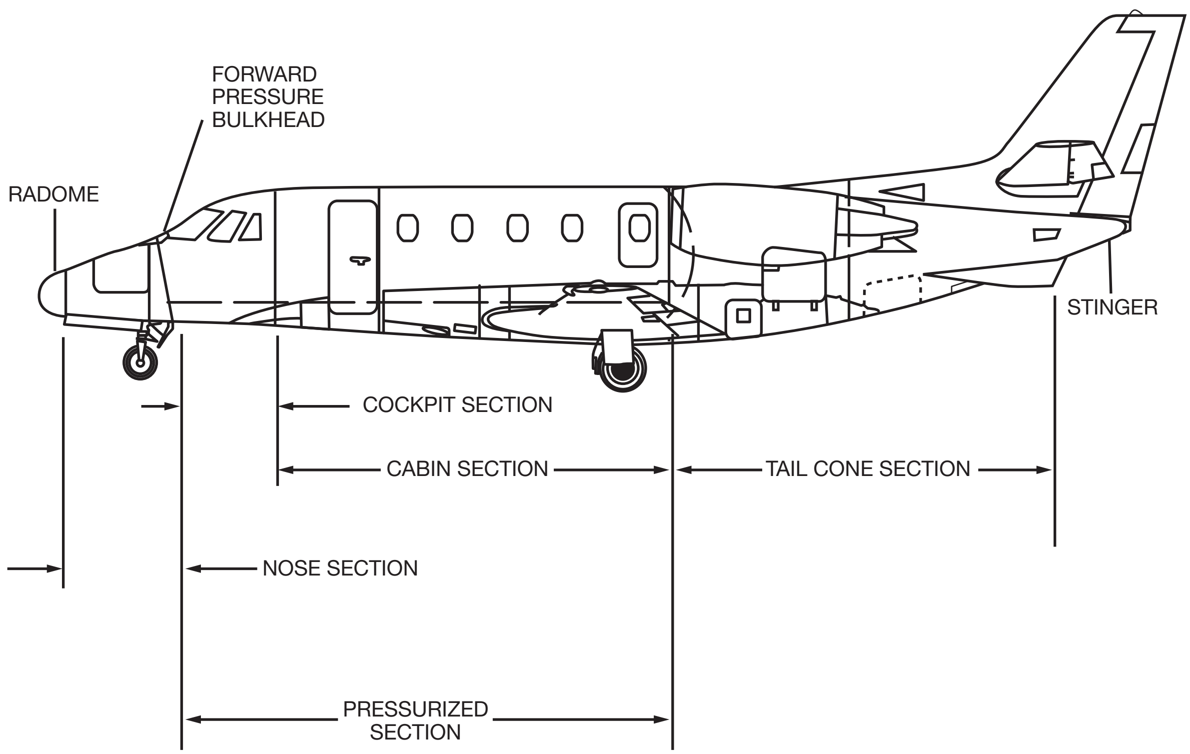


Figure 53-1. Fuselage Section

DESCRIPTION

This chapter describes the structural portion of the fuselage, which makes up the compartments for equipment, flight crew, passengers and cargo/baggage. Included are the main frame, auxiliary structure, and aerodynamic fairings (Figure 53-1).

Main Frame

Description

This section describes the fuselage main frame. The following specifics are covered for maintenance:

- Airframe moisture drainage
- Nose radome
- Nose wheel well door seals
- Dorsal fin
- Tail cone stinger

The following information describes the primary envelope of the fuselage, including:

- Construction of nose section
- Cabin section
- Tail cone/aft fuselage section

Also described in this section are major structures, including:

- Nose landing gear wheel well
- Forward pressure bulkhead
- Mid pressure bulkhead
- Aft pressure bulkhead
- Engine carry-thru beams

For repair/maintenance of fuselage structures, refer to the *Structural Repair Manual*.

Components

The fuselage main frame utilizes bonded skin assemblies and stringers for aerodynamic smoothness and strength. The unpressurized nose section provides space for avionics equipment and incorporates nose landing gear wheel well structure. The pressurized cockpit/cabin section (pressure vessel) is circular in shape and includes both flight crew and passenger compartments. The tail cone section includes a baggage compartment/equipment compartment in the aft tail cone. The baggage compartment contains electrical/ avionics equipment shelves overhead (behind forward and aft niche panels). The equipment compartment contains various equipment and components, including environmental control units (ECUs).

Nose Section

The nose section of fuselage extends from FS 23.20 (at the leading edge of the nose radome) to FS 94.00 (at the forward canted pressure bulkhead).

Left and right nose compartments are left and right of nose landing gear wheel well. They incorporate various avionics equipment supports. Compartments extend from FS 39.60 to forward pressure bulkhead (FS 94.00), on top of and on each side of nose wheel well.

Nose landing gear wheel well is centerline of nose section between FS 39.60 and forward pressure bulkhead (FS 94.00). It is constructed of:

- Angles
- Frames
- Stiffeners
- Doubler
- Hinge supports
- Webs
- Channels

Nose landing gear side doors enclose the lower portion of the wheel well when the nose gear is retracted.

Cockpit/Cabin Section

Cockpit/cabin section of fuselage is a pressurized area of the fuselage (extending from FS 94.00 at forward pressure bulkhead to FS 373.75 at pressure bulkhead). The cockpit/cabin structure consists of bonded assemblies:

- Frames
- Angles
- Doublers
- Stringers
- Splice straps with bonded skin

Sealer is utilized with permanent fasteners and splice joints to form an airtight cockpit/cabin section.

Forward Pressure Bulkhead

The forward pressure bulkhead is canted (from FS 94.00 at top to FS 102.01 at bottom) is a bonded construction of:

- Vertical posts
- Stiffeners
- Webs
- Doublers
- Frames
- Spacers

All parts (except some stiffeners) are bonded using film adhesive and curing-type corrosion-inhibiting adhesive primer. Some stiffeners are room-temperature bonded with permanent fasteners for increased fatigue resistance. Provisions are made for mounting components, assemblies and feed-throughs for:

- Avionics/electrical systems
- Plumbing
- Mechanical control cables

Pressure sealant is utilized where required to maintain a sound pressure barrier. The forward pressure bulkhead not only forms forward cabin pressure barrier, but also accepts loads imposed by nose landing gear.

Aft Pressure Bulkhead

Various electrical, avionics and bleed-air feed-throughs are on upper portion of pressure bulkhead.

Aft Fuselage Section (Tail Cone)

The tail cone extends aft from FS 373.75 (aft of the pressure bulkhead).

Engine Carry-Thru Beams

Forward and aft engine carry-thru beams extend through the tail cone to support pylons and engines. Forward carry-thru (at FS 421.50 to FS 424.50) consists of:

- Angles
- Webs
- Doublers
- Upper and lower caps
- Web stiffeners
- Upper and lower straps

There are fittings on each end of the beam for mating with engine mounts. Aft carry-thru (at FS 460.50 to FS 462.50) consists of the same type construction used for forward carry-thru. Carry-throughs are bonded assemblies utilizing adhesive primer at upper and lower caps. Webs incorporate lightening holes to reduce weight. Provisions are made to attach system components to carry-throughs with mounts and brackets.

Fuselage Opening Frames

Fuselage openings include:

- Frames around nose compartment doors
- Main entrance door
- Emergency exit door
- Tail cone baggage door
- Tail cone maintenance access door

For maintenance of doors, refer to Chapter 52—"Doors."

The nose compartment frame opening consists of a drip cap around a periphery of door openings. The door seal is bonded to a drip cap, which is bonded to nose frames, stringers and skin. A longeron is slotted at lower center for mating with the the door-lock cam.

The frame opening of the main entrance door is on the left side of fuselage (between FS 189.00 and FS 215.00). The frame is a bonded construction using Type I film adhesive and permanent fasteners. The frame consists of:

- An inner frame
- Outer frame
- Lower frame
- Gussets
- Intercostals
- Clips
- Seal depressors
- Doublers
- Lintel angle
- Threshold angles

Provisions are made for door support struts, door counterbalance cables, and eight guide blocks (latching door pin sockets), on the forward, aft, and upper portions of the frame. Pressure sealant is utilized where a sound pressure barrier is required.

The frame opening for the emergency exit door is on right side of fuselage (between FS 339.26 and FS 363.26). The frame consists of:

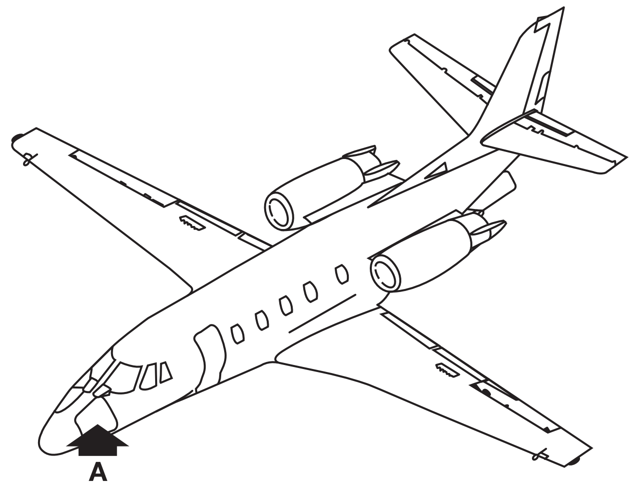
- Forward frame
- Aft frame
- Outer frame
- Intercostals
- Stiffener
- Door stop
- Clips
- Inner cap doubler

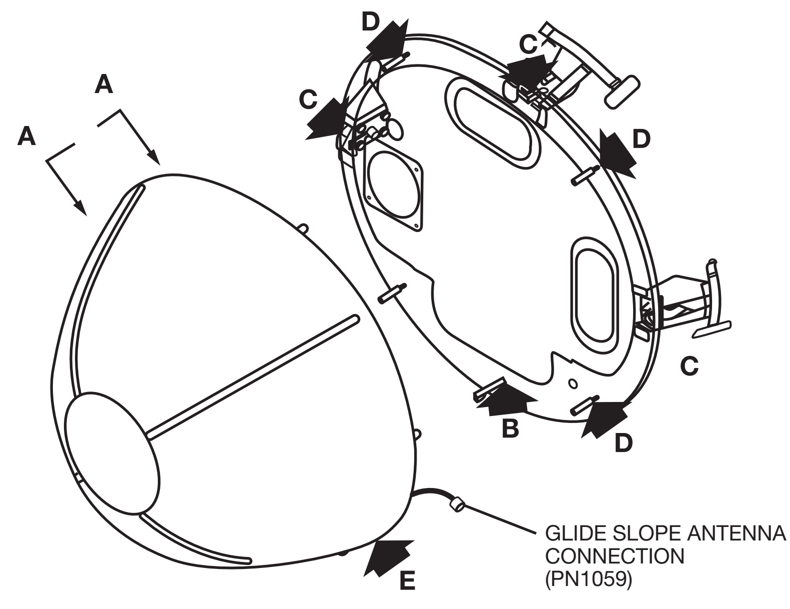
The inner cap doubler encompasses the door frame inside the fuselage structure. It incorporates lightening holes for weight reduction. The emergency exit opening frame assembly and inner cap doubler are secured to the fuselage structure and skin with bonding and permanent fasteners. Pressure sealant is utilized where a sound pressure barrier is required.

The frame opening of the tail cone baggage compartment door is on left side of aft fuselage (tail cone) (between FS 424.50 and 460.50). The frame consists of:

- Outer frames
- Inner frames
- Angles
- Doublers
- Four guide blocks (latching door pin sockets)
- Longeron
- Lintel

Provisions are made for door hinge installation. The frame is secured to the tail cone assembly and structure with bonding and permanent fasteners. Pressure sealant is utilized where a sound pressure barrier is required.





DETAIL A

Figure 53-2. Nose Radome Installation

Auxiliary Structure

This section describes the secondary structures of the fuselage, including the:

- Nose (avionics) compartment
- Flight crew floor panels
- Tail cone baggage compartment

The tail cone/aft fuselage section houses the aircraft systems, tail cone baggage compartment, and avionics/electrical equipment.

The nose section is constructed of frames, angles, doublers, gussets, and stiffeners with bonded skin assemblies and stringers. It houses and supports avionic components, nose landing gear and wheel well, oxygen and other aircraft systems equipment.

Nose compartments are accessed through hinged doors on the left and right side (at approximately FS 58.30 to FS 90.20). The right door is the larger door.

Cabin floor panels are flat behind the passenger compartment. The stand-up height created by center aisle is 70.0 inches (1.78 m).

The tail cone maintenance access door is on the right side of tail cone (between FS 479.50 and 512.50).

Nose Radome

The radome assembly is on the nose of aircraft. It is shaped for optimum anti-ice characteristics (Figure 53-2).

Description

The nose radome is a bonded assembly of prepreg epoxy glass (E-glass) with:

- Adhesive film autoclave bonding
- Corrosion-inhibiting adhesive primer
- Room temperature bonding

An anti-P static-conductive coating is applied to the surface area of the radome, before the topcoat finish is applied. Lightning diverter strips and a weather erosion boot are attached after the topcoat finish is complete.

Guide pin assemblies align the radome with the aircraft nose frame during installation. Rotary latches that mate with adjustable eyebolts attach the radome to the aircraft. Guide pins, rotary latches, and adjustable eyebolts provide an electrical bond path between the radome and the aircraft structure. A weather seal attached to the radome structure seals the radome to the nose frame mating surface.

Plates/Skin

Description

This section describes the exterior covering of the fuselage, including the bonded skin assemblies and access plates.

For repair of the bonded skin and structure assemblies, refer to the *Structural Repair Manual*. For corrosion treatment and control, refer to Chapter 51—"Corrosion."

Components

The exterior covering of the fuselage is engineered utilizing bonded skin assemblies, doublers, and permanent fasteners secured to stringers, frames and bulkheads.

The cockpit upper-bond assembly consists of doublers and skin. Waffle doublers mate with stringers across the top (crown) of the cockpit and just below the left and right windshield (extending from the forward pressure bulkhead, aft under the side windows).

There are other doublers on the windshield center bow, window sills, and posts.

Spacers are bonded below the lower sills. The skin is formed over the doublers. All parts are bonded using Type I, Class II film adhesive and curing-type corrosion-inhibiting adhesive primer, except where room temperature bonding is used on the spacers. The bonded assembly is reinforced with permanent fasteners.

Skin doublers are utilized around the periphery of the cabin with cutouts at the windows. There are other skin doublers around the door frames and main wheel well areas. Encompassing the cabin are left, right, lower aft and lower right skin panels with cutouts at the window, doors, and main wheel well areas. The skin panel assemblies are bonded at skin splices using Type I film adhesive. The adhesive is cured by applying external heat. Sealer is used with permanent fasteners during installation, to enhance the effectiveness of the pressure vessel (cabin).

The aft fuselage (tail cone) bonded skin assemblies are bonded at skin splices with doublers. They are reinforced with permanent fasteners. Bonding is accomplished with Type I, Class II primer and room temperature curing adhesive.

Access Plates

There are access plates under the fuselage and tail cone for access to plumbing and components. The access plates are secured with screws. Those at a fuel reservoir incorporate a seal. For identification and location of all access plates and panels, refer to Chapter 6—"Access Plates and Panels Identification."

NOTE

Access plates at fuel reservoirs incorporate a seal. DO NOT USE ADDITIONAL SEALANT ON AN ACCESS PLATE. If an access plate does not seal properly, replace the access plate. The use of a sealant can insulate the plate from the airframe and electrical conductivity is lost. Electrical conductivity is required for the protection of lightning strikes.

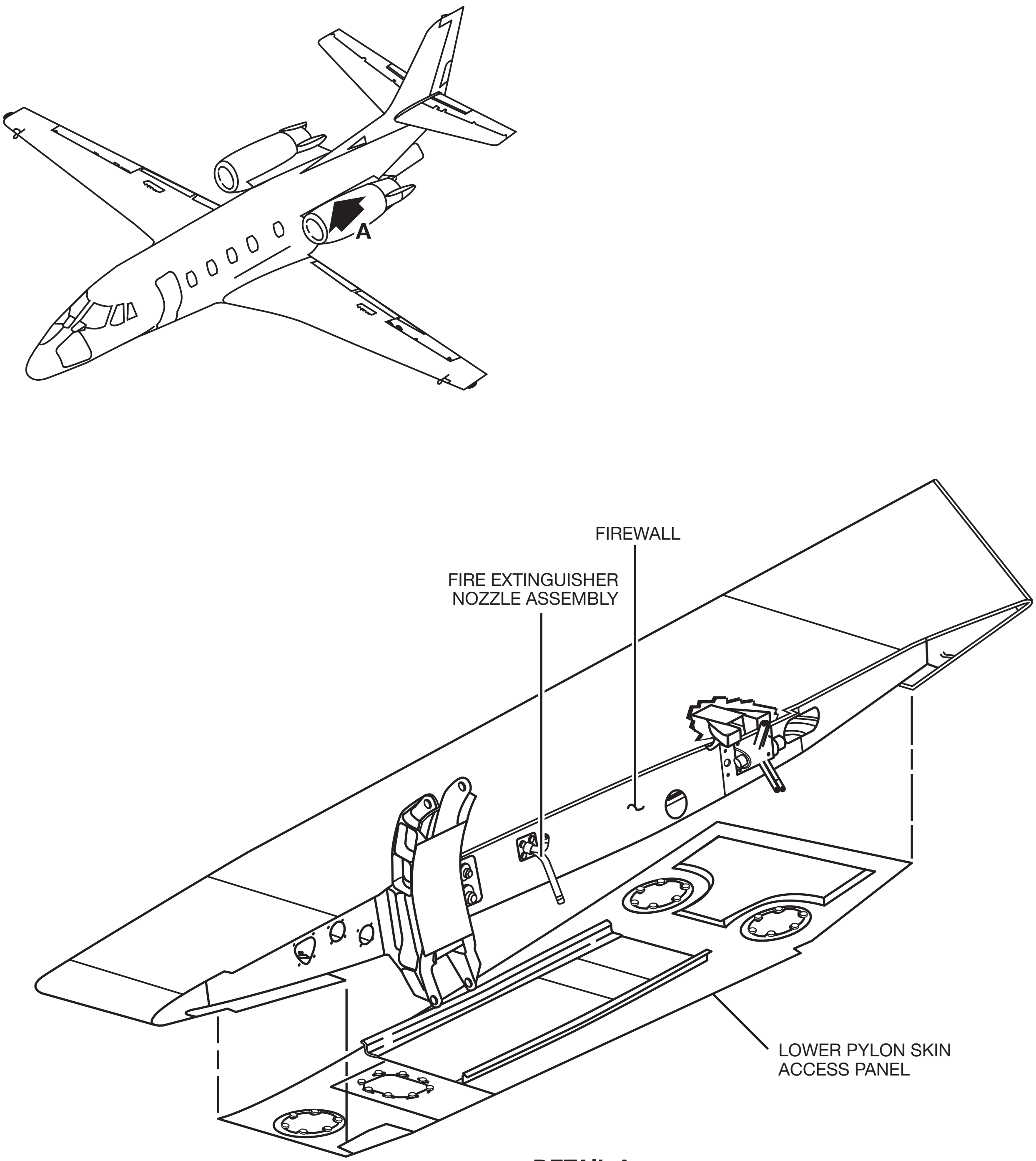
Aerodynamic Fairings

This section describes the structure of fixed and variable aerodynamic fairings, including the wing-to-fuselage fairings.

NOTES

CHAPTER 54

NACELLES/PYLONS



DETAIL A

Figure 54-1. Engine Pylon

PYLON

Description

There is a pylon is on each side of the aft fuselage tail cone (Figure 54-1). A forward and aft engine carry-thru spar travels through the fuselage, providing the weight-carrying structure for the pylons. There are engine-mounting attach fittings at each end of the carry-thru spars. The pylon houses the carry-throughs between the engine and fuselage.

The outboard ribs, extending the full length of the pylon, are constructed of stainless steel to form the firewall. A vapor barrier is formed by firewall sealant applied to the faying surfaces.

NOTES

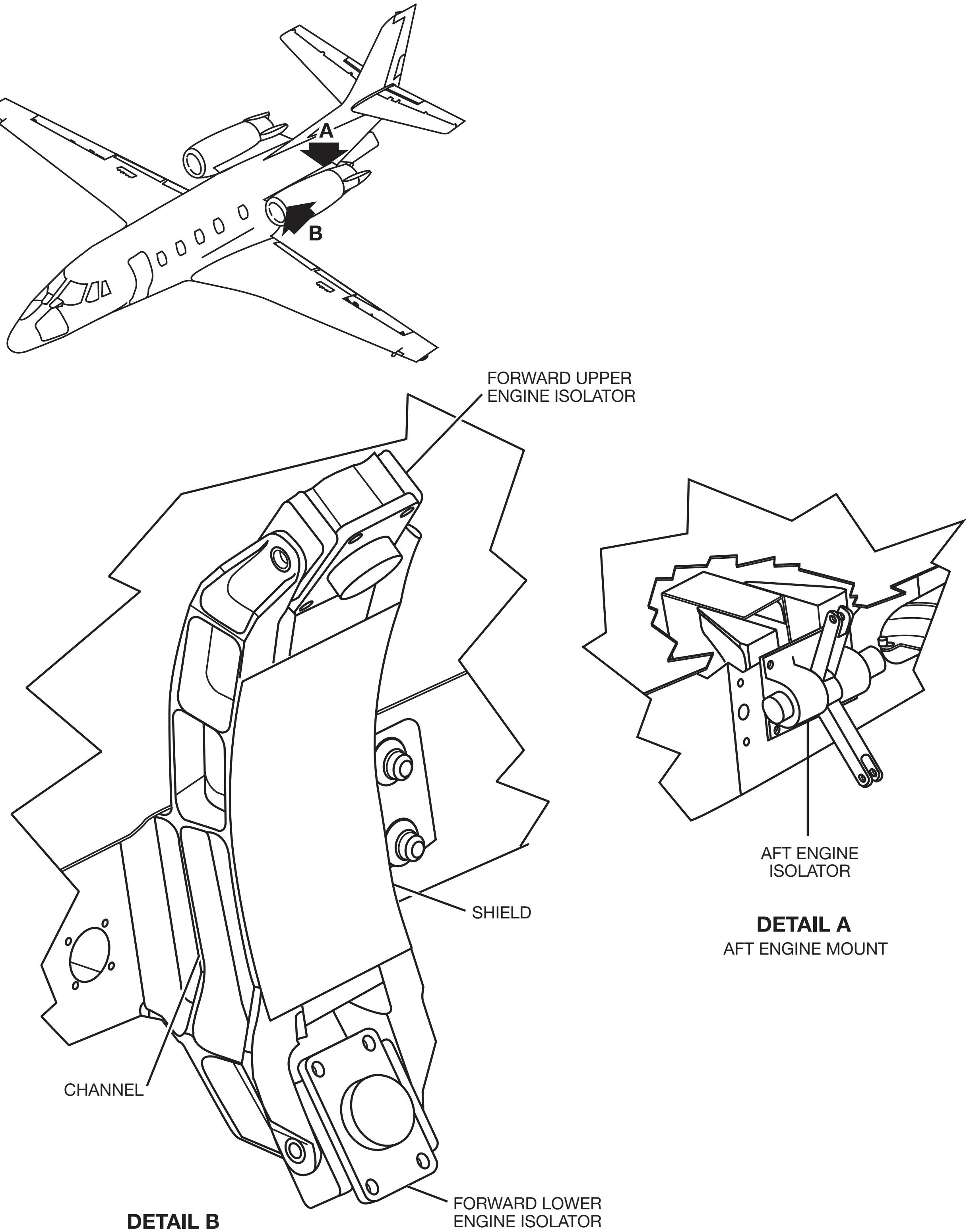


Figure 54-2. Engine Attach Fittings

Components

Engine Attach Fittings

The engine attach fittings consist of (Figure 54-2):

- The forward engine mount
- The aft engine mount
- Plumbing lines and fittings
- Electrical connectors
- Engine controls

For a description of the plumbing lines and fittings, electrical connectors and engine controls, refer to “Pylons” in this chapter, to Chapter 71—“Powerplant,” and to Chapter 76—“Engine Controls.”

Description

Refer to Figure 54-2 for Engine mounts.

Two forward engine isolators for each engine are assembled to the forward engine attach points and mate with the forward engine mount channels. The channels are part of the forward engine carry-thru assembly.

NOTE

The forward isolators are marked with an arrow to depict “forward.”

One aft engine isolator for each engine attaches to the aft engine mount fittings and mates with the aft engine attach points. The attach fittings are part of the aft engine carry-thru assembly.

NOTES

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CHAPTER 55

STABILIZERS

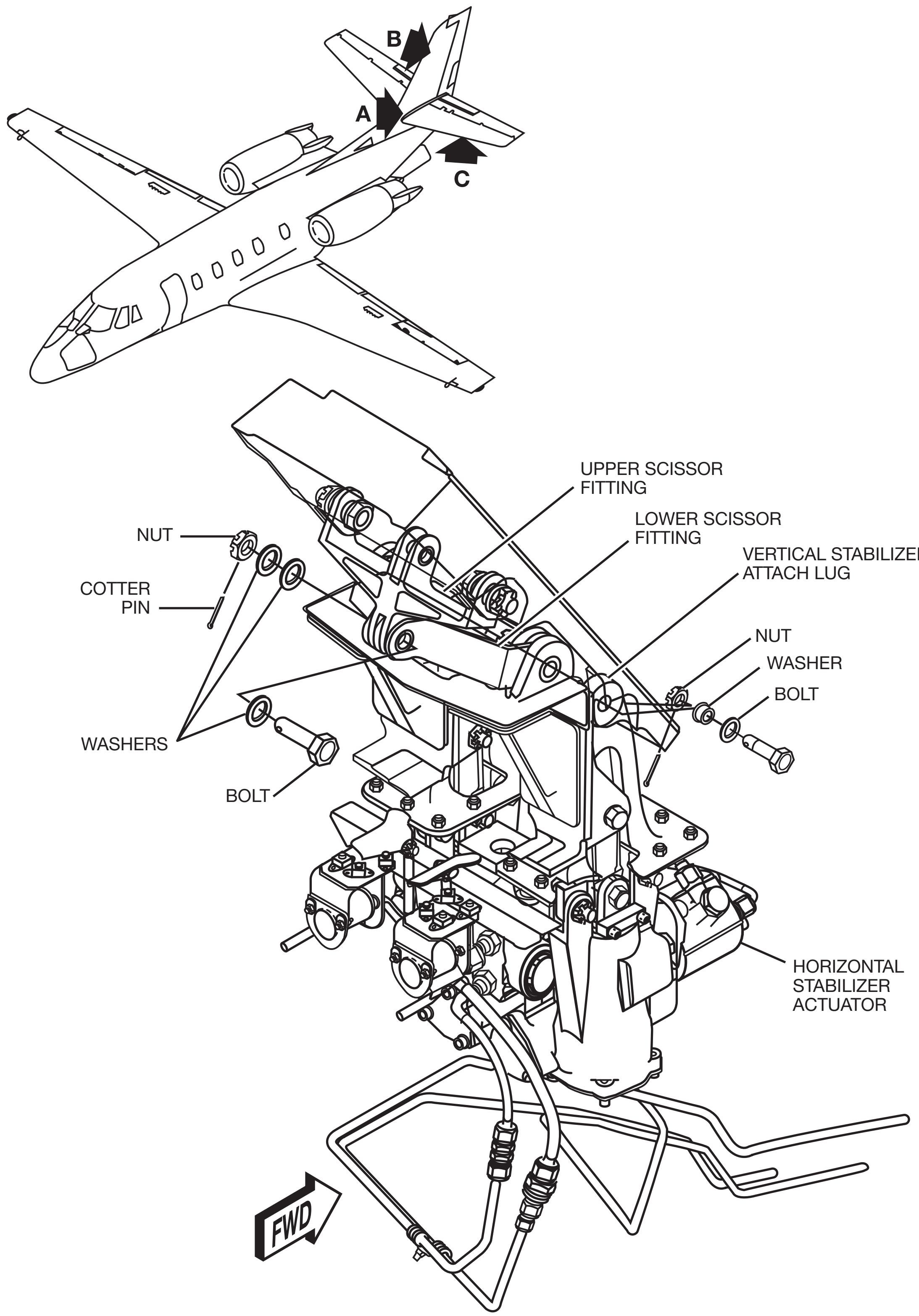


Figure 55-1. Horizontal Stabilizer Installation

TWO POSITION HORIZONTAL STABILIZER

Description

The two-position horizontal stabilizer is on the vertical stabilizer above the tail cone. The stabilizer is assembled/attached as a single assembly. The stabilizer is constructed of:

- Spars
- Ribs
- Stringers
- Doublers
- Skin

The spars extend the full length of the stabilizer outboard, with splice plates joining the two halves.

Components

The two-position horizontal stabilizer is connected to the vertical stabilizer at two attach points.

The forward attach point is the scissor fitting assembly (Figure 55-1). The upper scissor fitting is attached to the vertical stabilizer forward spar. The lower scissor fitting is attached to the horizontal stabilizer forward spar. The upper and lower scissor fittings are connected to allow up and down movement of the horizontal-stabilizer leading edge.

The aft attach point consists of pivot fittings on the horizontal stabilizer aft spar and vertical stabilizer aft spar.

There is an actuator attach fitting on the horizontal stabilizer forward spar, connected to the actuator through actuator links. For maintenance information refer to Chapter 27—"Horizontal Stabilizer Actuator" and "Two Position Horizontal Stabilizer System."

Access panels on the horizontal stabilizer provide access to control surface components, electrical and avionics components.

The elevator is a balanced airfoil on the trailing edge of the horizontal stabilizer. For maintenance information refer to Chapter 27—"Elevator" and "Elevator and Tab System."

There is an adjustable balance weight in the outboard forward end of each elevator tip. The weight is adjusted by adding lead weights to holes provided, or by drilling out additional depth in holes in the elevator tip rib. If an elevator has been repainted or repaired it must be rebalanced. Refer to the *Structural Repair Manual* for balancing procedures.

Static discharge assemblies connect to the outboard trailing edge of the elevators and stabilizers. For maintenance information refer to Chapter 23—"Static Discharge Wicks."

Brackets are provided to attach the elevator trim tab actuators. The design permits installation of the control cables and chain assemblies to operate the elevator trim tab actuators.

The elevator trim tabs are attached to the trailing edge of the elevator by a piano-type hinge. Trim tab actuator pushrods attach to the trim tab horns. If the trim tab has been repainted or repaired, the elevators must be rebalanced. Refer to the aircraft's *Structural Repair Manual* for balancing procedures. For maintenance information, refer to Chapter 27—"Elevator Trim Tab and Actuator" and "Elevator and Tab System."

There are flux detector transmitters in the right and left horizontal stabilizers. The transmitter supplies a slave signal to the compass system. For maintenance information refer to Chapter 34—"LCR-93 AHRS."

There is a fiberglass tip assembly on the outboard leading edge of the stabilizer.

The logo lights are in the horizontal stabilizer.

VERTICAL STABILIZER

The vertical stabilizer is on the tail cone. The stabilizer is constructed of:

- Spars
- Ribs
- Stiffeners
- Skin panels

The spars extend the full length of the stabilizer and into the tail cone for attachment.

Description

The vertical stabilizer connects to the tail cone at two attach points. The forward spar extends into the tail cone, attaching to the aft side of the forward-canted bulkhead. The aft spar is on aft side of the aft-canted bulkhead.

Access panels on the vertical stabilizer provide access to control surface components, electrical and avionics components.

CAUTION

All vertical fin access panels must be in place during engine run up to prevent damage to internal components.

The rudder is a balanced airfoil on the trailing edge of the vertical stabilizer. For maintenance information refer to Chapter 27—"Rudder and Tab System" and "Rudder."

The balance weight is in the upper-forward leading edge of the rudder. There is an access panel is in the leading edge of the rudder for access to the balance weight. If a rudder has been repainted or repaired it must be rebalanced. Refer to the *Structural Repair Manual* for balancing procedures.

Static discharge assemblies are at the top and trailing edge of the rudder. For maintenance information refer to Chapter 23—"Static Discharging."

There is a saddle assembly at the lower leading edge of the stabilizer (between the dorsal fin and the vertical stabilizer). This provides a smooth contour between the dorsal fin and vertical stabilizer. The saddle assembly provides a place to mount the locator beacon antenna.

A bracket is provided to attach the rudder trim tab actuator. Also, the design permits the installation of the control cables and chain assemblies to operate the rudder trim tab actuators.

The rudder trim tab connects to the trailing edge of the rudder by a piano-type hinge. Trim tab actuator pushrods attach to the trim tab horn. If the trim tab has been repainted or repaired, the rudder must be rebalanced. Refer to the *Structural Repair Manual* for balancing procedures. For maintenance information refer to Chapter 27—"Rudder and Tab System" and "Rudder Trim Tab and Actuator."

There are two VHF navigation antennas , one on each side of the vertical stabilizer near the top. For maintenance information refer to Chapter 34—"Honeywell Primus II VOR Navigation Receiver."

The emergency locator transmitter, battery pack and mounting hardware is in the saddle assembly. For maintenance information refer to Chapter 25—"Artex ELT 110-4 Emergency Locator Transmitter System."

There is a removable tip assembly at the top of the vertical stabilizer.

CHAPTER 56

WINDOWS

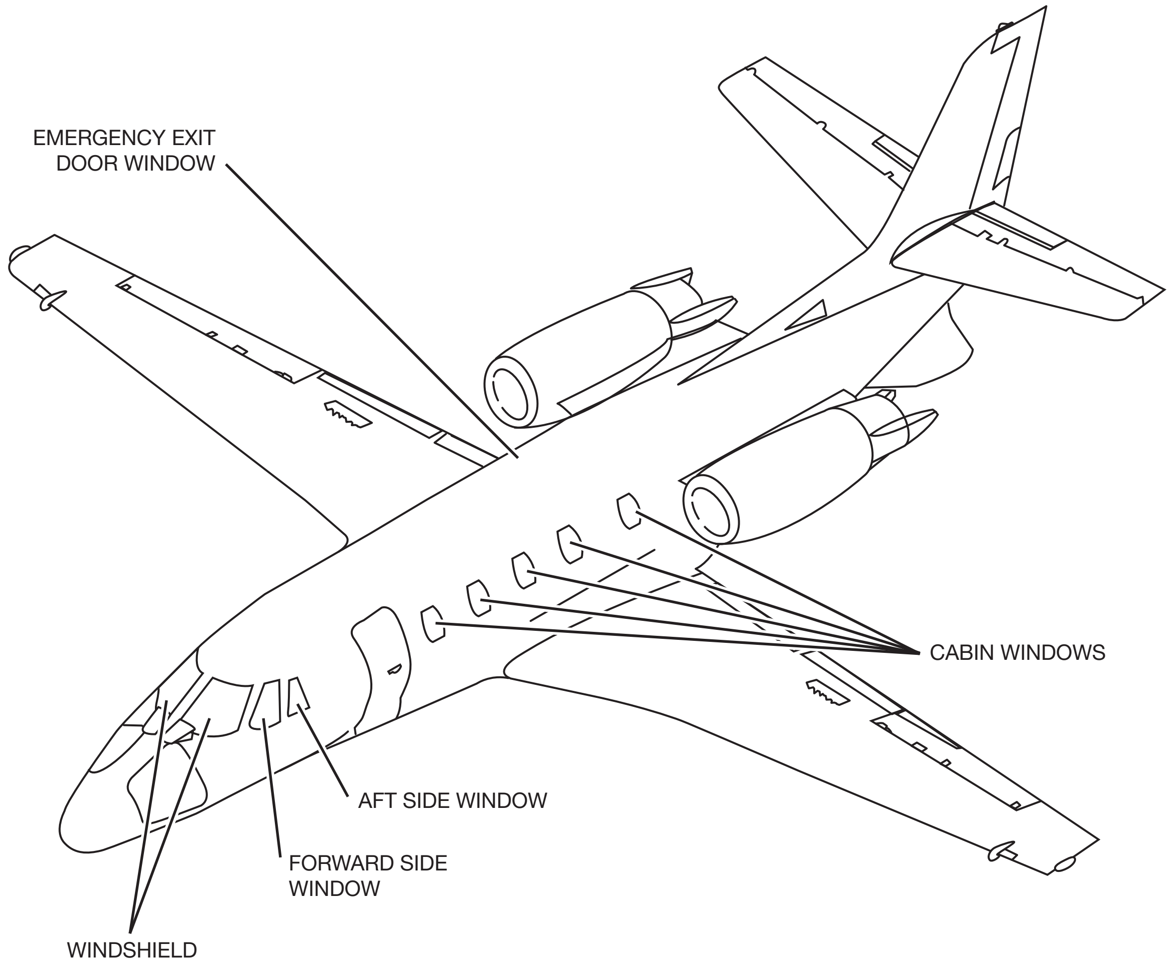


Figure 56-1. Windows

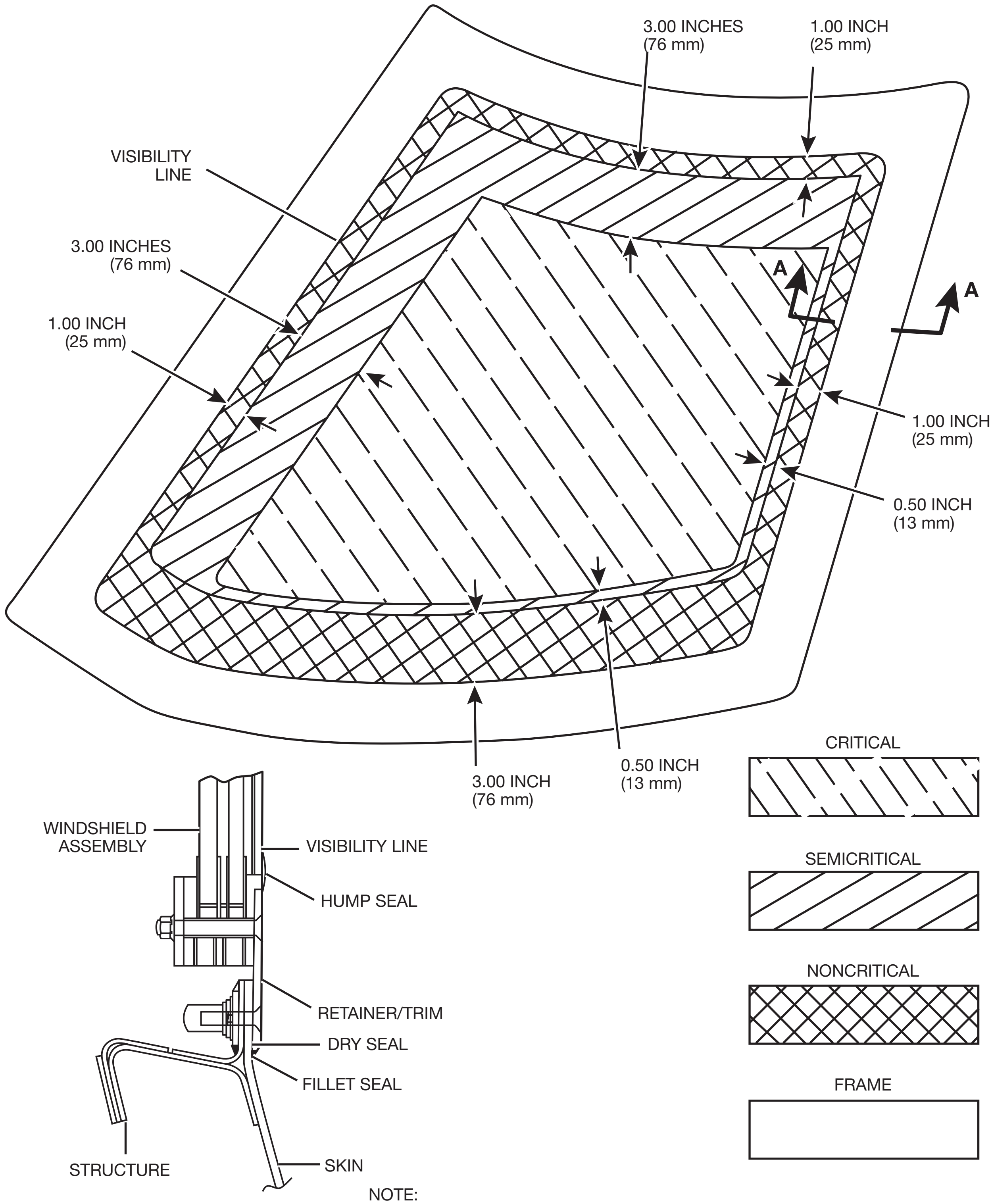
DESCRIPTION

This section includes the electrically heated glass windshield and forward side windows. Also included are the flight compartment aft openable side windows (Figure 56-1).

The windshields and flight compartment forward side windows are of laminated glass construction, with a laminated film heating element and bonded fiberglass edge attachments.

The aft openable side windows consist of laminated stretched acrylic construction. A aluminum frame is fastened around the periphery of the transparency.

NOTES



NOTE:
THE VISIBILITY LINE IS CREATED BY THE CIRCUMFERENTIAL EDGE OF THE HUMP SEAL CONTACTING THE OUTER GLASS SURFACE.

Figure 56-2. Glass Windshield View Area

51-57 STRUCTURES

COMPONENTS

Windshields

The windshield is laminated with an outer, middle, and inner pane of glass, with a layer of polyvinyl butyral between panes (Figure 56-2). There is also a heated interlayer under the outer-glass panel. The outer-glass panel has an anti-ice coating inside the panel.

Flight Compartment Side Windows

The flight compartment forward side windows are aft of the windshield. They are constructed with outer and inner panes of glass with a layer of polyvinyl butyral between panes. The outer-glass panel has an anti-ice coating on the inside of the panel.

The flight compartment aft openable side windows are constructed of laminated acrylic/polyvinyl butyral.

Windshield/Windows Storage

CAUTION

When windshields and/or windows are removed (to be reinstalled) store them in a cool, dry room away from radiating heat source and solvent fumes. Store with adequate support to prevent contour change or warping.

NOTE

New windshields and windows must not be removed from their shipping containers until preparation for installation is completed.

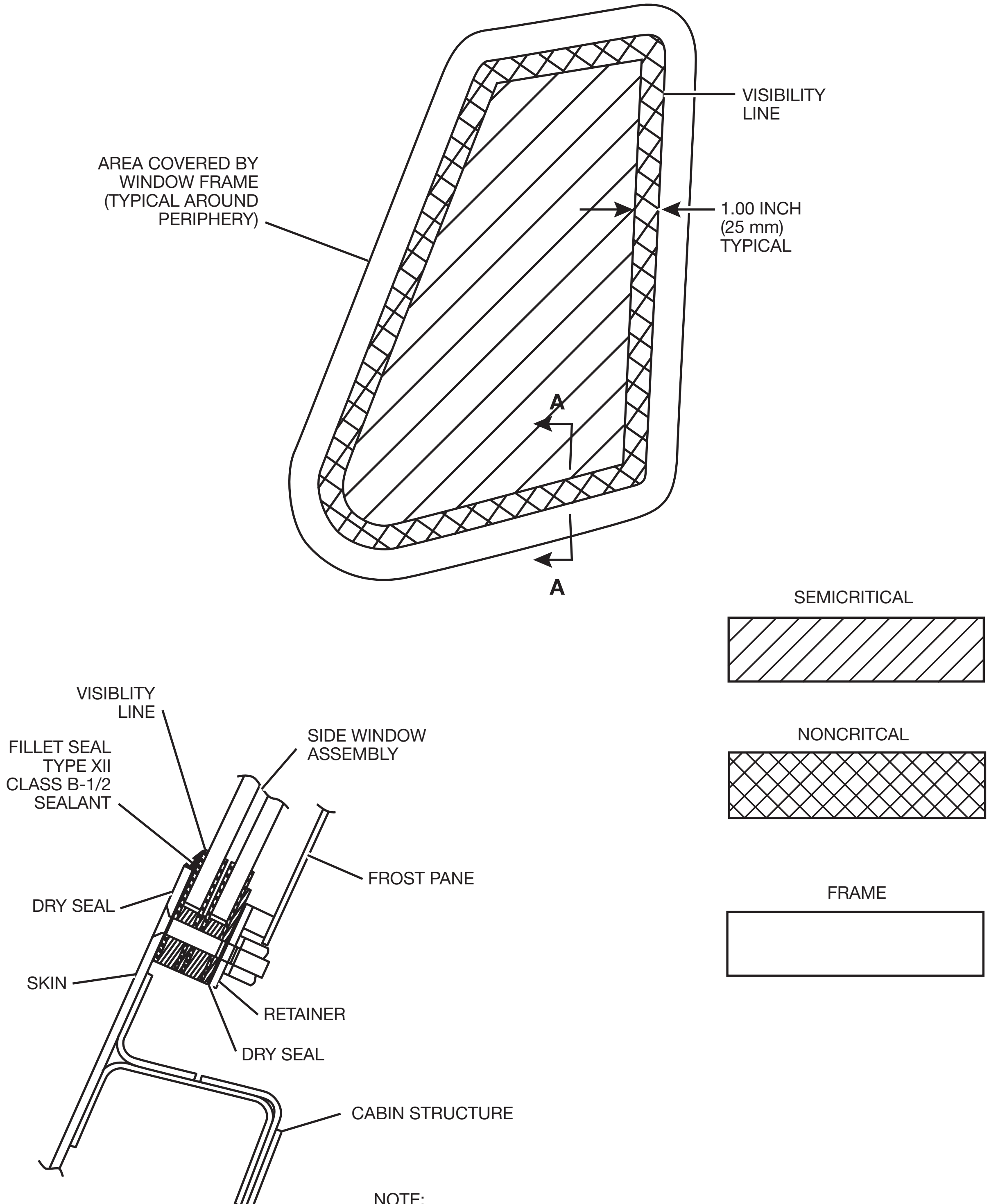
Windshields and windows must be stored in a cool, dry room away from heating coils, radiators, solvent fumes (that can exist near paint storage areas) and in an area where the ambient temperature does not exceed 100°F (43.3°C).

Windshields and Flight Compartment Side Windows Critical Viewing Area

Critical viewing areas of the various windshields and flight compartment side windows are defined in Figures 56-2 thru 56-4.

NOTES

51-57 STRUCTURES



NOTE:
THE VISIBILITY LINE IS CREATED BY THE CIRCUMFERENTIAL EDGE OF THE FILLET SEAL CONTACTING THE OUTER GLASS SURFACE.

Figure 56-3. Glass Forward Side Window Viewing Area

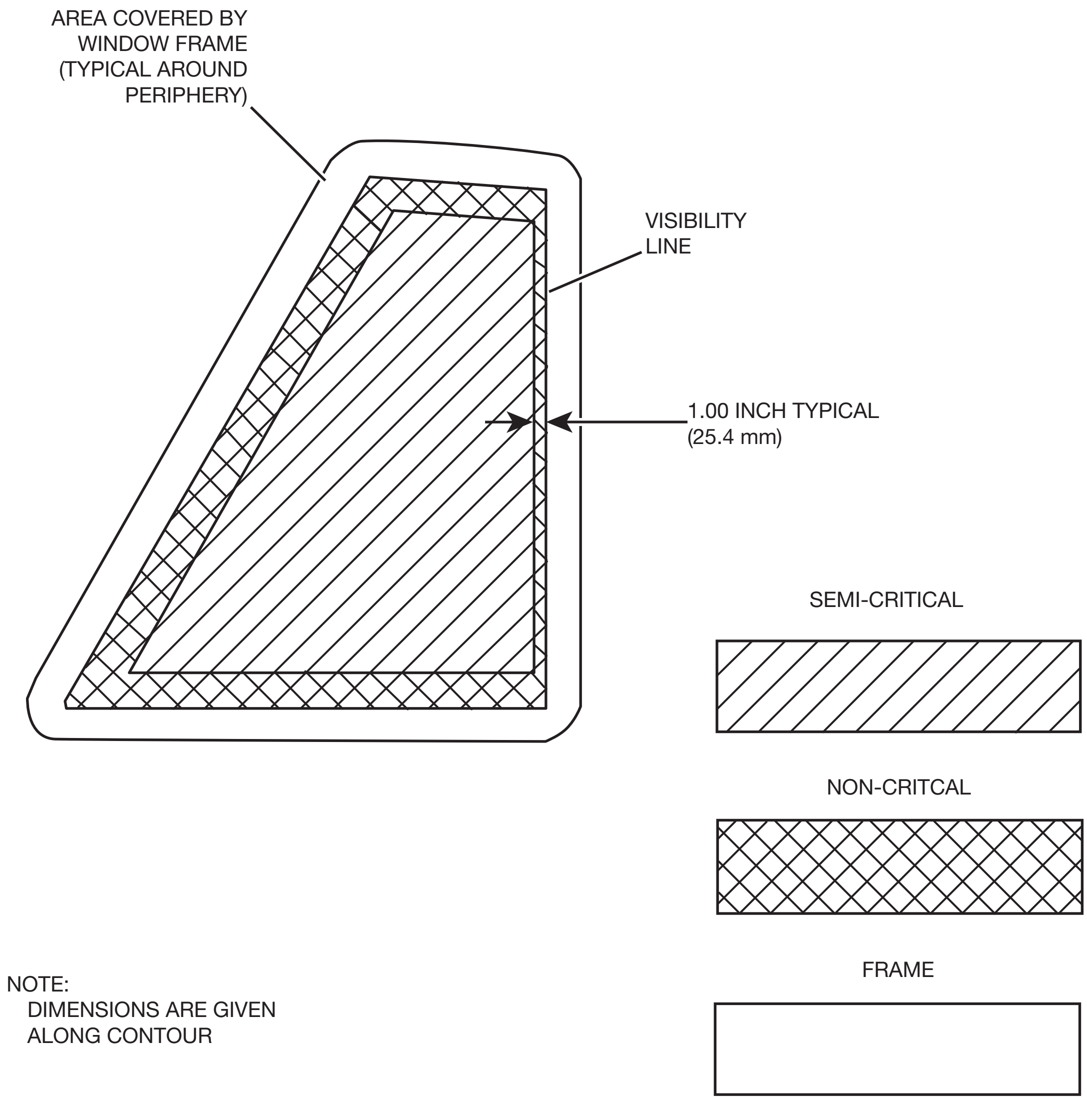


Figure 56-4. Acrylic Aft Side Windows Viewing Area

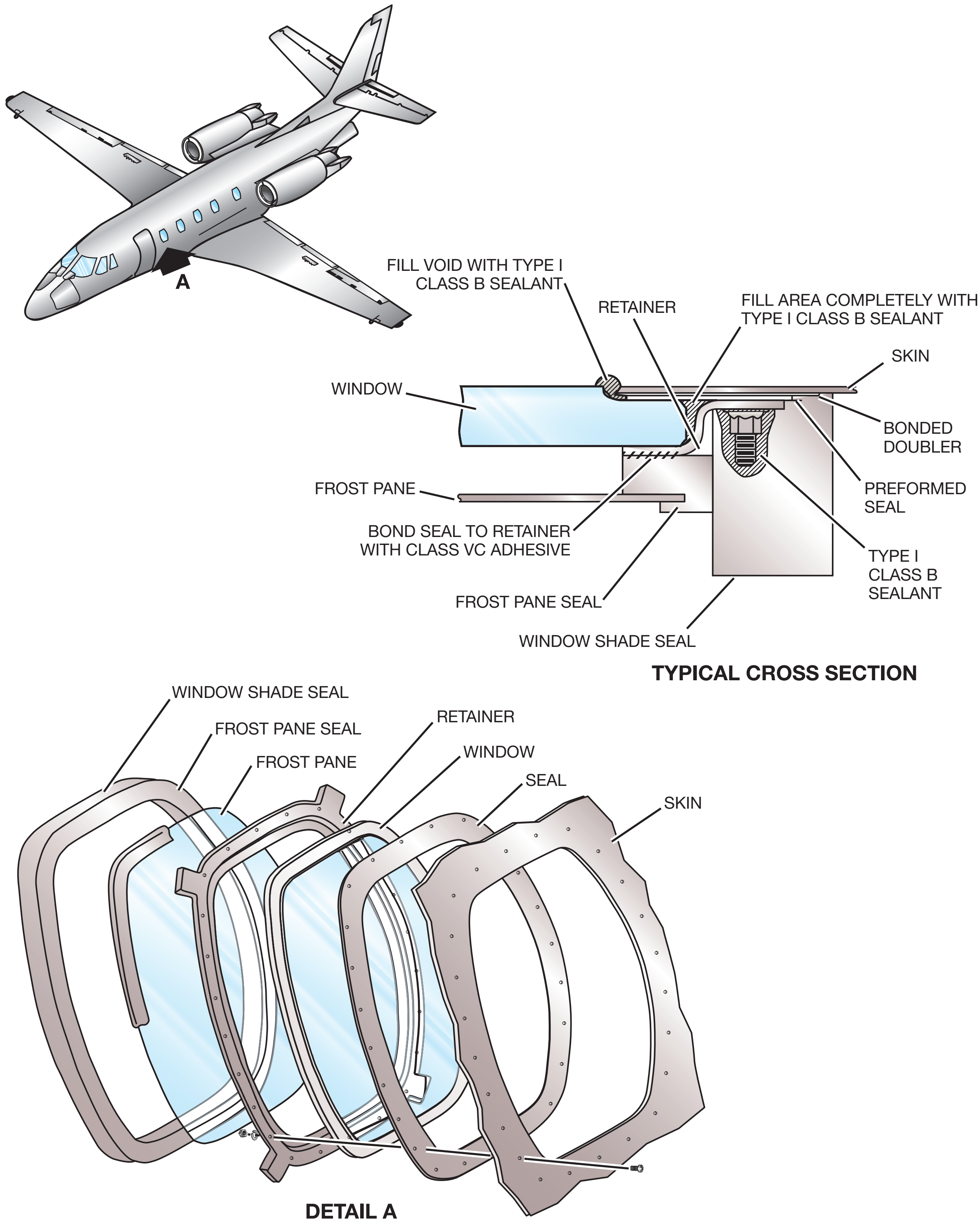


Figure 56-5. Cabin Window Installation

CABIN WINDOWS

This section provides information concerning the cabin windows. All cabin windows, including the emergency exit door window consist of laminated stretched acrylic construction.

Description

Windows

The cabin windows are of laminated acrylic/polyvinyl butyral construction that incorporate a frost pane on the interior side of the window frame (Figure 56-5).

Window Storage

CAUTION

When windows are removed (to be reinstalled) store them in a cool dry room away from radiating heat source and solvent fumes. Store with adequate support to prevent contour change or warping.

NOTE

New windows must not be removed from their shipping containers until preparation for installation is completed.

Windows must be stored in a cool dry room away from heating coils, radiators, solvent fumes (that can exist near paint storage areas) and in an area where the ambient temperature does not exceed 110°F (43.3°C).

NOTES

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CHAPTER 57
WINGS

MAIN FRAME

Description

The aircraft wing assembly is a single unit design, attached under the fuselage at nine points:

- Three on each side
- Locator bolt center forward
- Two yaw links center aft

The wing assembly incorporates a center wing and a wing carry-thru with left and right outer wings permanently attached.

The center wing structure main frame consists of:

- Three spars (forward, center main, rear) crossing the fuselage (at FS 286.12, 335.38, and 370.29)
- Chordwise ribs
- Skin
- Associated structure

The outer wing structure main frame consists of:

- Upper and lower stringers
- Three spars (forward, center (main) and rear)
- Chordwise ribs
- Leading edge
- Skin
- Associated structure

Except for the area above the main landing gear well, the integral fuel tank consists of the entire wing area between the front and rear spars. Liquid-tight ribs at the outboard ends of the wing complete the boundaries of the fuel tanks. Holes in the ribs and spars permit movement of fuel within the wings.

Metal-to-metal joints in the fuel tank structure are sealed to form a liquid-tight structure. The interior surface of the tank is chemically treated and coated with epoxy primer for corrosion resistance.

AUXILIARY

Description

The auxiliary structure of the wing includes:

- The leading edge
- Wingtip
- The area aft of the rear spar (except for the control surfaces)

The wing leading edge is a two-piece fixed design. The inboard leading edge extends form WS 34.000 to WS 101.073. The outboard leading edge extends from WS 101.073 to WS 303.023.

WARNING

During cleaning and polishing operation of wing leading edges, take extreme care not to radius or break sharp corners of boundary layer energizers. Boundary layer energizer edge sharpness must be maintained within a maximum allowable 0.080 inch radii. If radii exceed that limit, boundary layer energizers must be replaced.

There is a stall strip on the inboard leading edge. Eleven boundary layer energizers are on the outboard wing leading edge. The outboard leading edge fence is at W.S.187.95. Each wing has 26 vortex generators on the upper wing surface at 41.4% of wing cord.

The wingtip encloses the outboard end of the wings. The wingtip consists of:

- Three spars (forward, center (main) and rear)
- Chordwise ribs
- Skin
- Associated structure

The wingtip contains:

- Wing recognition
- Landing
- Navigation
- Anti-glare fence
- Strobe/anticollision lights

The wingtip is attached to the wing with screws and bolts.

The wing trailing edge is constructed of ribs, stiffeners and panels.

Supporting structure and fittings are attached for the:

- Speedbrakes
- Flaps
- Ailerons

Outboard of the aileron, the aft edge is enclosed with an triangular aluminum alloy extrusion.

A seal is attached to the wing trailing edge along the flap area to smooth out the airflow between the wing trailing edge and flaps.

FLIGHT SURFACES

The flight control surfaces on the wing include:

- Ailerons
- Flaps
- Speedbrakes

There is an aileron trim tab on the inboard end of the left aileron trailing edge.

Description

The ailerons consist of:

- Spanwise spars
- Chordwise ribs
- Skin
- Associated structure that attach to the wing rear spar by means of three hinge brackets which are on the aileron leading edge

Tuning weights, required to achieve static balance, are attached to the inboard leading edge with screws. The trim tab is attached to the inboard trailing edge of the left aileron with piano-type hinges. The trim tab is constructed of:

- Spanwise spars
- Chordwise ribs
- Skin
- Associated structure

Refer to Chapter 27—"Aileron And Tab System" for additional information.

The flaps (two on each wing) are constructed of:

- Spanwise spars
- Closure ribs
- Graphite composite laminates that attach to flap islands on the inboard trailing edge of the wings

There is a flap track on the end of each flap that mates with permanent rollers (attached to the wing trailing edge). Pushrod attach fittings are on top of the flaps at each end (near the flap tracks). Refer to Chapter 27—"Flaps" for additional information.

Speedbrakes are attached to the upper and lower surfaces of the wing by means of pianotype hinges. They are constructed of magnesium-alloy sheet material and are reinforced with aluminum-alloy tee stiffeners. An attaching lug for the speedbrake actuator is on the inner surface. In the retracted position, the speedbrakes are flush with the contour of the wing. There is a seal on the upper speedbrake that prevents airflow through the speedbrakes when retracted. Refer to Chapter 27—"Speedbrake" for additional information.

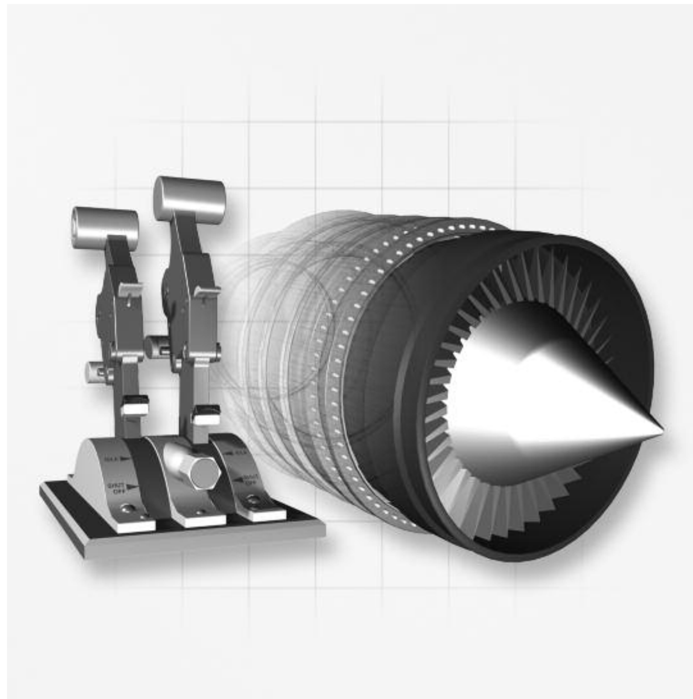
CHAPTER 71–80
POWERPLANT

CONTENTS

ILLUSTRATIONS

TABLES

CHAPTER 71–80
POWERPLANT



INTRODUCTION

This chapter describes powerplants installed on the Citation XL/XLS. Included are descriptions and operation of the major sections, secondary air systems, engine anti-ice system, interstage turbine temperature (ITT) indicating system, oil system, ignition system, fuel system, instrument system, synchronizing system, and thrust reversers. General maintenance considerations are included with an introduction to functional checks for fault analysis. Powerplant limitations are listed. The values listed are intended for training and illustrative purposes. References for this chapter and further specific information can be found in Chapter 5—"Time Limits/Maintenance Checks," Chapter 12—"Servicing," Chapter 54—"Nacelles/Pylons," Chapter 71—"Powerplant," Chapter 73—"Engine Fuel Control," Chapter 74—"Ignition," Chapter 76—"Engine Controls," Chapter 77—"Engine Indicating," Chapter 78—"Exhaust," and Chapter 79—"Oil" of the *Citation XL/XLS* Aircraft Maintenance Manual (AMM).

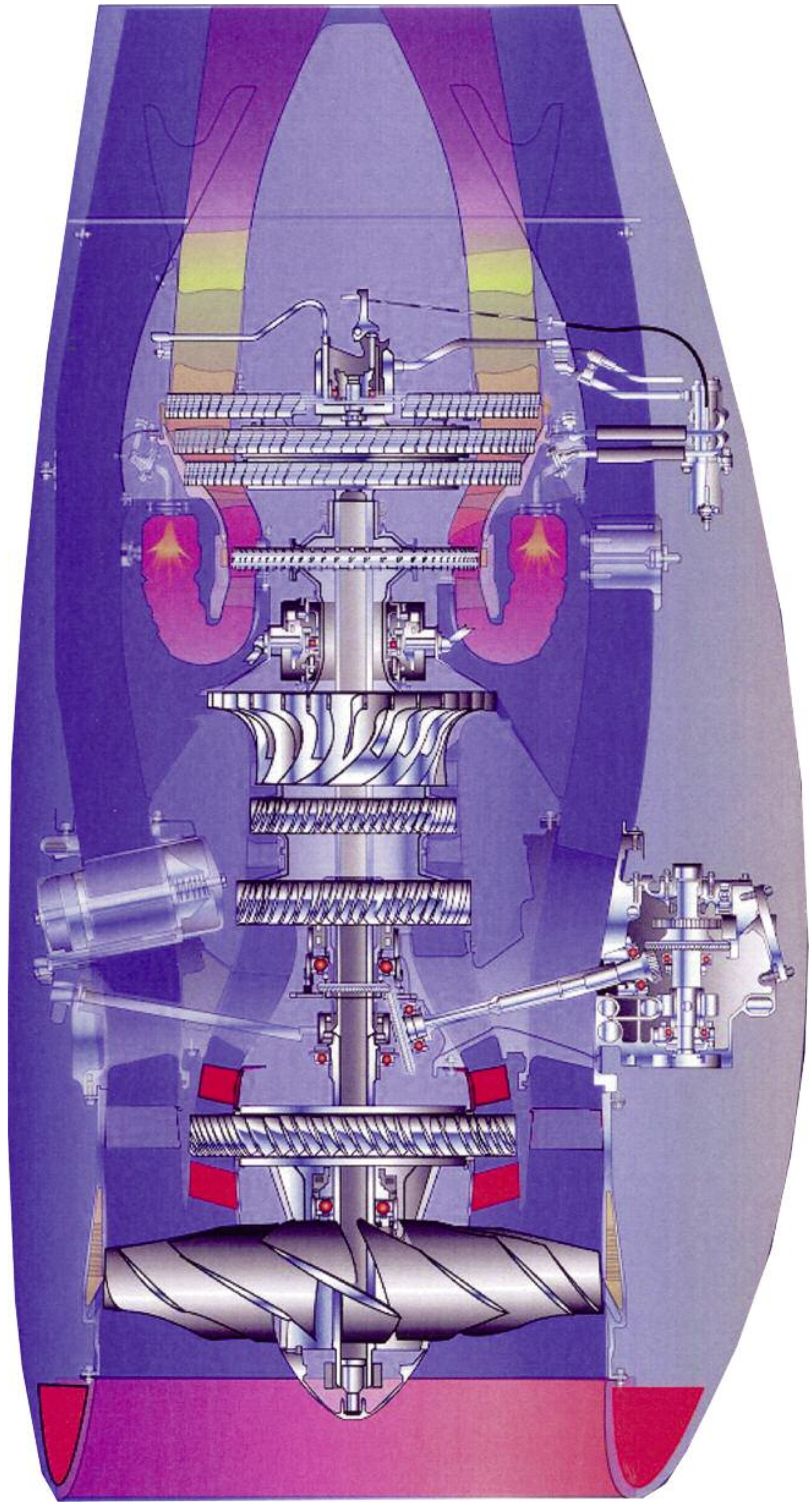


Figure 71-1. PW545A Engine (Sheet 1 of 2)

GENERAL

The 560XL series of aircraft are powered by two the Pratt and Whitney 545 series engines. The PW545 is a low-noise, light-weight, high-bypass ratio, twinspool turbofan engine. (Figure 71-1 through 71-3)

The PW545A engine is installed on aircraft -5001 through 5500 and has a maximum rated thrust on an 83°F (28.3°C) day at sea level of 3,804 pounds (1,725 kg).

The PW545B engine is installed on aircraft -5501 through 6000 and has a maximum rated thrust on an 83°F (28.3°C) day at sea level for each engine is 4,119 pounds (1,794 kg).

The PW545C engine is installed on aircraft -6001 and subsequent and the maximum rated thrust on an 77°F (25°C) day at sea level for each engine is 4,119 pounds (1,868 kg).

Sixty percent of the thrust produced at sea level is from the bypass air and 40% is from the core airflow. Only 40% of the thrust produced at 45,000 feet is from bypass air, and then 60% is from the core airflow.

Overall dimensions of the PW545 are as follows:

- Diameter.................. 32.0 in. (81.3 cm)
- Length .... 68.6 in. (174.2 cm)
- Inlet diameter .... 27.3 in. (69.3 cm)

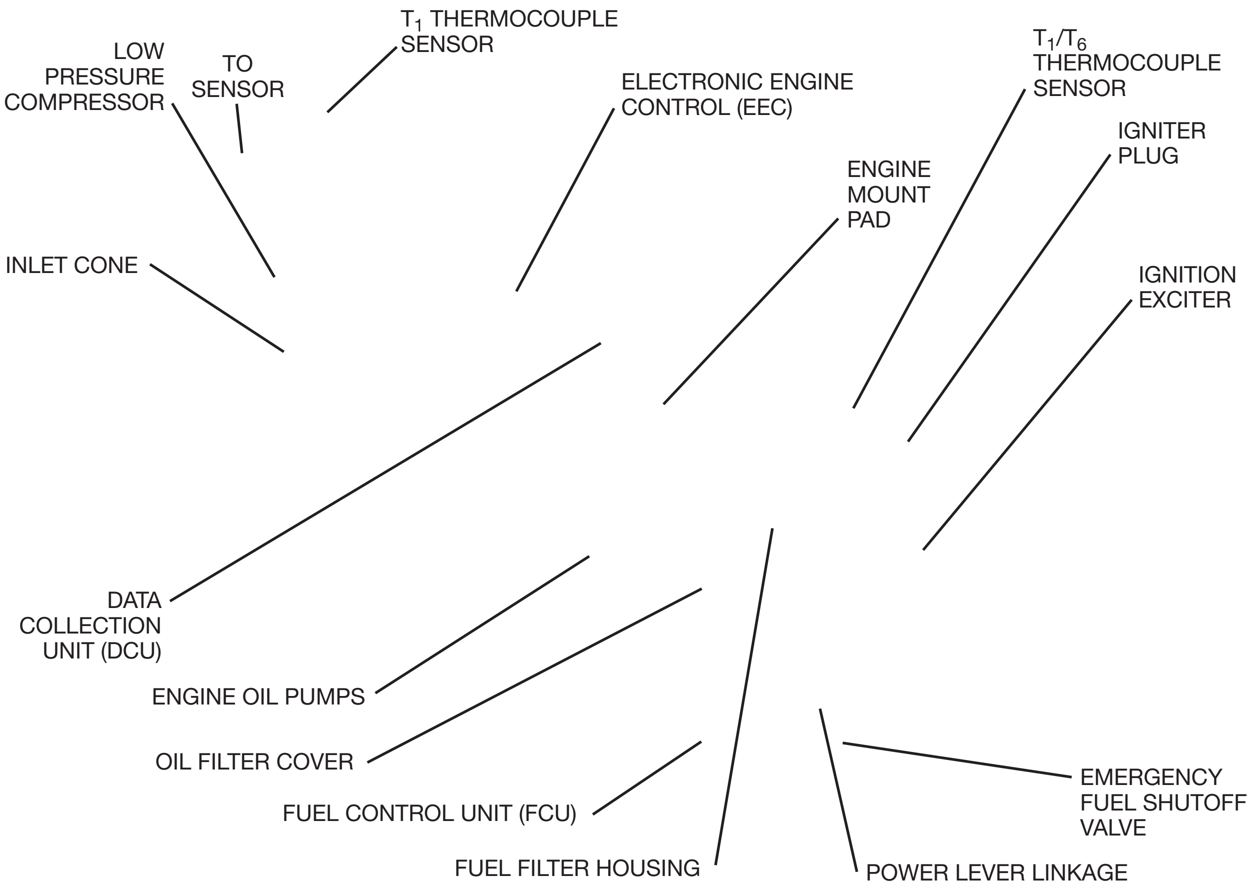


Figure 71-1. PW545A Engine (Sheet 2 of 2)

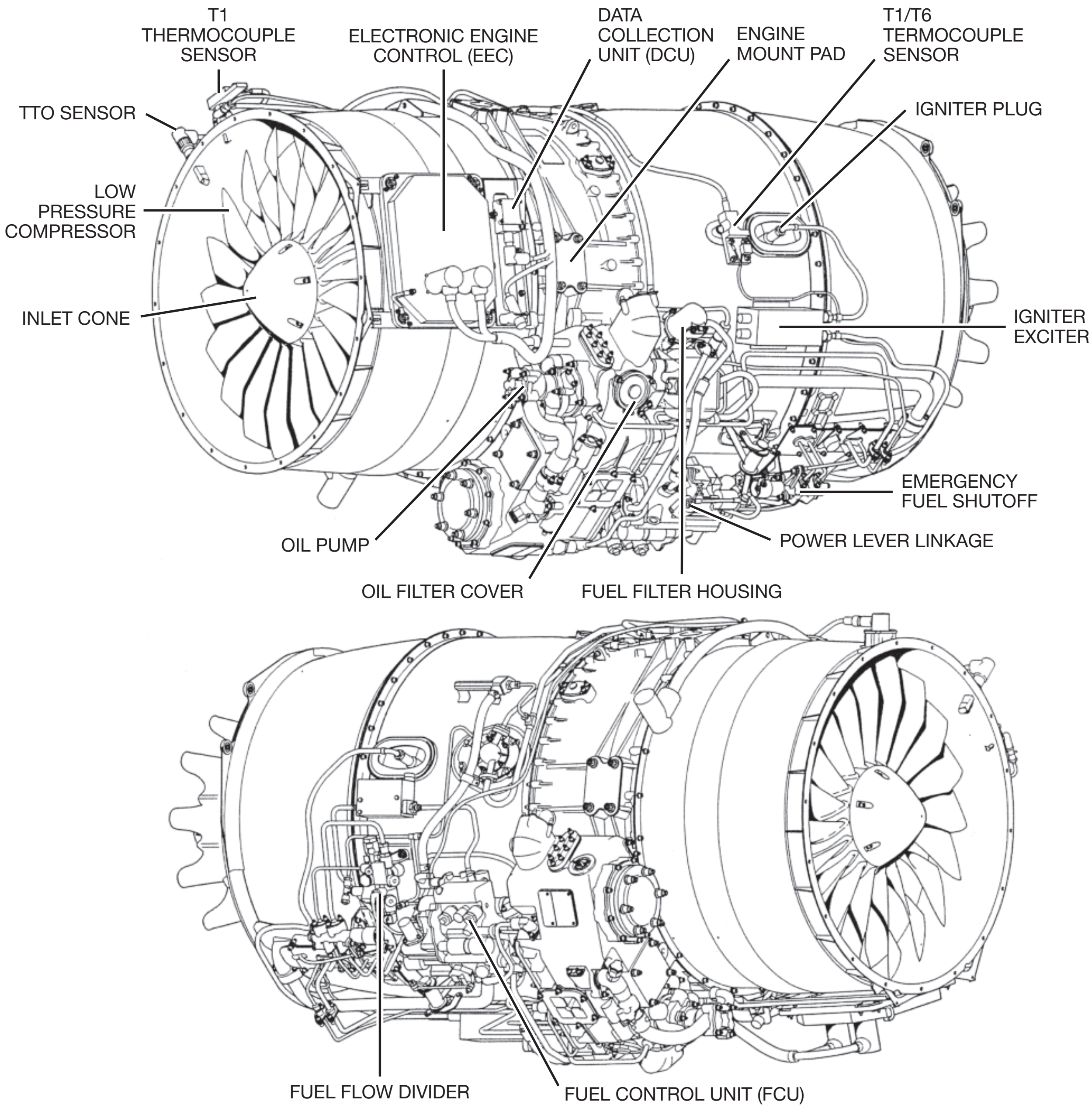


Figure 71-2. PW545A/B Engine

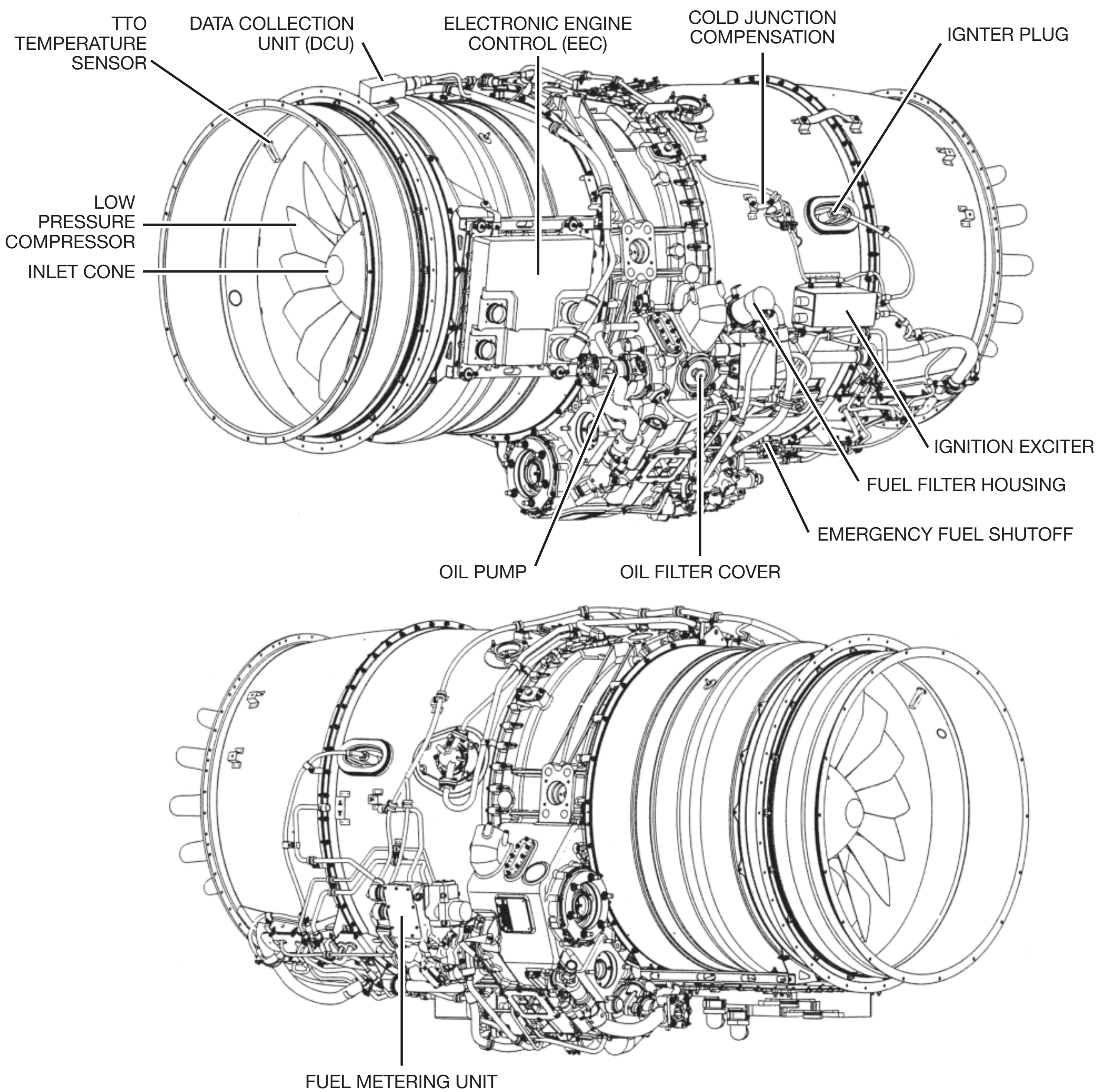


Figure 71-3. PW545C Engine

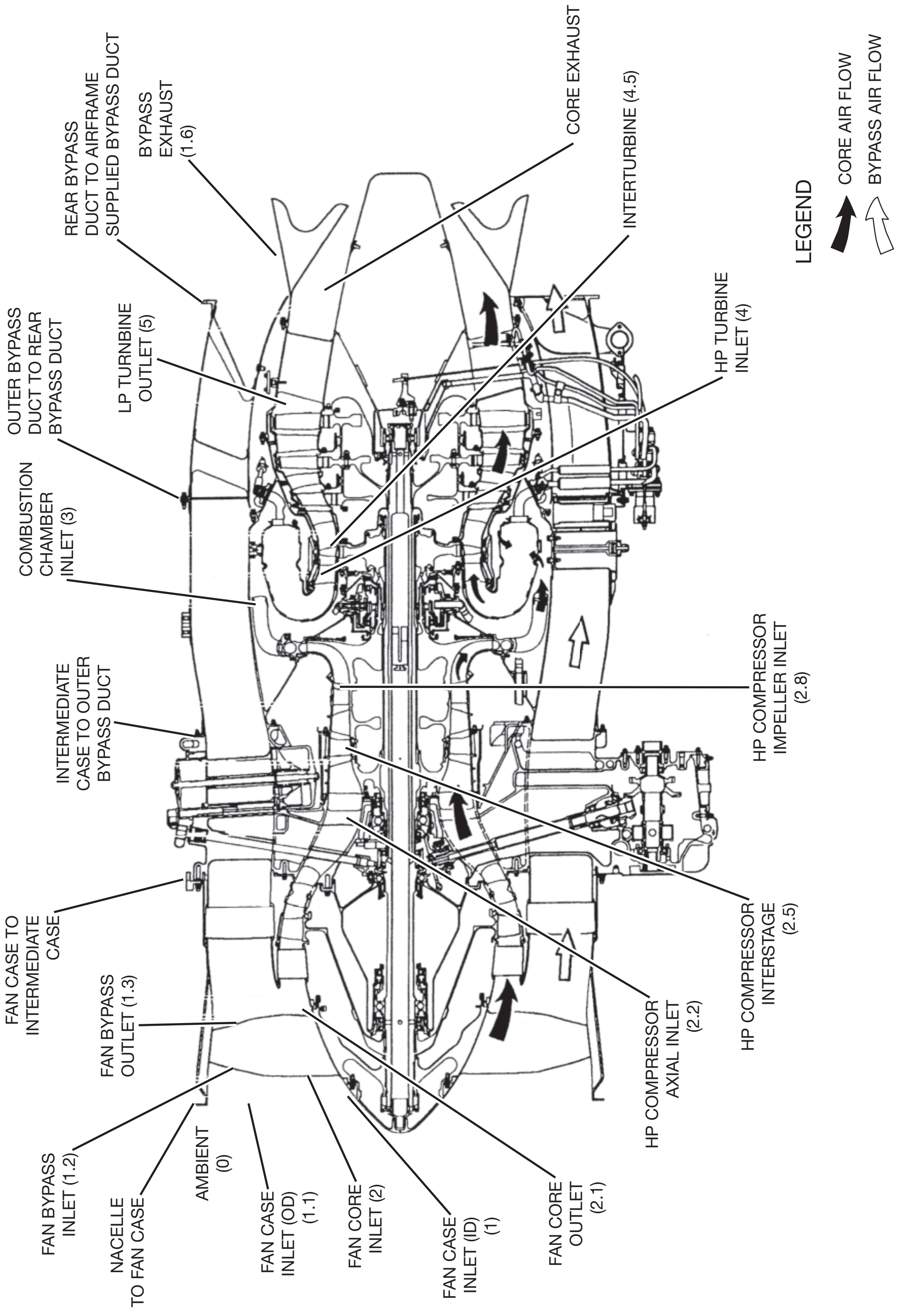


Figure 71-4. Cross Section View

DESCRIPTION

The PW545 engines are nine-stage, twin-spool, turbofan propulsion units that have a full length annular bypass duct (Figure 71-4). The low-pressure compressor is a fan-stage, integrally bladed rotor and boost stage rotor, driven by the three-stage turbine. The high pressure compressor is driven by a singlestage turbine.

The low-pressure (LP) and high-pressure (HP) shafts are counter rotating, turning in counter-clockwise and clockwise directions respectively. Thrust and roller anti-friction bearings are provided on each shaft.

Air enters the engine through the fan case. Fan bypass air flows through a single stator into the bypass duct. The core air flows through an anti-iced stator, a boost stage compressor rotor and intercompressor duct into the HP compressor. Air exits the HP compressor through bolted diffuser pipes. A single bleed valve prevents surge by discharging air into the bypass duct. A supply of core air is taken from the impeller for engine cooling and sealing purposes.

The combustion chamber liner consists of an annular, reverse flow weldment with varying sized perforations that allow entry of compressed air. The primary combustion air enters the combustion chamber liner and mixes with fuel.

Fuel is injected into the combustion chamber by 11 hybrid nozzles. During engine start, the air/fuel mixture is ignited by two igniters, which protrude into the combustion chamber liner. The resulting gases expand from the combustion chamber, reverse direction, then pass through the first-stage vane to the single-stage HP turbine. The first-stage vane is an integrally cast ring with 16 individual cooled airfoils. The gases then pass through the second-stage uncooled LP turbine; then to the three-stage LP turbine and associated stator vanes. Air exits the LP turbine through the exhaust case and a forced air mixer.

The intermediate case forms the main structural path to the airframe and carries the forward engine mounts. The intermediate case also contains an integral accessory gearbox and associated drive system from the HP compressor shaft. All engine-driven accessories, with the exception of the LP (N_1) rotor speed sensor, are on the accessory gearbox. The accessories are driven by a tower driveshaft geared to the HP rotor shaft (N_2), passing down through the intermediate case to mesh with a bevel gear in the accessory gearbox.

The PW545A/B engines can operate in the electronic or mechanical mode.

In the electronic mode (AUTO), a single-channel, electronic engine control unit (EEC), controls the low rotor (fan) speed N_1 response to a pilot-demanded throttle lever angle (TLA). The EEC receives an electronic signal from a rotary variable differential transformer (RVDT) which is connected to the throttle lever. The EEC controls thrust management, compressor surge control, and high and low pressure compressor rotor overspeed protection.

In mechanical mode (MAN), throttle lever angle is transmitted to the hydromechanical fuel control unit (FCU) through the throttle cable. The FCU controls the high compressor rotor speeds and schedules fuel flow during acceleration and deceleration with automatic compensation for ambient temperature conditions.

The PW545C engines are controlled by an engine-mounted, dual-channel, full authority digital engine control (FADEC).

The FADEC controls the high pressure rotor speed and schedules fuel flow during acceleration and deceleration with automatic compensation for ambient conditions. The FADEC also controls the bleed valve to provide compresson surge protection.

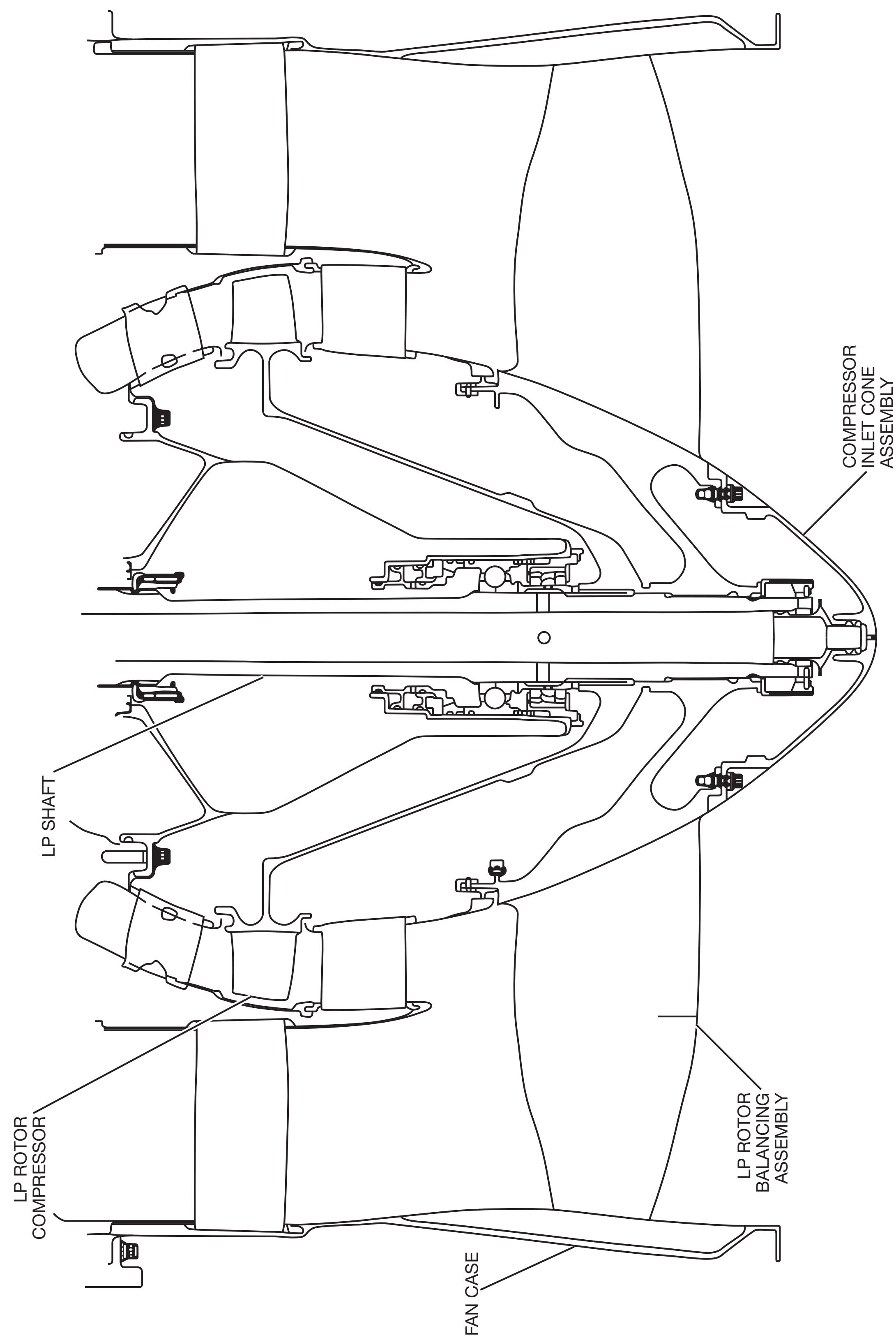


Figure 71-5. Low-pressure Compressor

All engine-driven accessories are installed on the accessory gearbox that is below the engine on the intermediate casing. The accessories are driven by a tower driveshaft geared to the high pressure rotor. The tower shaft passes through the intermediate casing and meshes with a bevel gear on the startergenerator driveshaft to drive the accessories.

MAJOR SECTIONS

DESCRIPTION

For descriptive purposes, the engine is divided into eight major sections as follows:

- Low-pressure compressor (N_1)
- Intermediate case
- High-pressure compressor (N_2)
- Combustion section
- High-pressure turbine (N_2)
- Low-pressure turbine (N_1)
- Exhaust section
- Accessory gearbox

COMPONENTS

Low-pressure Compressor

The low-pressure (LP) compressor accelerates air through the bypass duct and into the core of the engine (Figure 71-5). The booster stage also begins to compress the air to a ratio of 1.25:1 before reaching the high-pressure compressor.

Fan and Booster Stage

The LP rotor balancing assembly and the compressor inlet cone assembly are on the front end of the LP compressor shaft and are housed by the fan case.

The titanium fan is an integrally bladed rotor (IBR) incorporating 19 blades. Fan balancing is achieved with counterweights riveted to front and rear balancing rims. In addition to the fan assembly, an IBR booster stage rotor, with 55 blades is integrated into the LP rotor balancing assembly. The booster-stage rotor and the fan assembly are fitted into the shaft with splines and secured with a fan retaining nut, cupwasher, keywasher, and retaining ring.

Inlet Cone

The compressor inlet cone assembly is a single-skin aluminum-type, which is secured to the fan by six bolts. Each mounting bolt has provisions for a maximum of two counterweights (for field trim balancing). Holes are in the nose cone to LP shaft transfer tube to allow secondary air to pass along the shaft to the nose cone for deicing.

NOTES

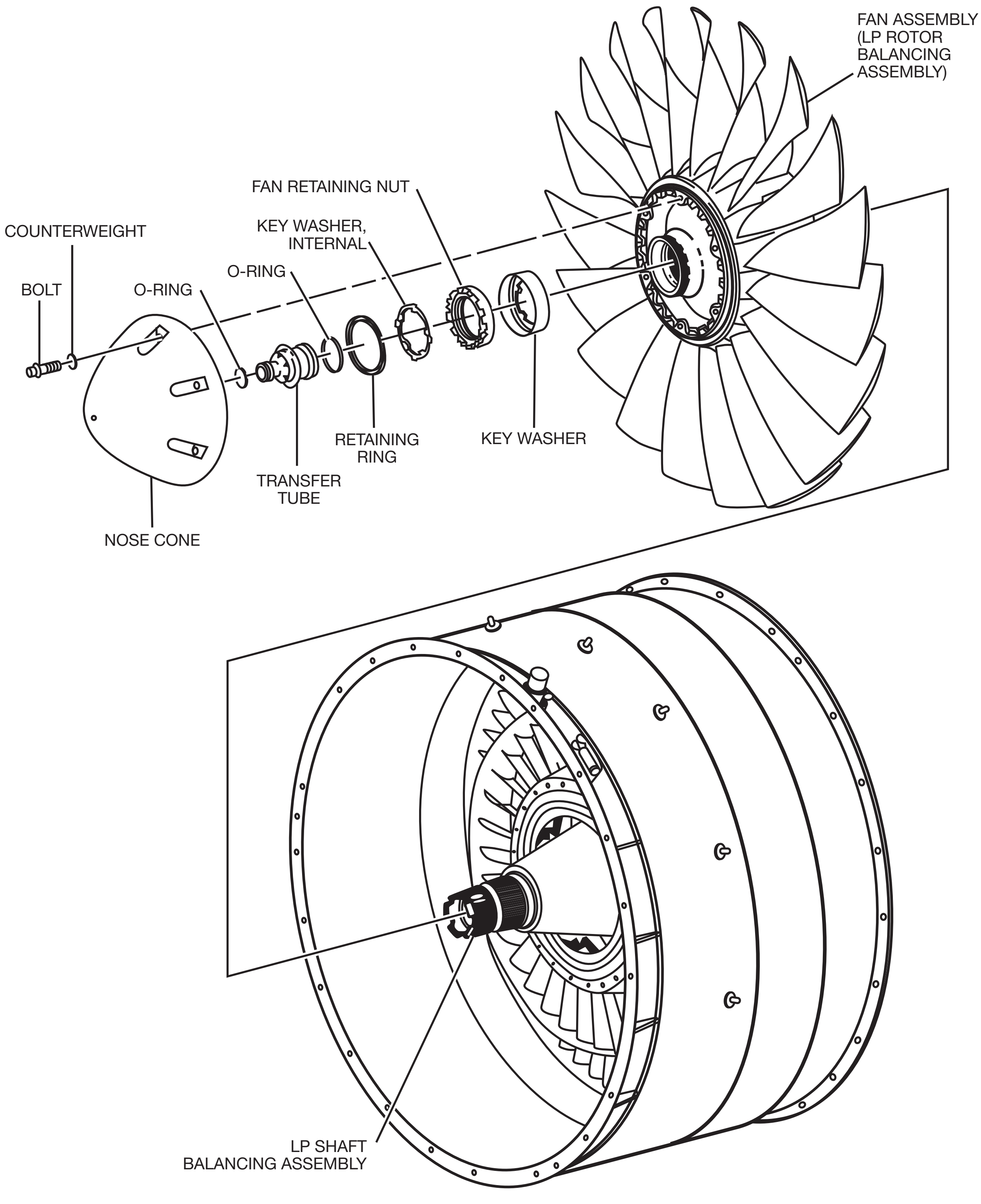


Figure 71-6. Fan and Fan Case

Fan Case

The fan case is a hardwall steel design with an epoxy abradable on the inside diameter (Figure 71-6). The fan stator assemblies divide the airflow, which is then directed from the fan. The inner stator has two rows of fabricated stainless steel guide vanes, which direct the airflow through the compressor intermediate case to the HP compressor. The first row consists of 49 vanes and the next row has 46 vanes. The vanes are brazed to their outer shroud, while a silicone rubber potting compound secures the roots of the vanes to the inner shroud. Anti-icing air passes through these vanes when anti-icing is selected.

The outer stator assembly has a single row of 60 aluminum guide vanes that direct air to the bypass duct. The vanes are secured to both the inner and outer shroud by a potting compound.

Low-pressure Compressor Shaft

The LP compressor shaft is a one-piece steel shaft supported by No. 1 and No. 2 ball bearings at the front, and the No. 5 roller bearing at the rear. The No. 1 and No. 5 bearings are oil-film dampened. The balanced shaft assembly contains an internal counterweight, riveted in place midway along the shaft. Holes in the shaft allow passage of $P_{2.8}$ and P_3 air for pressurizing of seals and nose cone anti-icing.

The No. 2 bearing compartment contains a phonic wheel for measuring N_1 (LP) speed. The phonic ring is aligned with the tip of the speed sensor and is trapped between the No. 2 bearing and a shoulder on the LP shaft. The phonic ring has 29 teeth with one offset gap. It is aligned by a dowel pin in the shaft. The fabricated titanium bearing housing assembly holds:

- The No. 1 and No. 2 bearings
- Bearing oil feeds
- An oil-fed balance piston for supplementary No. 2 bearing load

The housing assembly is secured to the LP shaft by a retaining nut, keywasher, and lockwasher.

NOTES

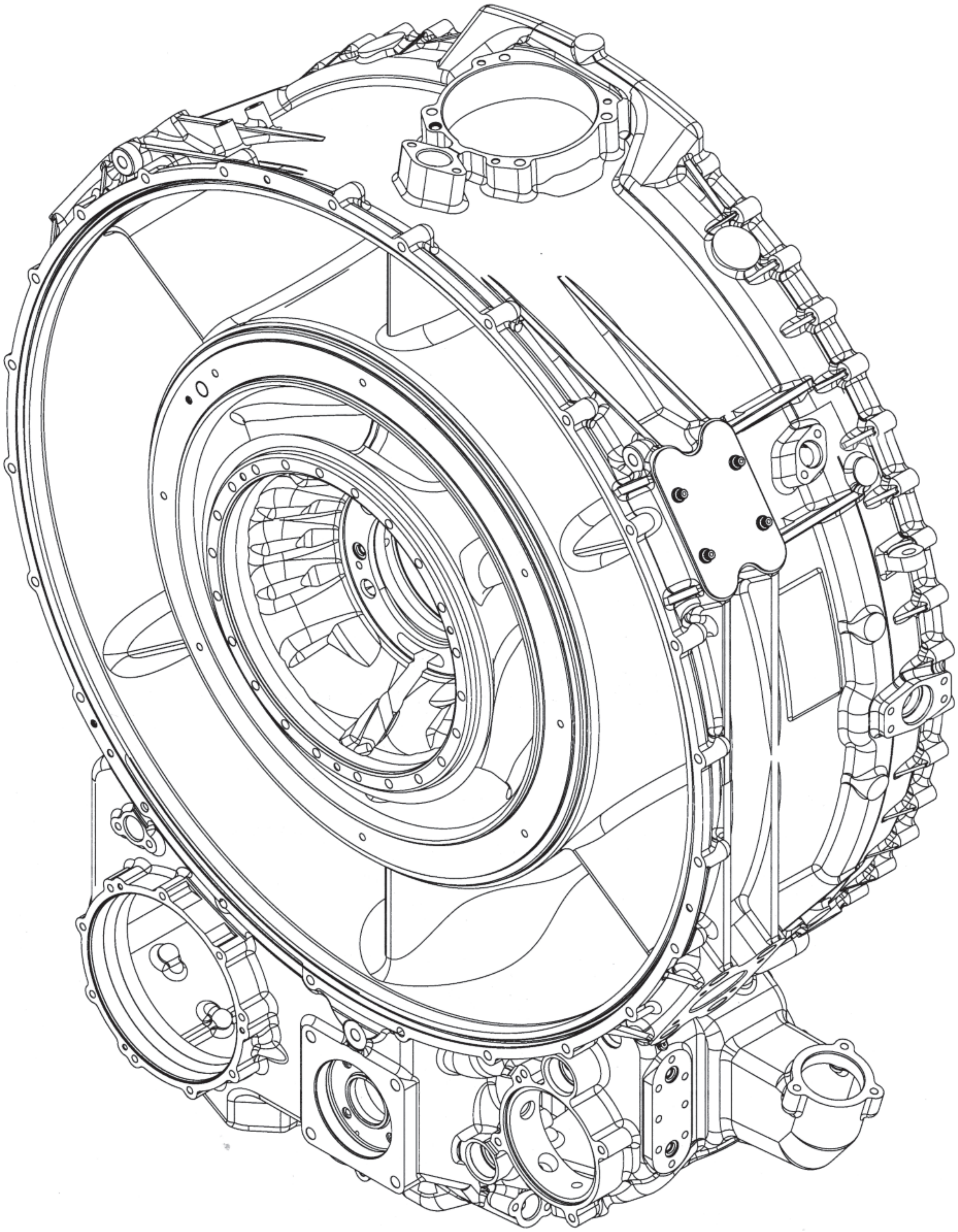


Figure 71-7. Intermediate Case

Intermediate Case

The intermediate case assembly is a machined magnesium casting with two concentric, annular passages for directing core and bypass airflows (Figure 71-7). The case is the main structural member of the engine that supports:

- The fan casing at the front outer flange
- The bypass duct at the rear outer flange
- The gas generator case at the rear inner flange

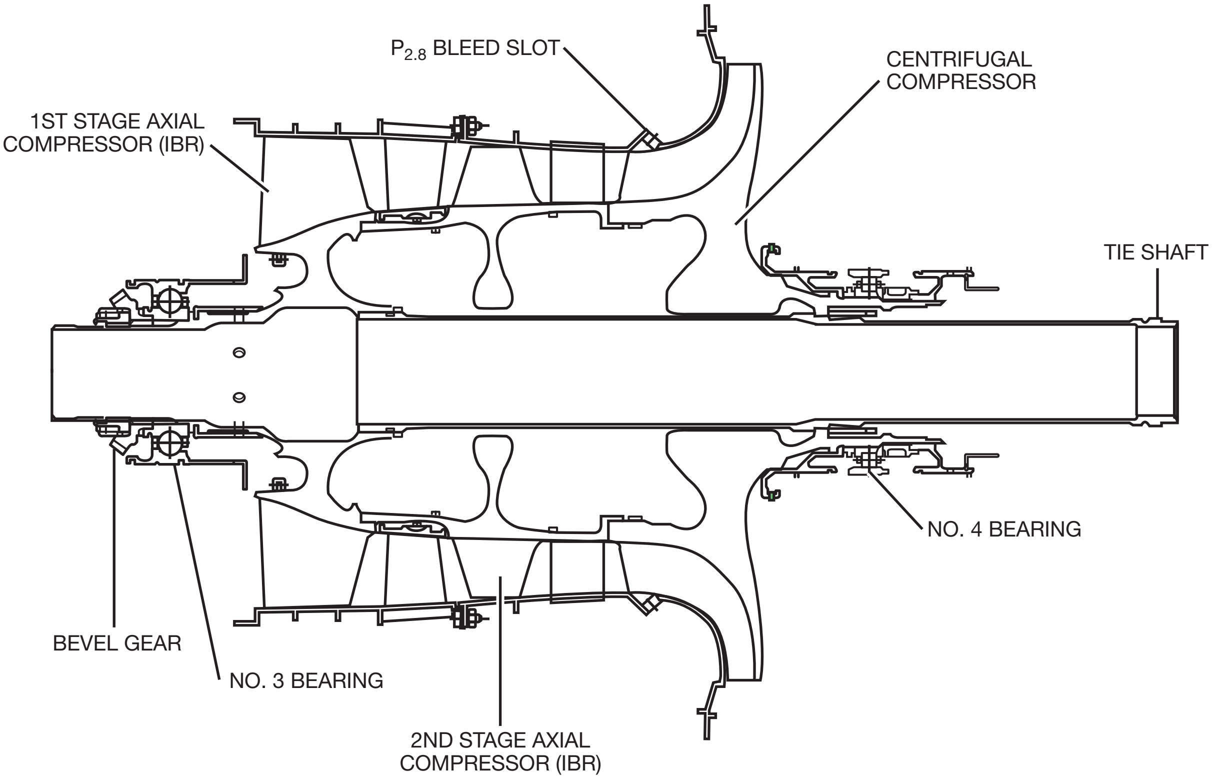
Core airflow passes through the inner passage to the HP compressor while bypass airflow passes through the outer passage to the bypass duct. Six struts join the annular passage walls in the outer passage and six airfoil guide struts in the inner passage. The outer and inner struts are on common radial axes. Struts in the 6 and 12 o'clock positions are hollow.

Four mount pads on the outer wall are inline with the four struts 30° above and below the horizontal centerline. Passageways and bosses provide for oil and air transfer, and various accessories.

On the rear of the case, the No. 3 oil dampened ball bearing and the related aft carbon seal assembly are fitted into the case bore. The No. 1 and No. 2 bearing housing assembly is bolted onto the front of the intermediate case. The N_1 speed sense monopole passes through the hollow strut at top dead-center. It fits into a platform bolted to the No. 2 bearing support. Ahead of the N_1 monopole, the vibration pick-up is mounted with a bracket to the intermediate case front-outer flange. The $P_{2.8}$ bleed valve assembly is behind the N_1 monopole.

The accessory gearbox housing, including a saddle-type oil tank, is integrally cast with the intermediate case.

NOTES



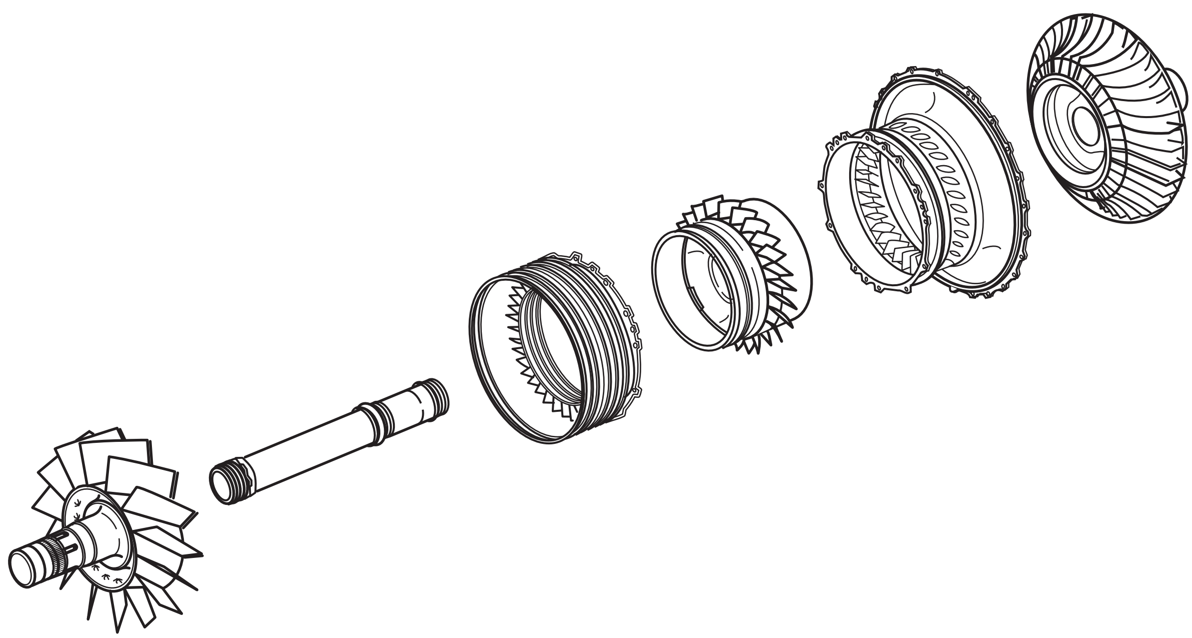


Figure 71-8. HP Compressor

High Pressure Compressor

The HP compressor consists of two axial stages and a single centrifugal stage (Figure 71-8). The HP compressor and rotor assembly (N_2) are detail balanced using a computerized geometric stacking procedure to optimize rotor alignment at engine assembly (Axiam Inc. method.)

The first- and second-stage HP rotors are flank-milled titanium IBRs, clamped together with the titanium impeller using a preloaded central tie shaft, which passes through the bores of these components. The first-stage HP rotor contains 14 integral blades while the second-stage HP rotor contains 19 integral blades. The first-stage stator is an integral-cast steel-shrouded assembly, spigotted (at the front) to the intermediate case and bolted (at the rear) to the second-stage stator assembly. The second-stage stator is a fabricated steel cantilevered airfoil assembly, integral with the impeller housing that bolts to the gas generator case at the diffuser ring.

The impeller, or centrifugal compressor, is forged titanium with 26 machined vanes. The impeller simultaneously compresses and accelerates the air outward, toward the diffuser casing. $P_{2.8}$ air bleeds through slots over the impeller inducer and into a plenum formed by the intermediate case and gas generator case. This air is then discharged into the bypass flow through the bleed-valve assembly, as needed, to prevent stall and surge.

NOTES

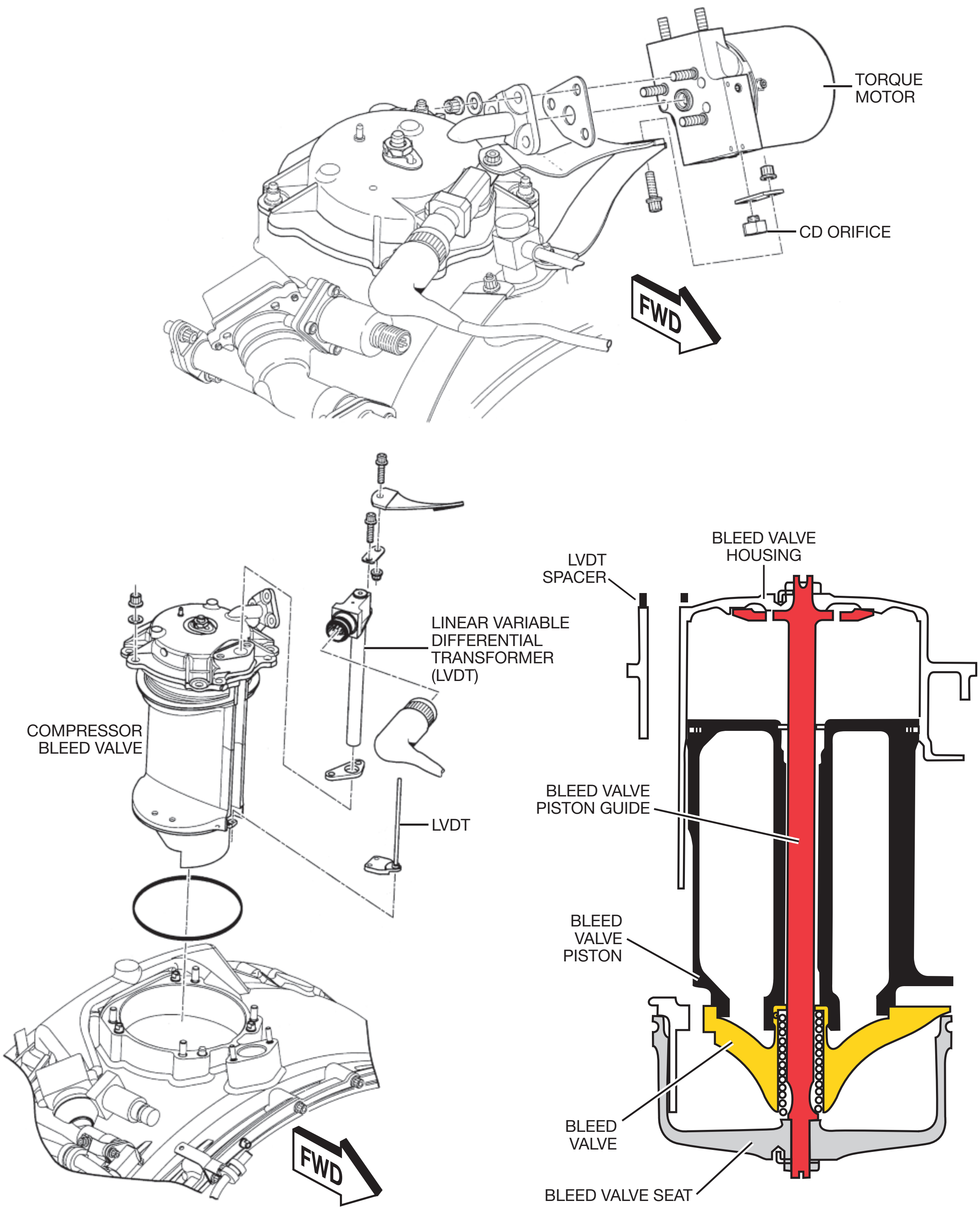


Figure 71-9. Compressor Bleed Valve

Compressor Bleed Valve and Control

The bleed-off valve (BOV) allows for surge-free operation of the engine throughout various operating conditions (Figure 71-9). The BOV is a pneumatically operated piston-type valve that is used to bleed $P_{2.8}$ air off the HP compressor into the bypass airflow. The bleed-valve piston is accurately controlled by the electronic engine control (EEC), which optimizes engine efficiency during steady-state operation and increases surge margin during transient conditions. The EEC controls the BOV position based on corrected N_2 speed and ambient atmospheric conditions.

In the normal electronic mode, the bleed valve is trimmed via a bipolar torque motor in a pneumatic circuit. This arrangement allows the EEC to control the valve via the torque motor to provide optimum surge margin during transients. The torque motor modulates P_3 air into the top of the bleed valve to regulate the piston position (as required by the bleed-valve controller (BVC) portion of the EEC). This modulated P_3 air is referred to as Px. The bleed-valve piston position information is then sensed by a linear variable differential transformer (LVDT) and sent back to the BVC. The BOV is regulated to a steady-state schedule based on corrected compressor speed, with transient offsets during engine acceleration and deceleration.

The BOV reverts to a pneumatic back-up mode of operation in the event of an electronic failure. In pneumatic back-up mode, the BVC signal to the BOV is lost. While the torque motor is deenergized, the BOV assumes a fixed neutral position. Px is now directly proportional to P_3 air and is sufficient to keep the BOV closed. In the event of compressor surges, the BOV opens. When the surge condition has passed, the BOV closes.

NOTES

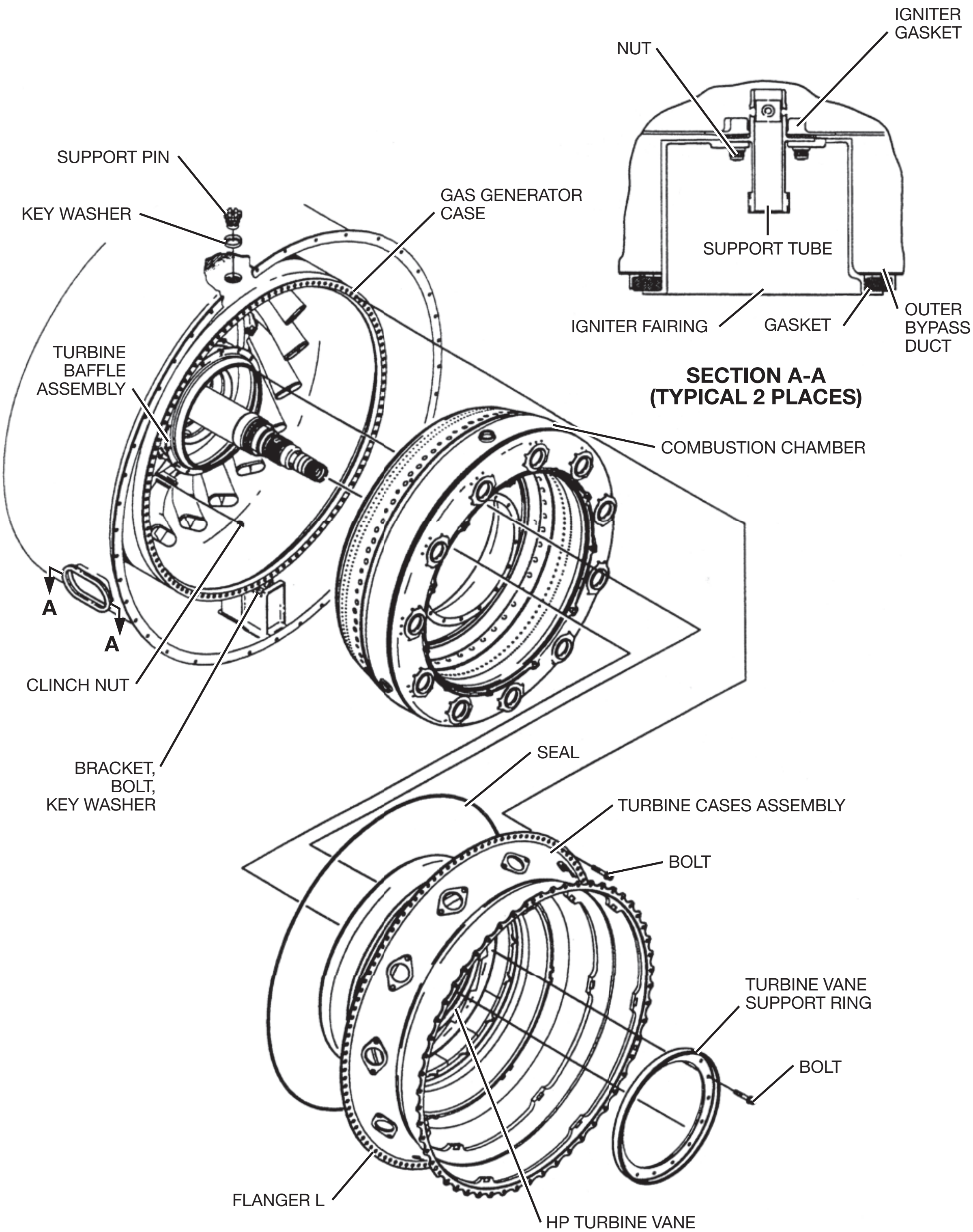


Figure 71-10. Combustion Chamber Liner and Turbine Case

Combustion Section

The combustion section of the engine is contained in the rear section of the gas generator case and consists of the combustion chamber liner, the turbine case assembly, and the HP turbine vane ring.

Combustion Chamber Liner

The combustion chamber liner is a reverse-flow, split-dome design contained within the gas generator case (Figure 71-10). The front end is open. The rear end is domed. It consists of two nickel alloy weldments: the inner assembly and outer assembly. The inner assembly is secured to the outer assembly at the domed end with 11 rivets positioned between the fuel nozzle housings. It is supported in the gas generator case by three pins:

- One at the top-dead-center
- One at the 4 o'clock position
- One at the 8 o'clock position

Spark igniters pass through the support pins at the 4 o'clock and the 8 o'clock positions. Fuel enters through 11 hybrid air-blast fuel nozzles in the domed end.

Compressor discharge air is directed from the diffuser section and discharges into the combustion chamber through metered holes where it is mixed with fuel from the manifold then ignited. The expanding gases flow forward into the domed end where they are turned inward through 180°, then through the HP turbine vane ring to the HP turbine.

A series of perforations allows air to enter the liner in a manner to provide the best fuel/air ratios for engine starting and sustained combustion, with minimum exhaust smoke. Airflow direction is controlled by cooling rings. The perforations ensure uniform temperature distribution at the HP compressor turbine inlet.

Turbine Case

The turbine case assembly consists of inner and outer opposing cones. The front flange is bolted to the gas generator case assembly, and the rear flange is bolted to, and supports, the turbine exhaust duct assembly. The forward edge of the inner cone mates with the combustion chamber liner. The HP turbine liner segments are on the inner cone. The outer cone provides mounting for the 11 fuel nozzles and the fuel manifold assembly.

NOTES

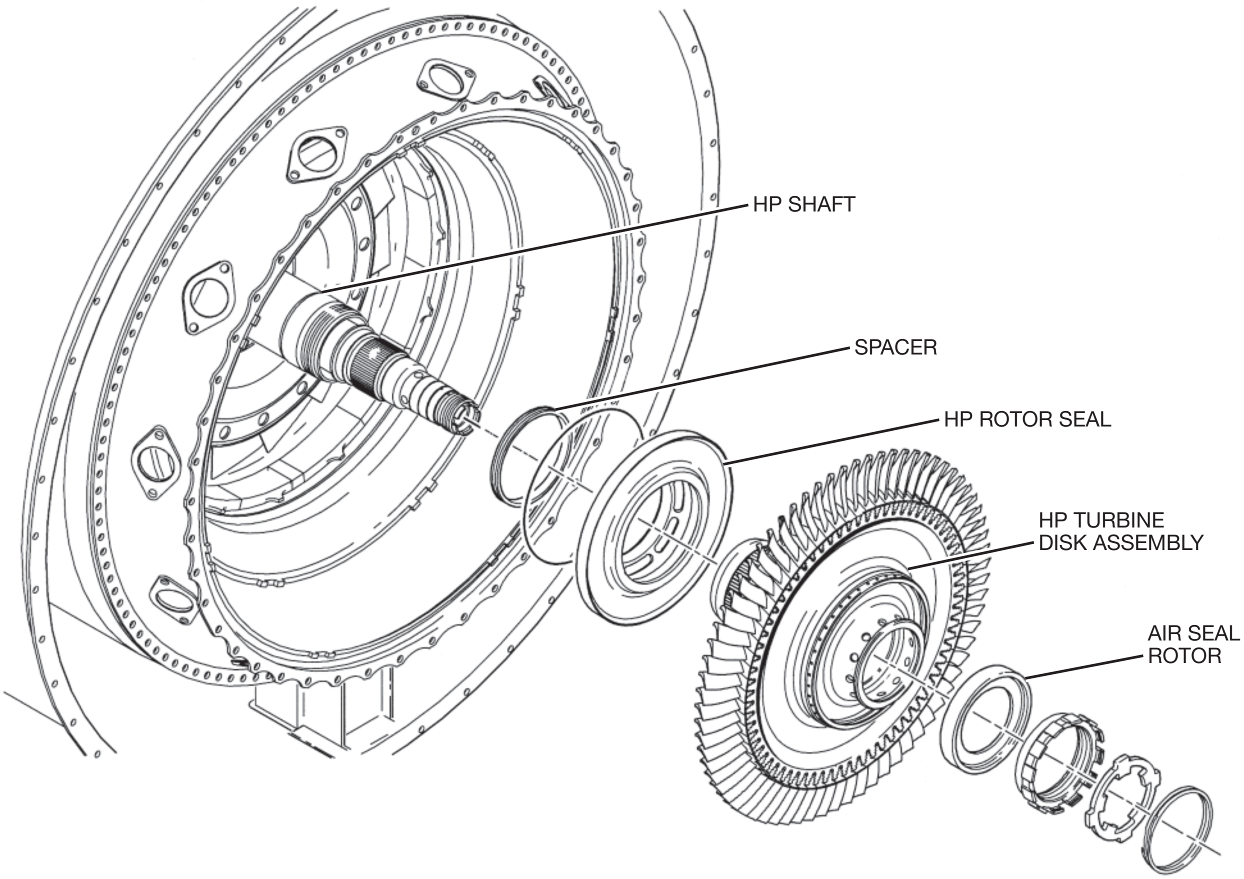


Figure 71-11. HP Turbine Disk Balancing Assembly

High-pressure Turbine Section

The HP turbine section consists of the HP turbine vane ring and the HP turbine disk assembly. The HP turbine section follows immediately after the combustion section and is the first area that the hot expanding gases pass through. The HP turbine drives the two-stage axial HP compressor and the single-stage HP impeller. Splines on the integral stubshaft on the disk mate with splines on the impeller. The disk is secured to the HP shaft by the HP turbine nut.

HP Turbine Vane Ring

The HP turbine vane ring assembly is an integrally cast, nickel alloy ring with 16 air-cooled vanes. Cooling air enters the vane outer ring and exits into the gas path through holes in the vane trailing edges. The vane ring is between the combustion chamber and the HP turbine rotor. The vanes direct the expanding gases to the HP turbine blades at the proper angle to drive the turbine.

HP Turbine Disk

The HP turbine disk balancing assembly is a single-stage turbine consisting of a two plane balanced turbine disk, 70 blades, and classified counterweights (Figure 71-11). The blades are of a fir tree type design. They are secured in the disk with tubular rivets.

The assembly is an independently balanced unit that can be replaced by another suitably balanced unit without affecting the HP rotor balancing assembly. The class and location of the counterweights is recorded on the engine data plate and in the engine logbook. A geometric stacking procedure, which determines the best angular alignment of the disk, is also used. The angular alignment information for replacement disk assemblies is available from Pratt & Whitney Canada, Inc.

NOTES

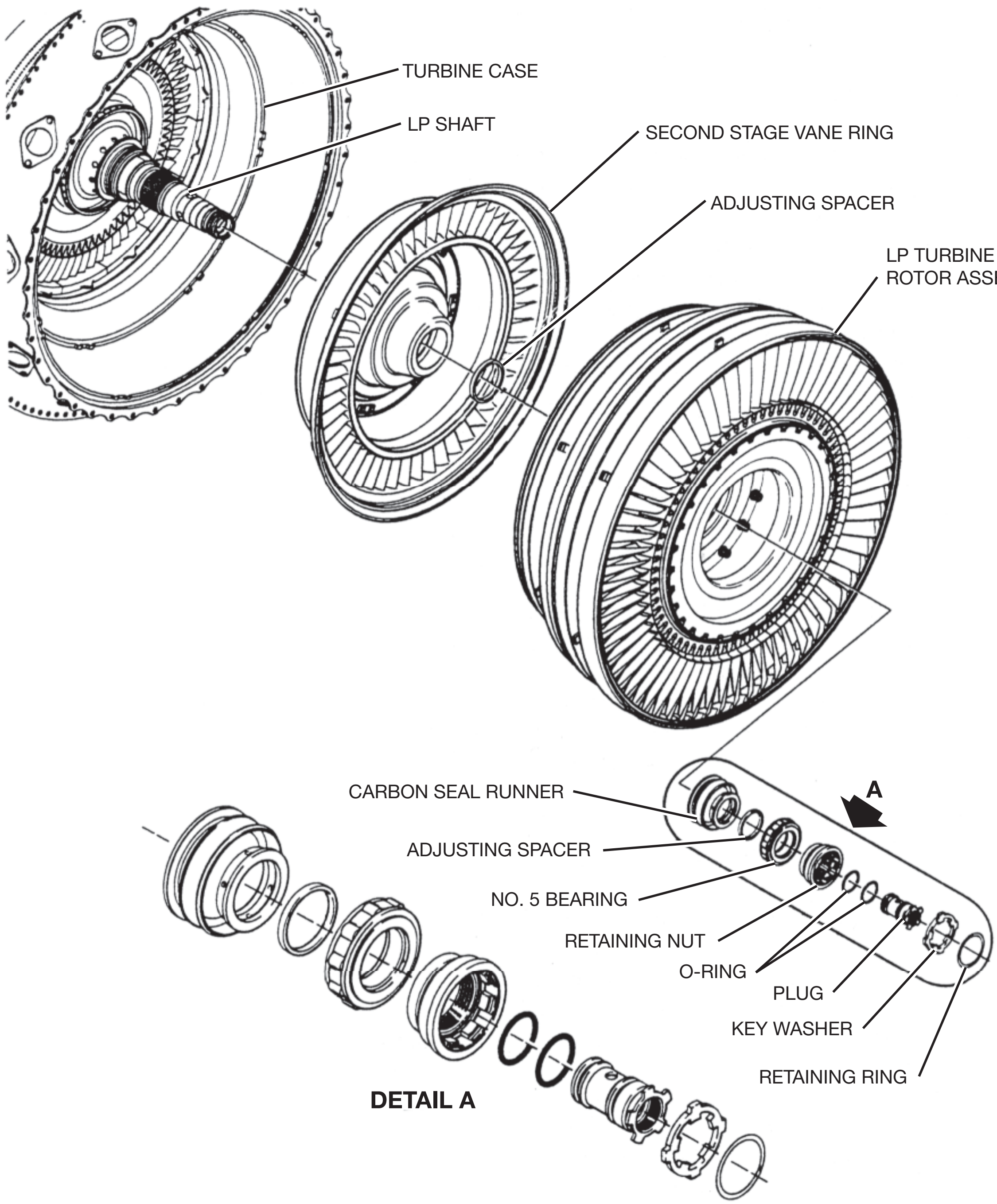


Figure 71-12. LP Turbine Assembly

Low-Pressure Turbine Section

The function of the low-pressure (LP) turbine assembly is to extract energy from the remaining hot gases in order to turn the LP compressor. The LP turbine disks are just aft of the HP turbine disk (Figure 71-12). These three disks are considered the second, third, and fourth stage turbine disks. The HP disk is the first stage.

LP Turbine Rotors

The LP turbine rotor-balancing assembly consists of three LP turbine disk assemblies with two interstage vane rings supported between the disks.

The second-stage disk has 64 blades; the third-stage disk has 62 blades; and the fourth-stage disk has 64 blades. The blades are secured in the disks with rivets. The assembly is balanced with counterweights on the second-stage and fourth-stage disks.

The LP turbines drive a single-stage fan and a boost stage compressor rotor. Splines on the third-stage disk hub mate with splines on the LP shaft. The LP rotor assembly, carbon seal runner, adjusting spacer, and No. 5 bearing are secured to the LP shaft by a retaining nut.

LP Turbine Stators

An LP turbine stator assembly (second stage vane ring) is just upstream of the LP turbine rotor assembly. Two interstage stators are sandwiched between the second, third, and fourth stage rotors. These sandwiched stators are considered the third and fourth stage stators (respectively). All of the LP stator assemblies are cast of a nickel alloy material, and have only one vane flow class for each stage stator. The second stage vane ring incorporates 51 vanes; the third stage has 63 vanes; and the fourth stage has 65 vanes.

NOTES

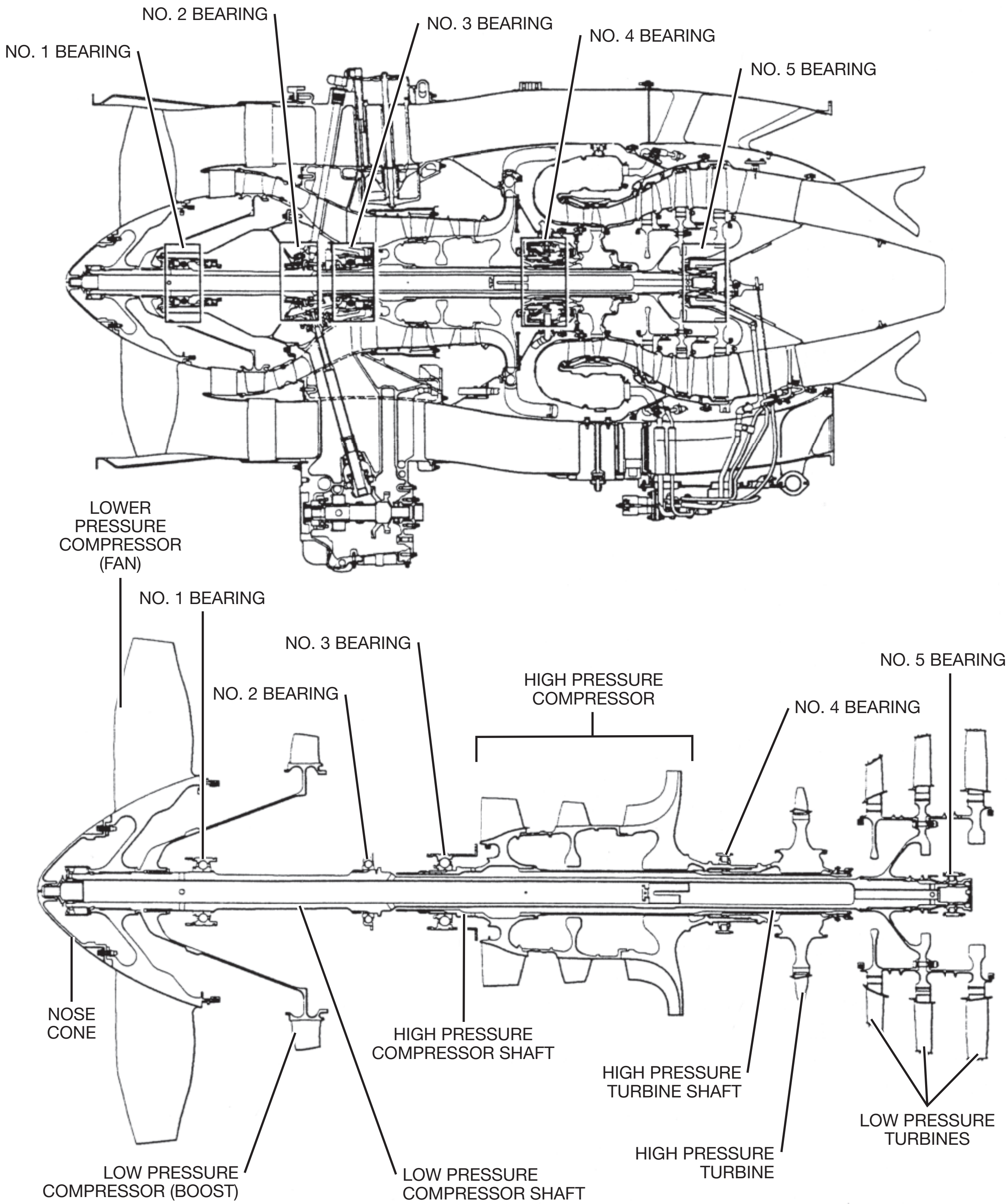


Figure 71-13. Main Engine Bearings

Main Engine Bearings

The main engine bearings support the low-pressure (N_1) and high-pressure (N_2) rotors; and absorb radial and axial loads (Figure 71-13). Ball bearings are designed to absorb both the radial and axial loads, while roller bearings are designed to only support radial loads. The PW545A incorporates both types, depending on the loads at the bearing's respective position in the engine. The following is a list of the bearings, position, and type:

Roller bearings are composed of an inner race with an incorporated roller cage, and an outer race.

Ball bearings are composed of two split inner races, a ball cage, and an outer race.

Table 71-1. BEARINGS LIST

BEARINGS	POSITION	TYPE
LP rotor	No. 1	Ball
	No. 2	Ball
	No. 5	Roller
HP rotor	No. 3	Ball
	No. 4	Roller

Nos. 1, 3, and 5 bearings are oil dampened to absorb vibrations. "Oil dampening" is a term that means supplying HP oil under the bearing race, allowing the bearing to ride on a film of oil, to avoid metal-to-metal contact. The race is then keyed to prevent it from spinning in the housing.

Oil pressure is also applied to a piston behind the No. 1 bearing, in order to hydraulically load it forward. This loads the No. 2 bearing forward as well, and minimizes axial floating and vibration as the engine accelerates and decelerates.

NOTES

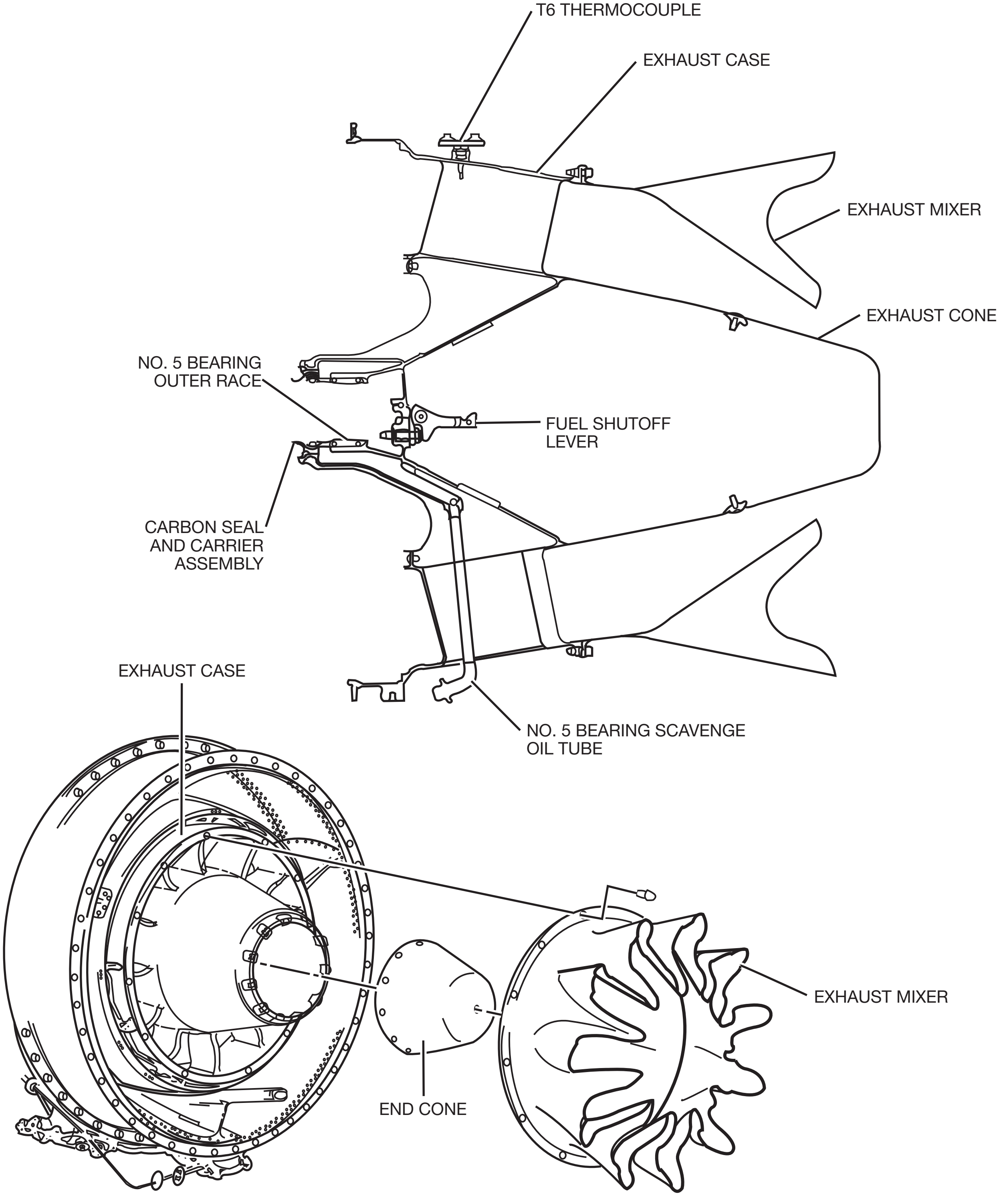


Figure 71-14. Exhaust Section

Exhaust Section

The turbine exhaust assembly consists of:

- Exhaust case assembly
- Exhaust mixer
- Exhaust cone (Figure 71-14)

The front flange of the exhaust case is bolted to the turbine case. The exhaust mixer is bolted to the exhaust case rear outer flange, and the exhaust end cone is bolted to the rear inner wall.

Exhaust Case

The exhaust case supports the No. 5 bearing outer race and a fuel shutoff mechanism. The T6 thermocouples are in the outer wall. Eleven hollow struts, which pass through the engine gas path, support the inner and outer walls of the exhaust case. The pressure oil tube for the No. 5 bearing cavity passes through the strut at the 9 o'clock position. The scavenge oil tube and the fuel shutoff cable pass through the strut at the 6 o'clock position.

Exhaust Mixer

The exhaust mixer is a twelve-lobed, forced mixer configuration. The forced exhaust mixer reduces turbulence by gradually mixing core and bypass flow for a smoother exhaust evacuation. This enhances performance and also helps reduce noise levels by increasing the frequency of the sound, which reduces more rapidly as the distance from the source increases.

Exhaust End Cone

The exhaust end cone completes the core section of the engine. The end cone incorporates a hole in the end for borescope inspection of the fuel shutoff mechanism, which is housed inside.

NOTES

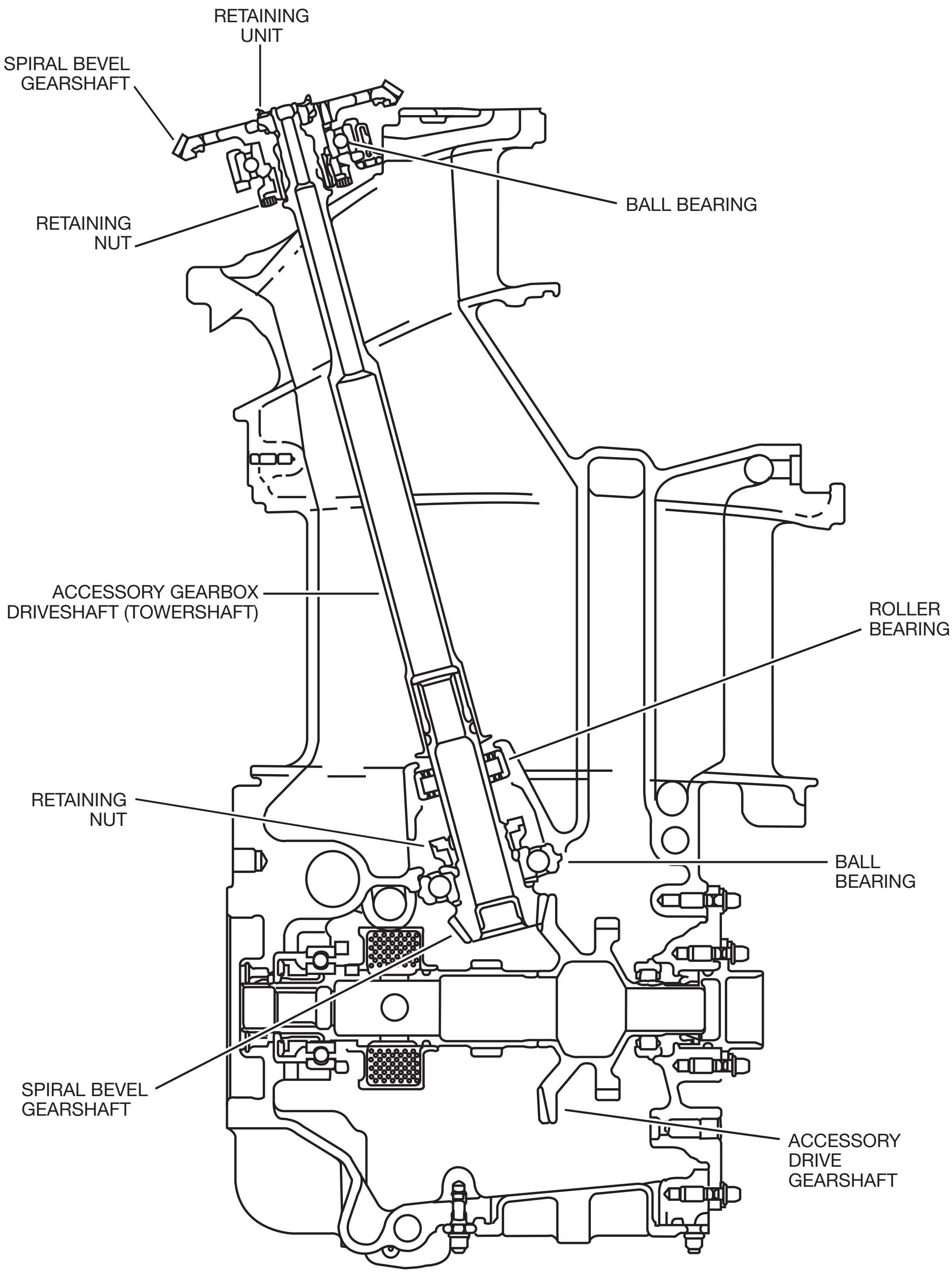


Figure 71-15. Towershaft

Towershaft

The towershaft is the connecting link between the HP section and the accessory gearbox (Figure 71-15). The accessory gears are driven by the HP rotor shaft (N_2) via the tower drive shaft assembly, as it passes down through the intermediate case. A ball bearing at the top and a ball bearing roller bearing at the bottom support the tower drive shaft assembly. A spiral-bevel gearshaft at the top of the tower drive shaft meshes with a spiral-bevel gear on the HP rotor shaft. A spiral-bevel gearshaft at the bottom of the tower drive shaft meshes with a spiral bevel gear on the accessory drive gearshaft.

The starter motor turns the HP section during start using the towershaft, and during all other engine operations, the towershaft enables the HP section to turn the accessory gearbox.

NOTES

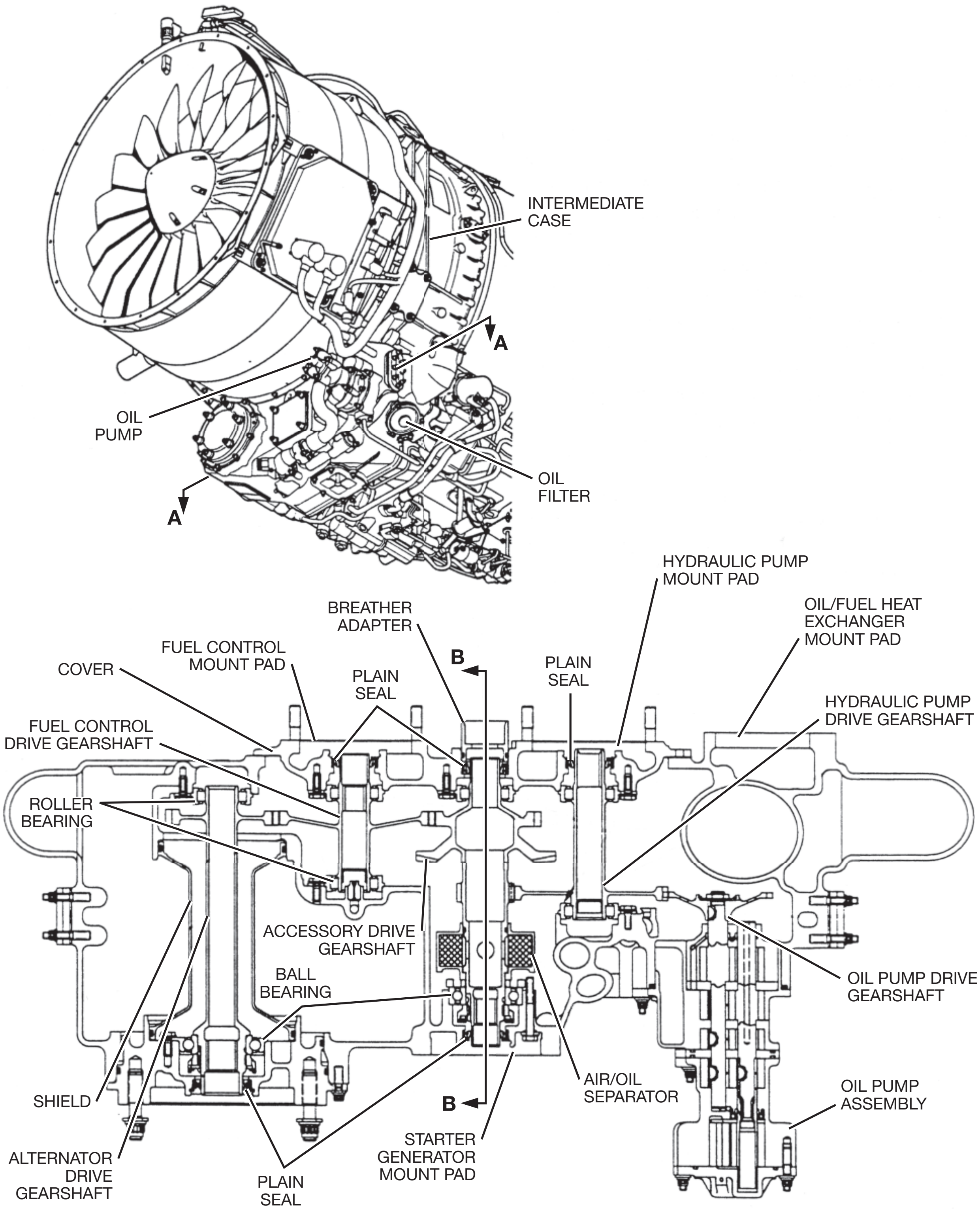


Figure 71-16. Accessory Gearbox

Accessory Gearbox

The accessory gearbox (AGB) is an integral part of the intermediate case assembly (Figure 71-16).

The AGB has a rear cover to support all necessary gears for each accessory. All engine driven accessories (with the exception of the Nl LP rotor speed sensor) are on the accessory gearbox.

The accessory drive gearshaft is supported by a ball bearing at the front and a roller bearing at the rear. Plain seals (garter-type) are at both ends of the gearshaft. Two spur gears on the accessory drive gearshaft drive the fuel control gearshaft and the hydraulic pump drive gearshaft. The spur gear on the hydraulic pump drive gearshaft drives the oil pump drive gearshaft. The spur gear on the fuel control drive gearshaft drives the alternator drive gearshaft. The accessory drive gearshaft also incorporates a retimet-type air/oil separator. Breather air passes through the breather adapter then exits toward the rear through a breather tube into the engine exhaust.

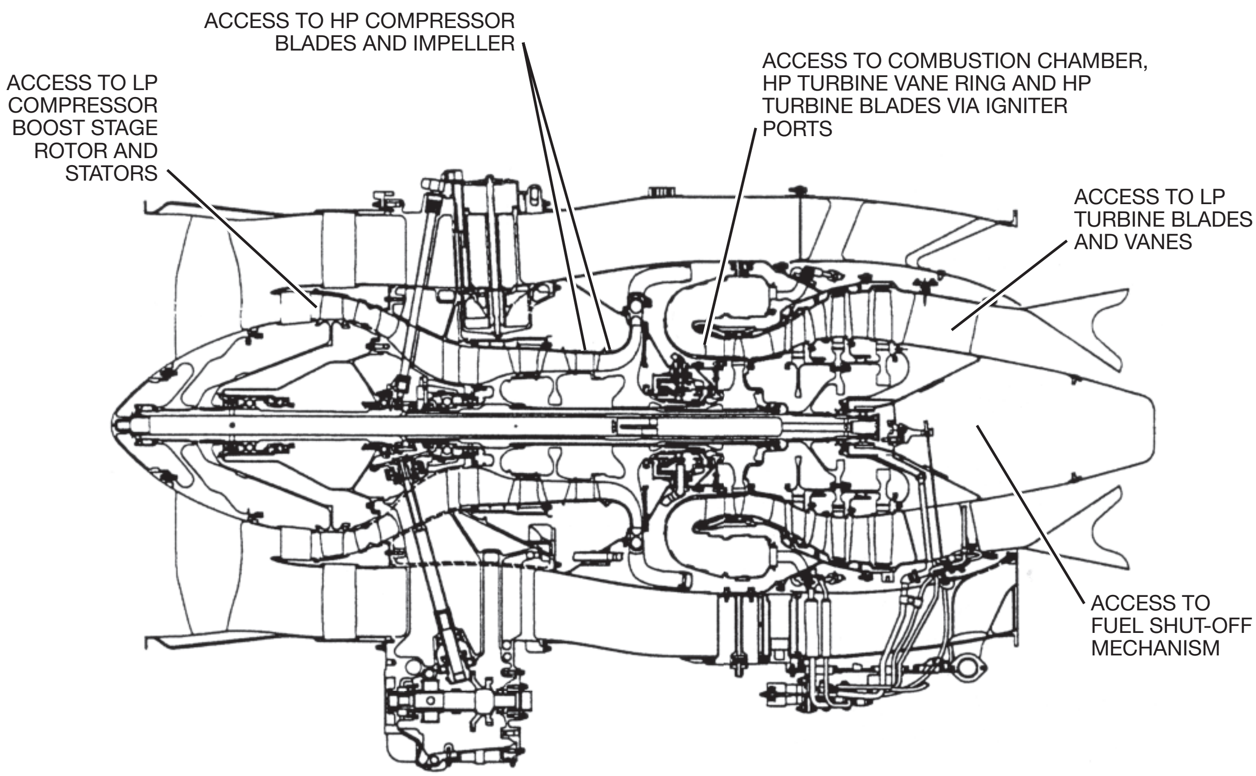
NOTE

Post SB engines use carbon front seals on the accessory drive gearshaft and alternator drive gearshaft.

The fuel control drive gearshaft and the hydraulic pump drive gearshaft are supported by roller bearings at the front and the rear. Plain seals (garter-type) are at the rear of the gearshafts. A ball bearing at the front and a roller bearing at the rear, support the alternator drive gearshaft. A plain seal (lip-type) is at the front of the gearshaft.

Lubricating oil under pressure for the accessory gearbox components is routed within the intermediate case and passes through integral passages in the housing and cover.

NOTES



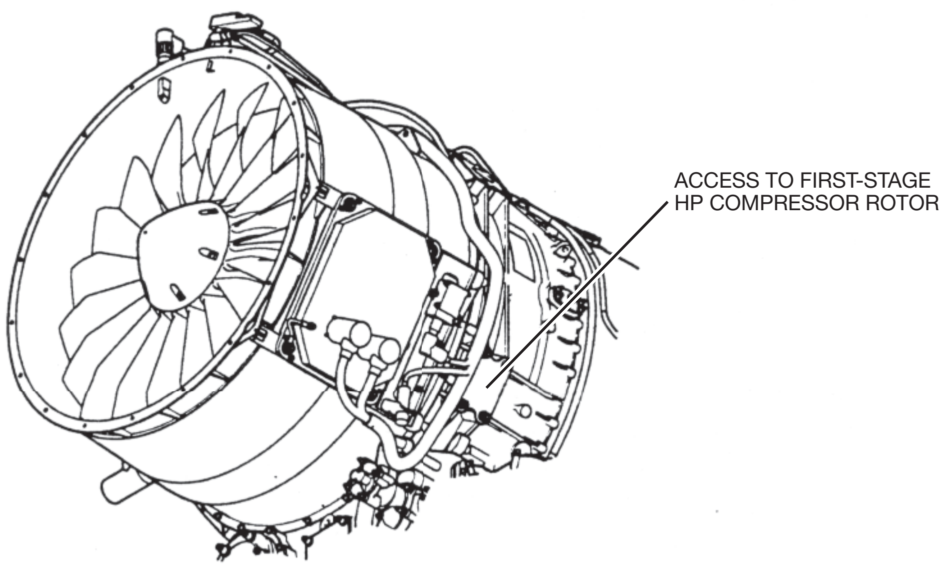


Figure 71-17. Borescope Access

DIAGNOSTICS

Borescope Inspection

The borescope is an optical device that enables an operator to perform visual inspections of the hot section and compressor areas of the engine while the engine remains on the aircraft with minimal disassembly.

A borescope inspection is to be performed at every three aircraft minor inspections (every 450 hours) pre-SB, or once a year if a performance run falls outside maximum limitations, whichever occurs first. The inspection includes (Figure 71-17):

- The LP and HP compressor and rotor sections
- Combustor
- Fuel nozzles

Access to these various components is achieved through different avenues. Table 71-1 lists the components that can be inspected through the different access ports.

NOTES

Table 71-2. BORESCOPE INSPECTION

ACCESS PORT	INSPECTION AREA
Fan Inlet	LP Compressor Boost Stage
Lower Engine Mount	First Star HP Compressor
Compressor Bleed Valve	Second Stage HP Compressor Impeller, HP Compressor
Igniter Boss	Fuel Nozzles Combustion Chamber Liner First-stage (HP) Turbine Vanes First-stage (HP Turbine Blades First-stage (HP) Turbine Liner Segments
Exhaust Case	Second-stage (LP) Turbine Vanes Second-stage (LP) Turbine Blades Third-stage (LP) Turbine Vanes Third-stage (LP) Turbine Blades Fourth-stage (LP) Turbine Vanes Fourth-stage (LP) Turbine Blades
Exhaust Case End Cone	Fuel Shut-off Mechanism

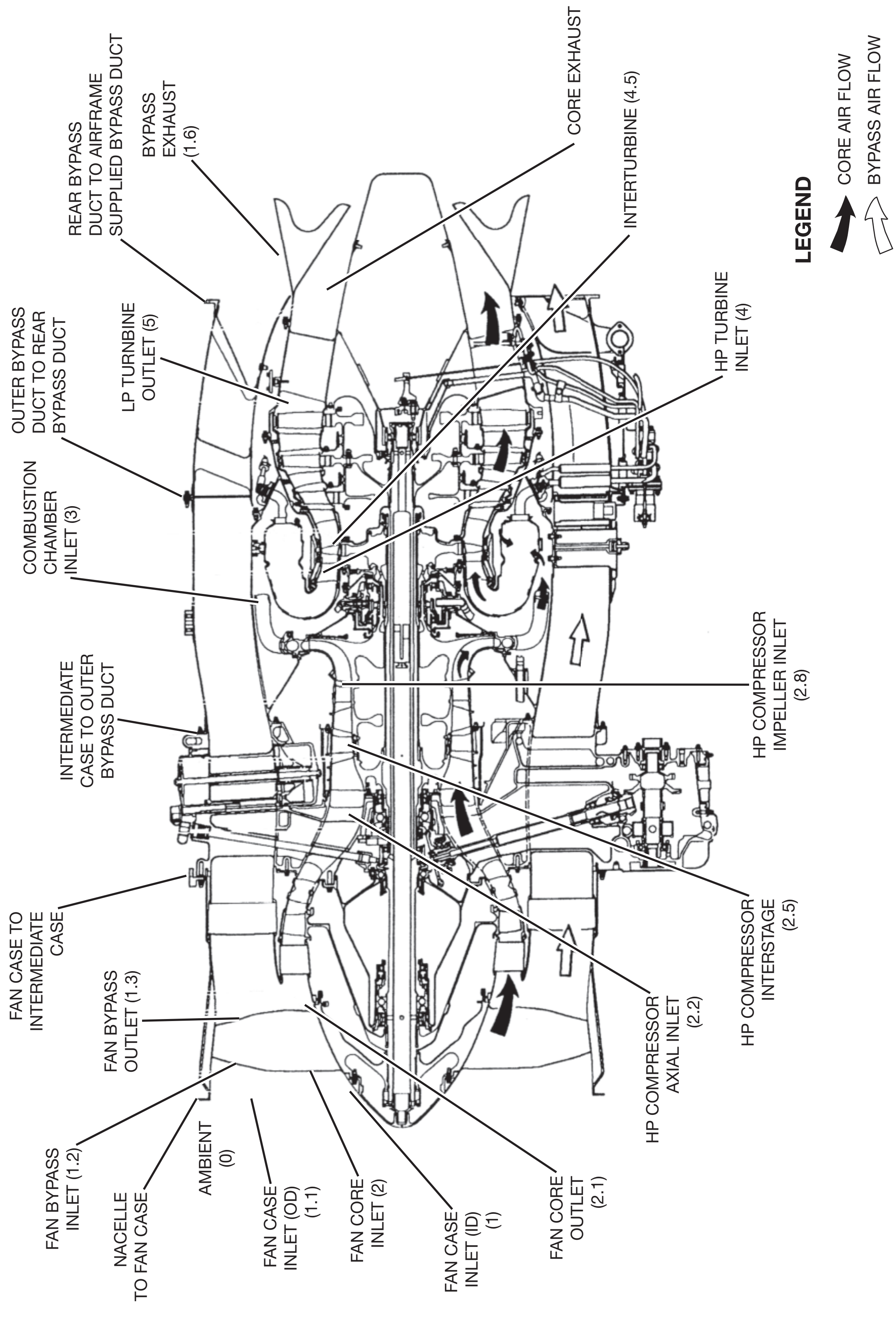


Figure 71-18. Secondary Air Systems

SYSTEMS

SECONDARY AIR SYSTEMS

The PW545A engine air system is supplied from the following sources:

- The high-pressure (HP) compressor-bleed cavity ($P_{2.8}$).
- The HP compressor impeller bleed.
- The HP compressor discharge (P_3).

$P_{2.8}$ Air System

Bleed air from the HP compressor ($P_{2.8}$) passes inboard across the gas path through three sets of passages in the intercompressor case to the intershaft region, adjacent to the No. 1, No. 2, and No. 3 bearing chambers (Figure 71-18). A small quantity of HP air goes through the two intershaft carbon seals into the bearing chamber. Some of the air goes into the interior of the fan shaft to:

- Anti-ice the nose cone
- Pressurize the carbon seals adjacent to the No. 5 bearing
- Cool the rear disk cavity of the first-stage LP turbine

The air that enters the bearing cavities then makes its way to the AGB, which provides for positive pressure on the oil. This prevents oil pump cavitation at altitude. Excess pressure is vented through the retimet-type breather, then overboard via the breather tube in the exhaust airflow.

The remainder of air from the intershaft region goes through the radial space between the shafts, pressurizing the carbon seals next to the No. 3 bearing and the front cavity of the first-stage HP compressor disk. This air then mixes with cooling air from the HP turbine; and passes through a seal between the HP turbine rear baffle and the fan shaft. From there, it goes into the front cavity of the first-stage LP turbine disk.

Impeller Bleed-Air System

Air is bled off the HP compressor impeller (through the impeller rear cavity) to the front seal of the No. 4 bearing buffer chamber. It then flows through passages around the No. 4 bearing buffer chamber to its rear seal.

Air around the No. 4 bearing buffer chamber passes through the front and rear labyrinth seals and into the No. 4 bearing chamber through carbon seals. The buffer chamber equalizes the pressures feeding the front and rear bearing seals. It prevents oil escape in the event that the seals increase their clearances during engine operation.

P_3 Air System

Air from the compressor discharge, surrounds and passes into the combustor to sustain combustion. This air is also used for:

- Cabin pressurization and temperature control
- Aircraft service air
- Aircraft/engine anti-icing and deicing
- HP turbine cooling and sealing

The air for HP turbine cooling passes through holes in the turbine case and into the HP turbine vane ring for cooling purposes. The air for HP turbine sealing goes through other holes in the turbine case, in order to prevent hot gas path air from getting into the HP turbine liner segments.

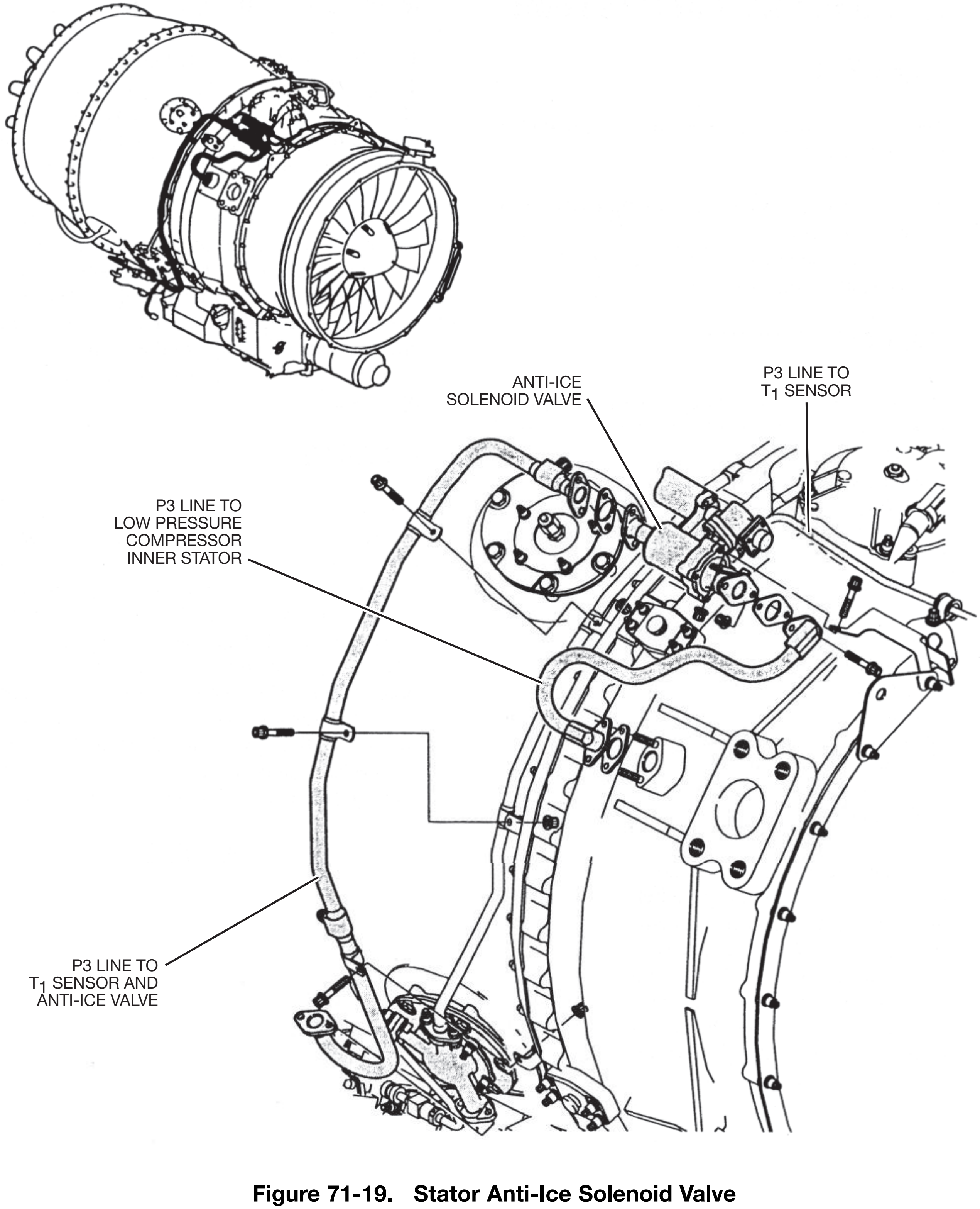


Figure 71-19. Stator Anti-Ice Solenoid Valve

ENGINE ANTI-ICING

Many areas of the PW545A are anti-iced to prevent ice ingestion into the engine, which could cause extensive damage. Some areas are continually anti-iced, while other areas are anti-iced by crew action (Figure 71-19).

Items that are anti-iced are as follows:

- Inlet cone
- TT0 temperature probe
- T1 temperature probe (XL/XLS)
- First and second stage stators

Inlet Cone

The engine inlet cone is continuously anti-iced anytime the engine is operating. Bleed air ($P_{2.8}$) is routed to the inlet cone through the LP shaft where a portion of air exits through the small hole in the cone center; while the remaining air impinges on the inside of the cone before exiting along the aft side of the inlet cone.

TT0 Temperature Probe

The fan inlet total temperature (TT0) temperature probe is anti-iced electrically anytime power is "on" and the aircraft anti-ice is selected "on." The TT0 probe provides the EEC with ambient temperature information. It is discussed in more detail, later in this chapter.

T_1 Temperature Probe (XL/XLS)

The T_1 temperature is continuously anti-iced with P_3 air anytime the engine is operating. The P_3 air originates at the P_3 services port and is routed externally to the engine. The T_1 probe senses inlet air temperature for ITT indication and incorporates a vane that prevents P_3 air from contacting the temperature sensors at the tip, which would influence the ITT indication.

First and Second Stage Stators

Stator anti-icing is accomplished by externally ducting P_3 air to the engine. The P_3 air originates at the P_3 services port and is routed through a stator valve before reentering the engine on its way to the first and second stage stators.

The first and second stage stators are in the core airflow between the fan and booster stage, and between the booster stage and the HP compressor (respectively). The stator valve is a solenoid-type valve that is energized "closed" and "opened" with upstream air pressure (and the removal of electrical power). The system is a fail-safe open design, so that, if aircraft electrical power were lost, anti-icing would continue on the engine stators.

NOTES

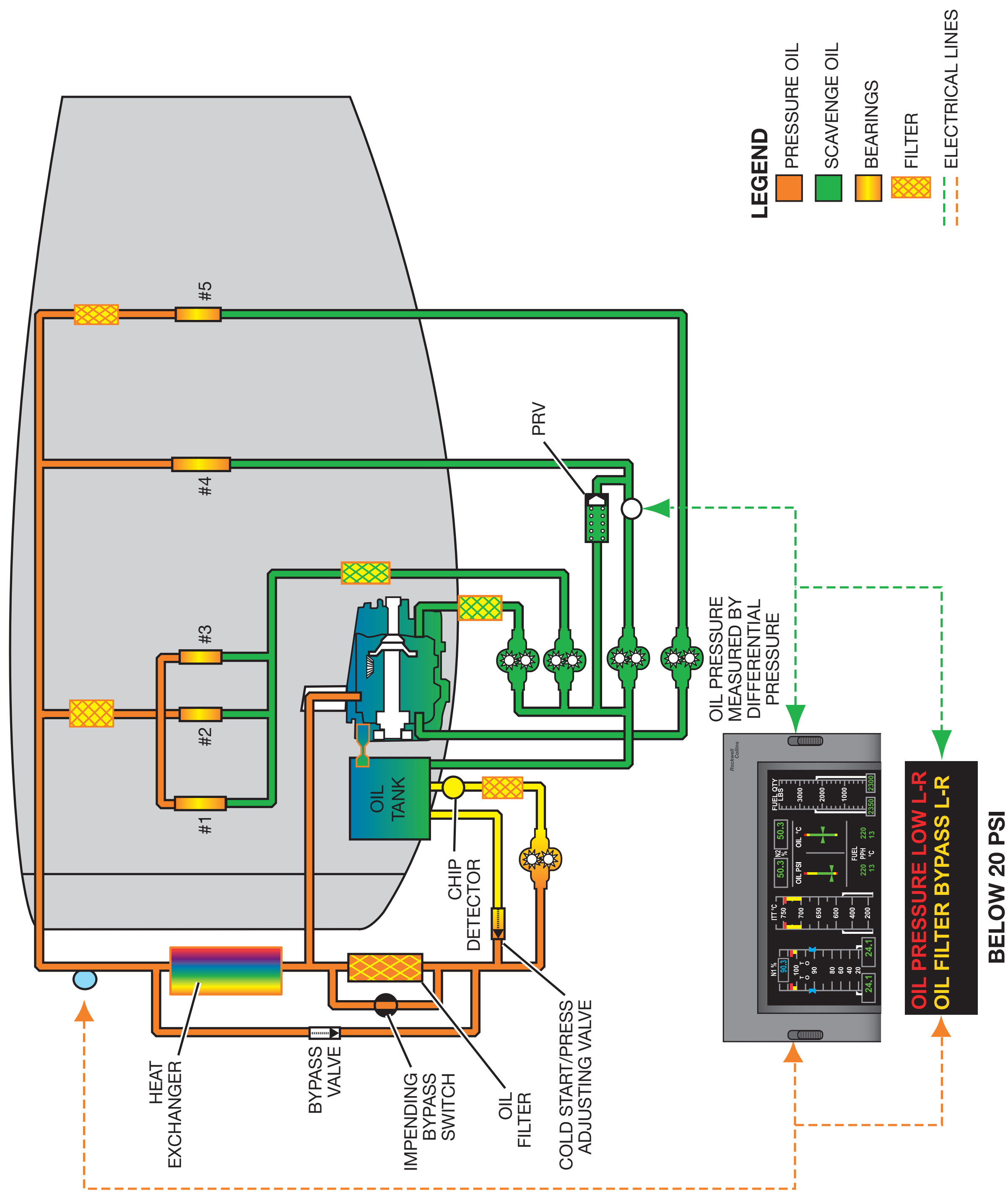


Figure 71-20. Engine Oil System

ENGINE OIL SYSTEM

Description

The engine lubrication system consists of a pressure and scavenge system, as well as a secondary air system (Figure 71-20).

The pressure system is a flow-regulated system that supplies oil, to satisfy the lubricating requirements throughout the engine operating range. Oil is supplied from the integral oil tank, past a magnetic chip collector, and to the pressure pump. It circulates through the engine oil filter and fuel/oil heat exchanger before being distributed throughout the engine. Calibrated oil nozzles deliver the necessary oil quantity to various bearings, gear meshes and splines.

The scavenge system returns oil to the tank directly, by means of a combination of blow down and dedicated pumps; or indirectly via the accessory gearbox dedicated pump.

The secondary air system uses compressor air to pressurize the various bearing cavity seals. The air/oil mixture from the bearing cavities returns to the tank and the vented air returns to the accessory gearbox (AGB). The air mixed with the oil in the AGB is separated by an air/oil separator, which vents to the engine exhaust duct.

NOTES

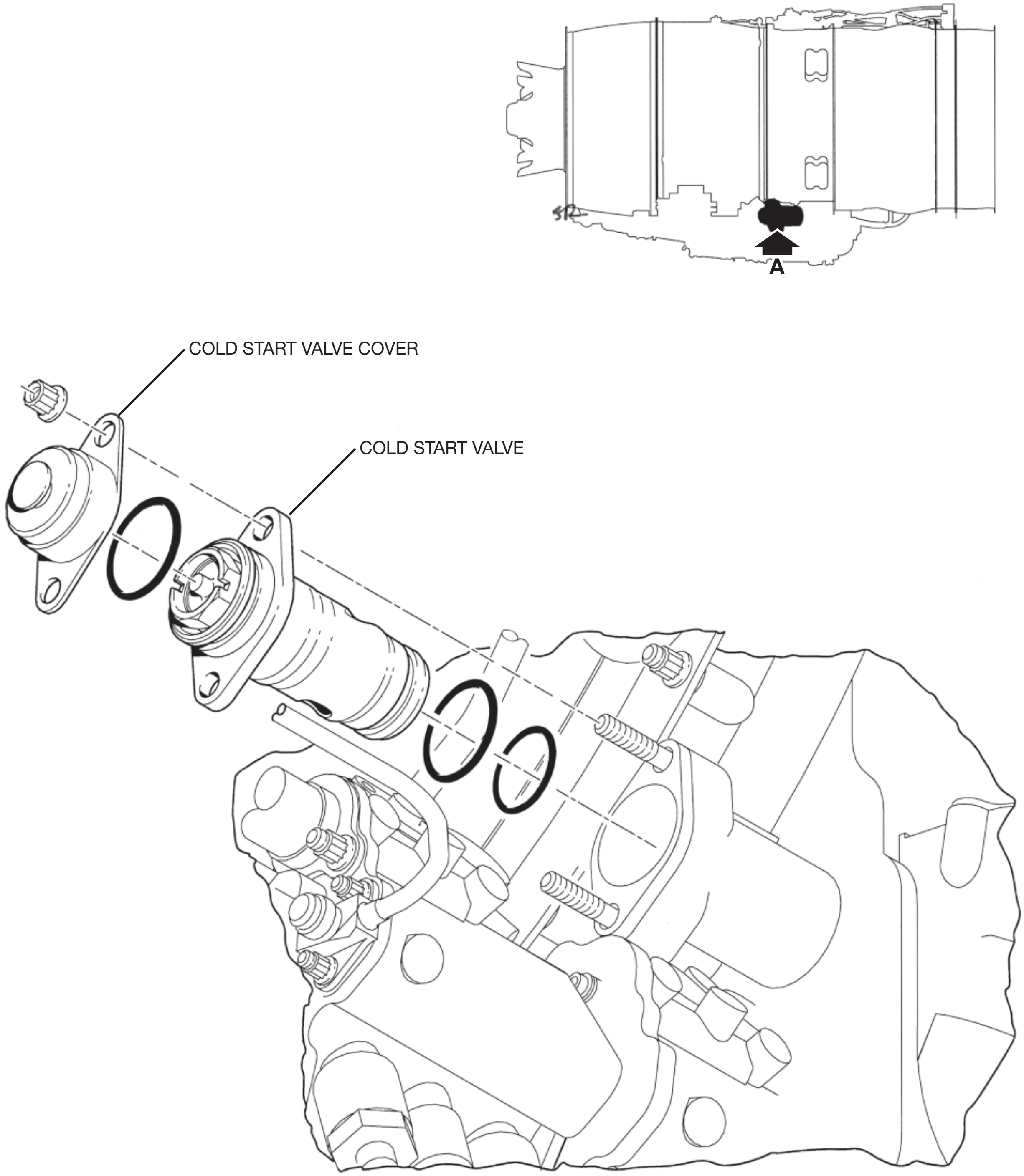


Figure 71-21. Cold Start Valve/Pressure Adjusting Valve

Pressure Oil System

The oil tank is integral with the intermediate case and is comprised of a main tank (on the right side of the engine), interconnected to a smaller auxiliary tank (on the left side by a tank-to-tank cored passage). Both tanks are equipped with sight glasses. The fill provision is only supplied when relevant to satisfy installation purposes. When not required, a blanking plug is installed. A drain plug is fitted on the main oil tank to permit drainage of both tanks.

Oil supplied from the tank flows past a magnetic chip collector, strategically placed on an elbow for ease of maintenance and maximum catch efficiency. There is a protective screen positioned between the chip collector and the high speed, gear-type oil pump.

Beginning at the oil pump, oil flows through a wrap-around line (integral with the intermediate case) to the combined cold start/pressure adjusting valve (Figure 71-21). At this point surplus pressure is bled off (via the valve) and returns to the tank, while engine flow oil passes through the oil filter then on to the fuel/oil heat exchanger.

The cold-start valve provides a safeguard against excessive pressure build-up due to high oil viscosity in cold weather operation. The cold-start valve opens when the pressure drop (across the valve) reaches 250 psid and diverts oil to the main oil tank.

The main oil filter is in the intermediate case near the gearbox and is retained in the housing by a cover. For servicing purposes, an oil drain plug is available to drain oil from the filter housing, fuel/oil heat exchanger, and anti-siphon line. A built-in bypass valve bypasses both the oil filter and the fuel/oil heat exchanger when the pressure differential at the oil filter becomes 99 psid or greater. An impending bypass switch (set at 28 psid) provides the annunciation of OIL FLTR BP L or R prior to bypass valve actuation.

NOTES

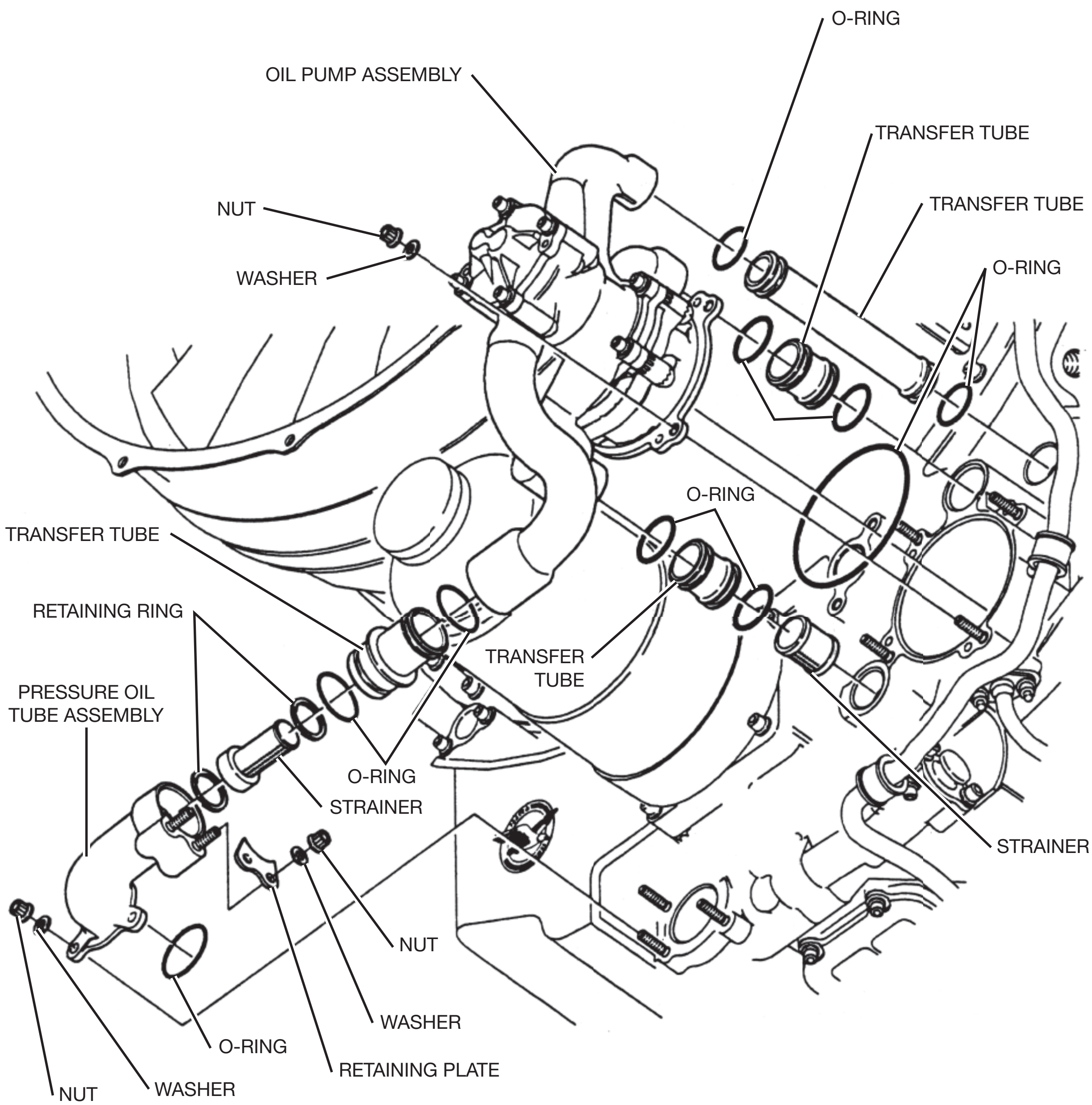


Figure 71-22. Oil Pump Assembly

SCAVENGE OIL SYSTEM

The oil supplied to the main shaft bearings, gears and splines is returned to the oil tank either directly (by a combination of blowdown and dedicated pumps) or indirectly (via the accessory gearbox scavenge pump) (Figure 71-22).

Oil that feeds the Nos. 1, 2, and 3 bearings, the upper towershaft gear mesh, and support bearing is scavenged by a dedicated (high speed) gear pump that is protected by an inlet screen.

The No. 4 bearing oil flow is scavenged by a combination of scavenge and blowdown. At low engine speeds the pressure inside the cavity is insufficient to ensure proper blowdown operation. Therefore, a dedicated scavenge pump is provided. At high engine speeds, there is enough pressure for the system to operate without assistance. A bypass valve is provided on the scavenge pump inlet to reduce the flow restriction created by the pump elements when the system is in a "blowdown" mode. Both the pump and blowdown-path oil combine together and mix with the Nos. 1, 2, and 3 bearing scavenge oil. All the oil then flows together within the pump housing and returns to the tank.

The No. 5 bearing flow is scavenged by a dedicated pump, which returns the oil to the accessory gearbox. This oil/air mixture plus the oil fed directly to the accessory gearbox (AGB) gears, bearings, and splines is scavenged from the AGB by a dedicated pump (protected at its inlet by a screen). As with the other scavenge pump outlets, this flow is combined, within the pump housing, and returns to the main oil tank.

The oil pump drive train is arranged so that the main pressure pump elements do not need to be driven if any of the scavenge pumps should fail. This prevents oil from flooding the engine core if a scavenge pump becomes inoperative.

Breather System

Air from the various bearing compartment seals makes its way to the main oil tank, then on to the accessory gearbox by way of a restrictor. The restrictor is sized to provide adequate tank pressure at altitude, to avoid cavitation of the pressure pump. The air is then vented to the engine exhaust through an oil/air separator on the starter/generator gearshaft in the gearbox.

NOTES

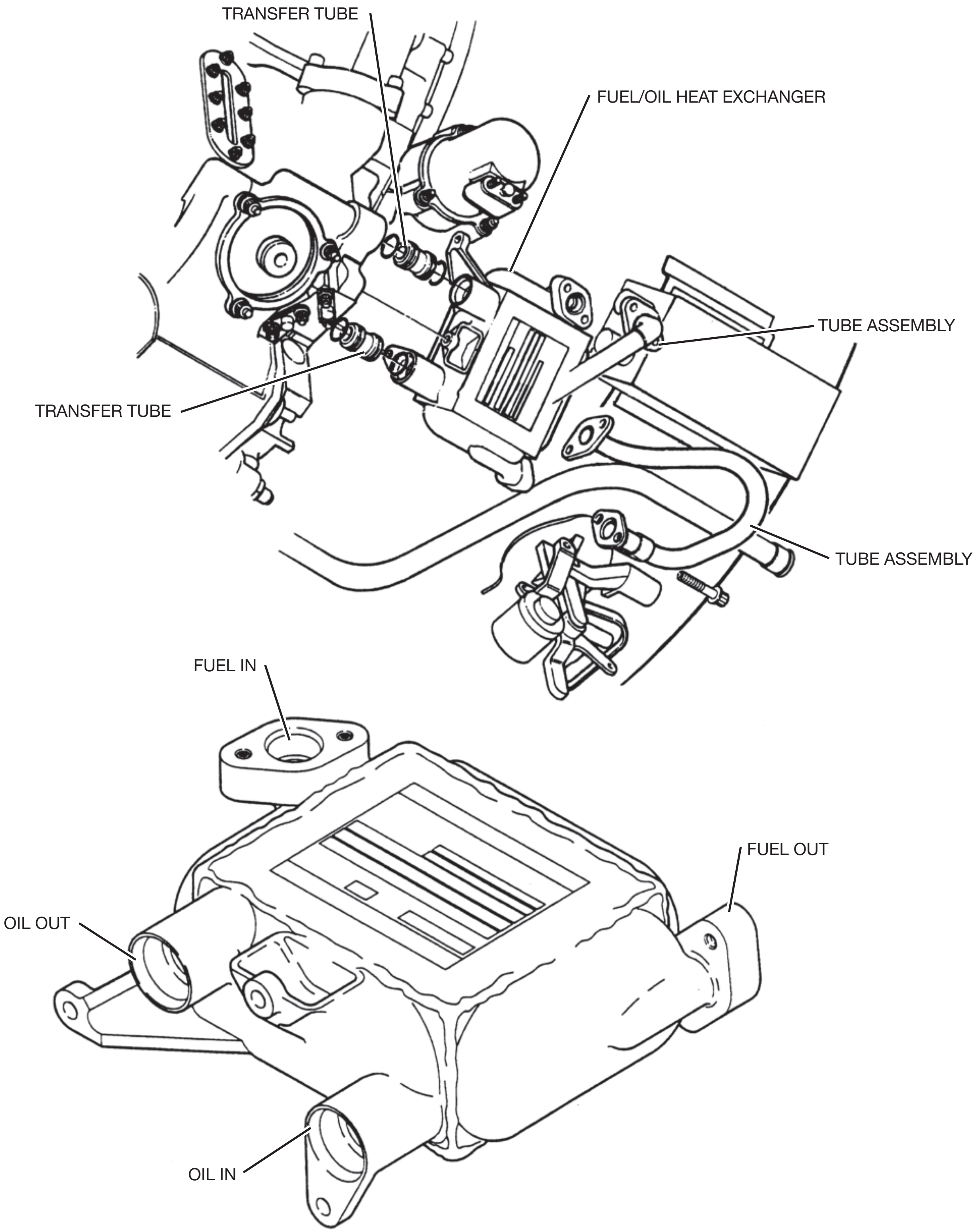


Figure 71-23. Fuel/Oil Heat Exchanger

Components

Fuel/Oil Heat Exchanger

The fuel/oil heat exchanger is on the oil pressure manifold, on the left side of the accessory gearbox (Figure 71-23). The fuel filter, fuel and oil filter impending bypass switches are also on the oil pressure manifold. The fuel and oil filter bypass valves are contained within the manifold.

Engine lubricating oil enters the heat exchanger from the oil pressure manifold and is directed by baffles to flow around the tubes through which fuel flows. Heat transfer (through the tube walls) raises the fuel temperature before it enters the fuel filter assembly. The fuel/oil heat exchanger also serves to cool the engine oil before distribution to the engine bearings and accessory gearbox.

The oil from the fuel/oil heat exchanger flows to a chamber between the AGB and manifold. From there, it is distributed through a cored passage around the intermediate case to feed the Nos. 1, 2, 3, and 4 bearings, plus the tower shaft bearings and gears. A separate external line from the manifold supplies oil to the No. 5 bearing and the accessory gearbox. All main shaft bearings are either side jetted or underrace lubricated by calibrated oil nozzles which, in turn, are protected by last chance screens.

The transfer tube between the fuel/oil heat exchanger oil outlet and the manifold is a restrictor. This ensures that oil pressure within the heat exchanger is always greater than the fuel pressure. It also ensures that, in the event of an internal failure, oil leaks into fuel as opposed to fuel into oil.

The fuel/oil heat exchanger body and matrix are made of aluminum. The whole assembly is brazed and welded into an integral unit. All external surfaces are anodized, coated with primer and finished with epoxy enamel, except fuel and oil mounting flanges.

NOTES

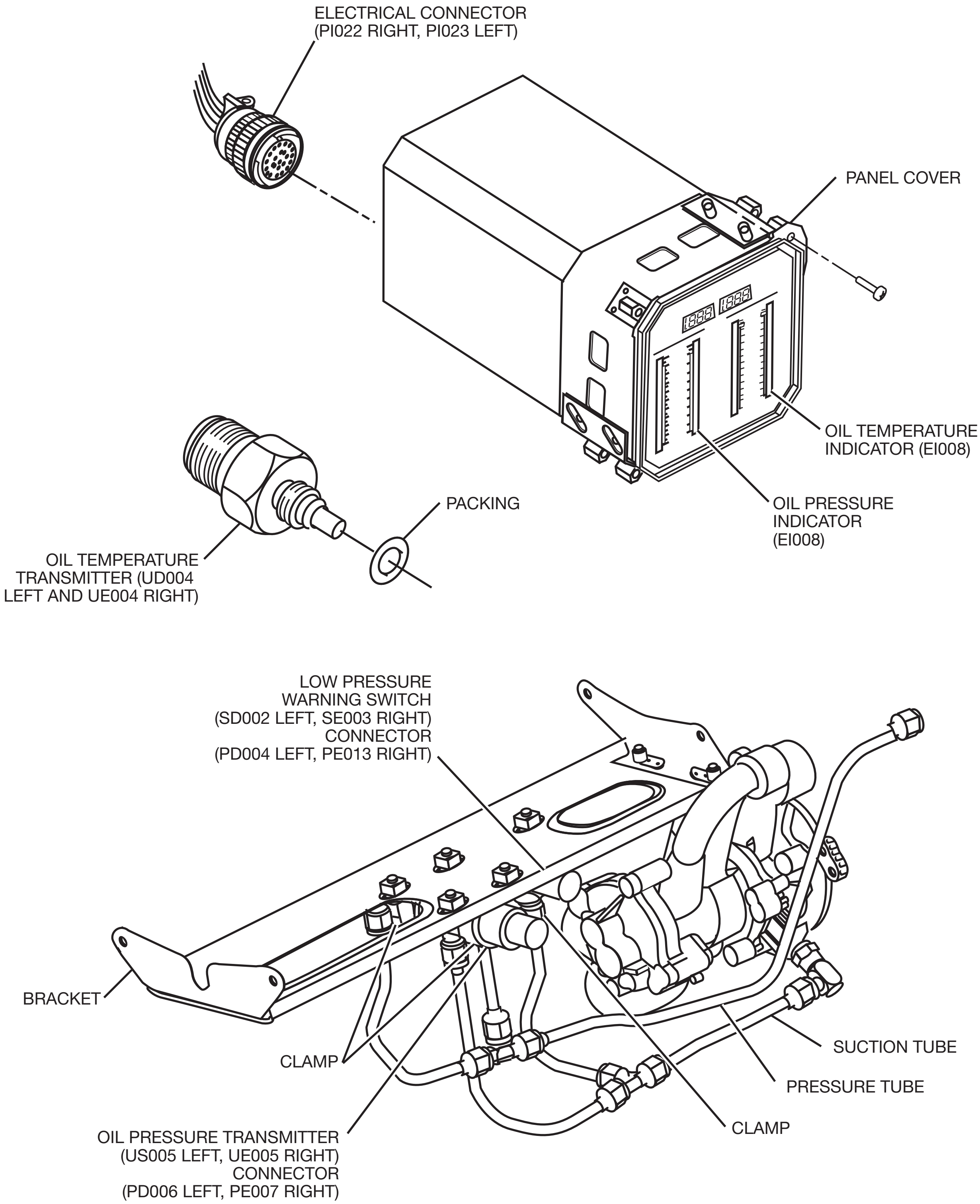


Figure 71-24. Engine Oil Indicating

Controls And Indications

Oil pressure, low oil pressure warning, and oil temperature indicating systems display engine oil conditions on the center instrument panel (Figure 71-24).

Oil Pressure Indicating

The engine oil pressure indicating system consists of an oil pressure dual indicator on the center instrument panel and an oil pressure transmitter on a bracket (just above the starter-generator and on the left side of the engine).

The oil pressure indicator is a dual vertical scale indicator incorporating two independent systems. One system indicates left engine oil pressure, and one system indicates the right engine oil pressure. The indicator is capable of indicating oil pressure from 0 to 260 psi.

The oil pressure transmitter is a variable resistance output-type transmitter that requires no adjustment. The oil pressure transmitter connects between the engine oil pressure line and the No. 4 bearing scavenge line. The transmitter senses the differential pressure between the two. The airplane oil pressure indicating wiring connects to the transmitter with an electrical connector.

Low Oil Pressure Warning System

The low oil pressure switch monitors differential oil pressure between the pressure line and the No. 4 scavenge line on the engine. When the differential oil pressure drops below 20 psid the electrical circuit is closed and the LO OIL PRESS L or R light (in the annunciator panel) illuminates. The pressure switch reopens at an increasing oil pressure of 25 psid.

Oil Temperature Indicating

The engine oil temperature indicator is a dual-vertical scale indicator, incorporating two independent systems: one system for the left engine and one for the right . The oil temperature indicator indicates oil temperatures from 0°C to 140°C.

The engine oil temperature transmitters are resistance-type bulbs consisting of an element sealed in an insulating material (within a housing). The housing has an electrical connector and external mounting threads that mate with a boss connection on the left side of the engine (just in front of the fuel filter).

NOTES

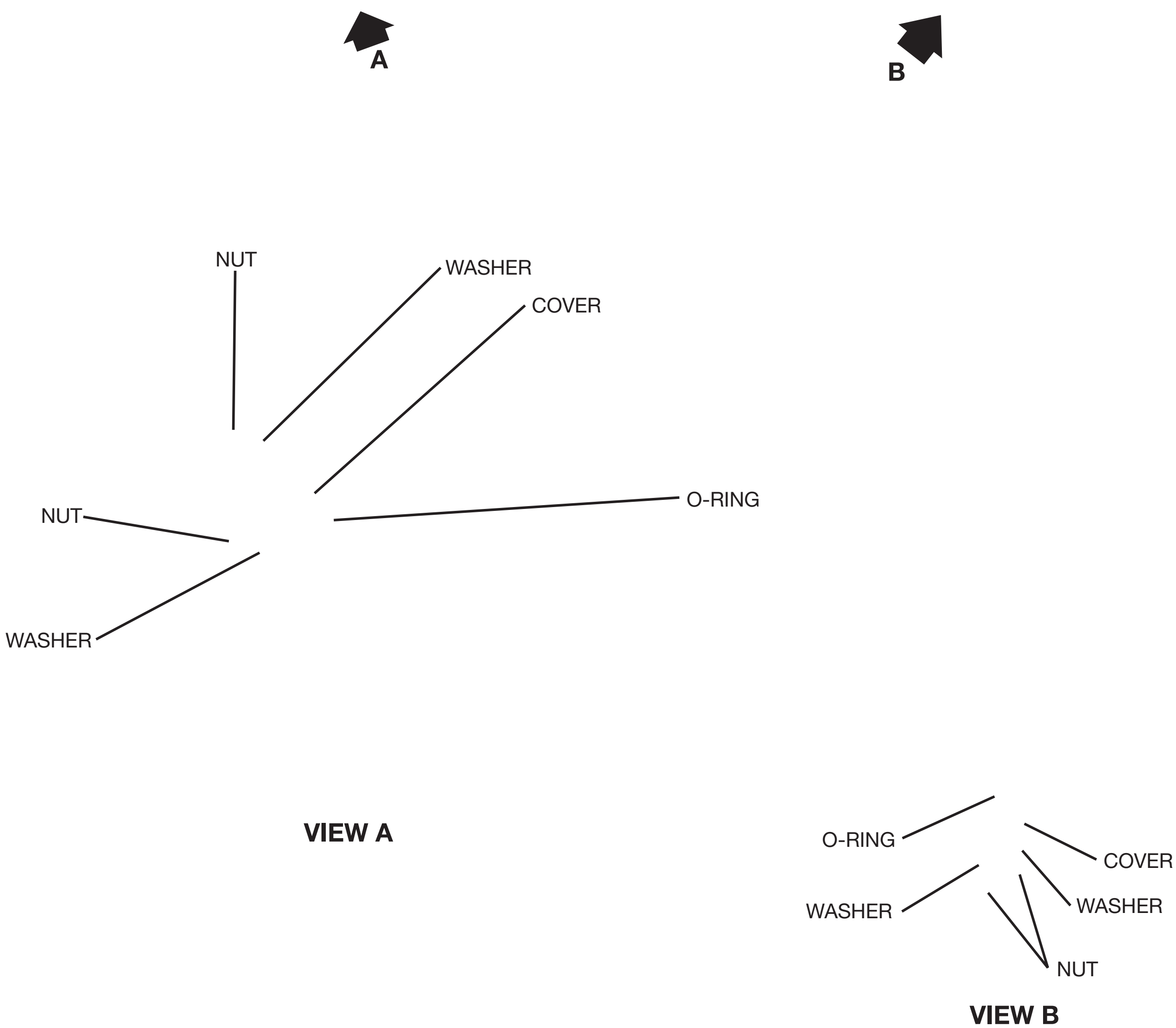


Figure 71-25. Oil System Draining

Diagnostics

Engine Oil Servicing

Servicing the engine oil system consists of initial filling after engine installation, normal servicing (adding oil), and draining the system. Servicing the left and right engines is typical.

Approved engine oils are listed in the "Limitations" section of this chapter.

An oil filler cap is provided (for oil servicing) on the outboard side of each engine. The oil filler cap is accessed through the oil access door, on the lower engine cowling.

Draining Engine Oil

NOTE

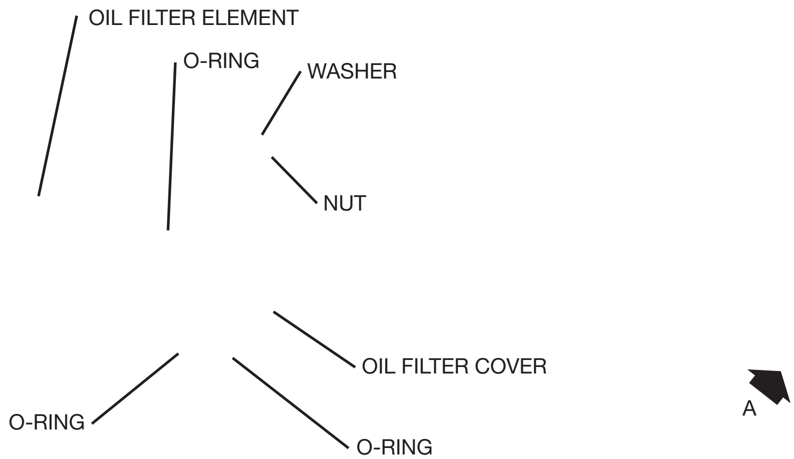
Drain the oil as soon as practical after engine shutdown.

WARNING

Persons who handle engine oil are advised to minimize skin contact with used oil, and promptly remove any used oil from their skin. A laboratory study, while not conclusive, found substances which cause cancer in humans. Thoroughly wash used oil off skin as soon as possible with soap and water. Do not use kerosene, thinners or solvents to remove used engine oil. If waterless hand cleaner is used, always apply skin cream afterwards.

1. Open the lower engine cowling.
2. Open the oil access door and remove the filler cap (Figure 71-25).

NOTES



SECTION THOUGH INSTALLED FILTER

DETAIL A

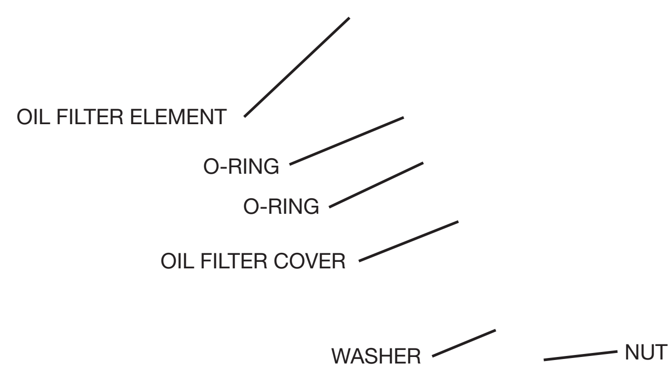


Figure 71-26. Oil Filter Element

3. Position catch-container(s) under the oil tank drain cover and oil filter drain cover (Figure 71-26).
4. Remove the nuts and washers securing the oil tank drain cover. Remove drain cover using puller (PWC66103).
5. Remove the nuts and washers that secure the oil filter drain cover. Remove the drain cover using puller (PWC66103).
6. Allow the system to drain completely.
7. Install drain covers with new packings (O-rings) and secure with washers and nuts. Torque nuts: 23 to 26 inch-pounds (2.6 to 2.9 Nm).
8. Replace the oil filter element if necessary.

Replenishing Engine Oil

1. Remove the reservoir oil filler cap.
2. With the proper oil, fill reservoir to the level on the sight glass. Install filler cap.
3. Start engine and operate at idle for 15 minutes.
4. Shut down engine and check oil level 10 minutes after shutdown.
5. Check reservoir sight glass oil level. Top off reservoir to the required level.

NOTE

If oil of the same brand (as used in the tank) is unavailable, then other approved oils may be intermixed, if the total quantity added does not exceed 2 quarts in any 400-hour period.

If more than 2 quarts of dissimilar oil brands have been intermixed in any 400-hour period, or if an unapproved brand; or a different viscosity oil is added, drain and flush the oil system. Refer to the *AMM*.

NOTES

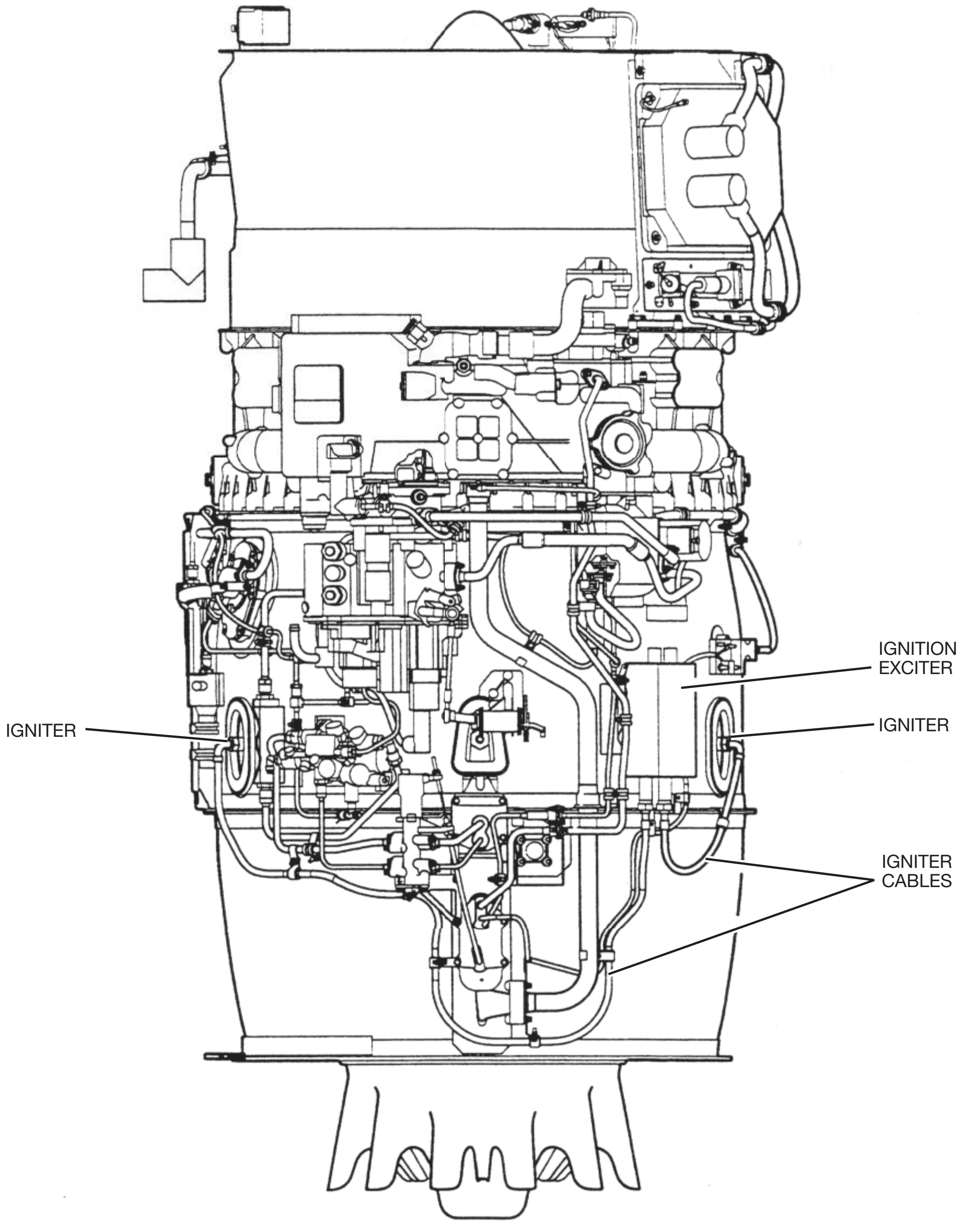


Figure 71-27. Ignition System Components

IGNITION SYSTEM

Description

The engine ignition system consists of (Figure 71-27):

- The engine ignition switching components
- The ignition unit
- Igniter leads
- Igniters
- Cockpit indication

Components

Ignition Switching

The engine ignition system is controlled by an ignition switch, throttle lever cutoff switch, anti-iceldeice switch (aircraft 5001 through 6000), and a engine starting relay. An ignition system is provided for each engine and each is equipped with one ignition exciter and two igniters. Refer to the applicable *Wiring Diagram Manual* for system electrical schematics.

The ignition switches (left and right) are on the pilot switch panel. Each ignition switch provides the following selections:

In the ON position a continuous 28 volts direct current (VDC) is supplied from the aft power junction box main bus to its respective exciters.

In the NORM (normal) position the electrical supply is interrupted to its respective exciter unless START or WING/ENGINE ANTI-ICE is on.

In the SEC (secondary) position (XL/XLS) a continuous 28 volts direct current (VDC) is supplied from the left circuit breaker panel emergency bus to its respective exciter.

Two throttle lever cutoff switches are located in the control quadrant. Each switch is actuated by its respective throttle lever. The throttle lever switch position is adjustable. The switch provides a means to inhibit ignition electrical supply until the throttle lever is moved out of cutoff, during start mode.

Left and right ANTI-ICE/DEICE switches are located on the pilot's switch panel. Each switch provides ignition switch control circuit for its respective ignition circuit. Ignition electrical power is supplied to each engine exciter when the respective engine anti-iceldeice switch is in WING/ENGINE ON or ENGINE ON position (XL/XLS).

The left and right engine starting relays are located in the aft electrical junction box. Each engine starting relay provides alternate ignition power to its respective exciters in the start mode with the throttles out of the cutoff position. Each relay, when actuated, provides power to illuminate its respective ignition light. Power must be present at BOTH circuit breakers feeding the exciters or the light will not be illuminated.

Ignition Unit

The ignition unit (on the left side of the fan bypass housing) is a radio-noise-suppressed, high-voltage, capacitor-discharge, burst-mode-type unit made by Unison. The unit utilizes 10–32 VDC input (airplane supply of 24–32 VDC drops to about 10 VDC during start). This provides separate and independent secondary outputs of 18,000 to 24,000 volts at 0.5 to 1.0 amps of power to the igniters. The system is capable of continuous operation when selected as follows:

- The ignition switch at ON or SEC (XL/XLS)
- The engine anti-ice/deice switch at WING/ENGINE ON or ENGINE ON positions (XL/XLS)

The unit produces approximately six to seven sparks per second for the first thirty seconds, then falls back to one per second for the rest of the time that ignition is activated.

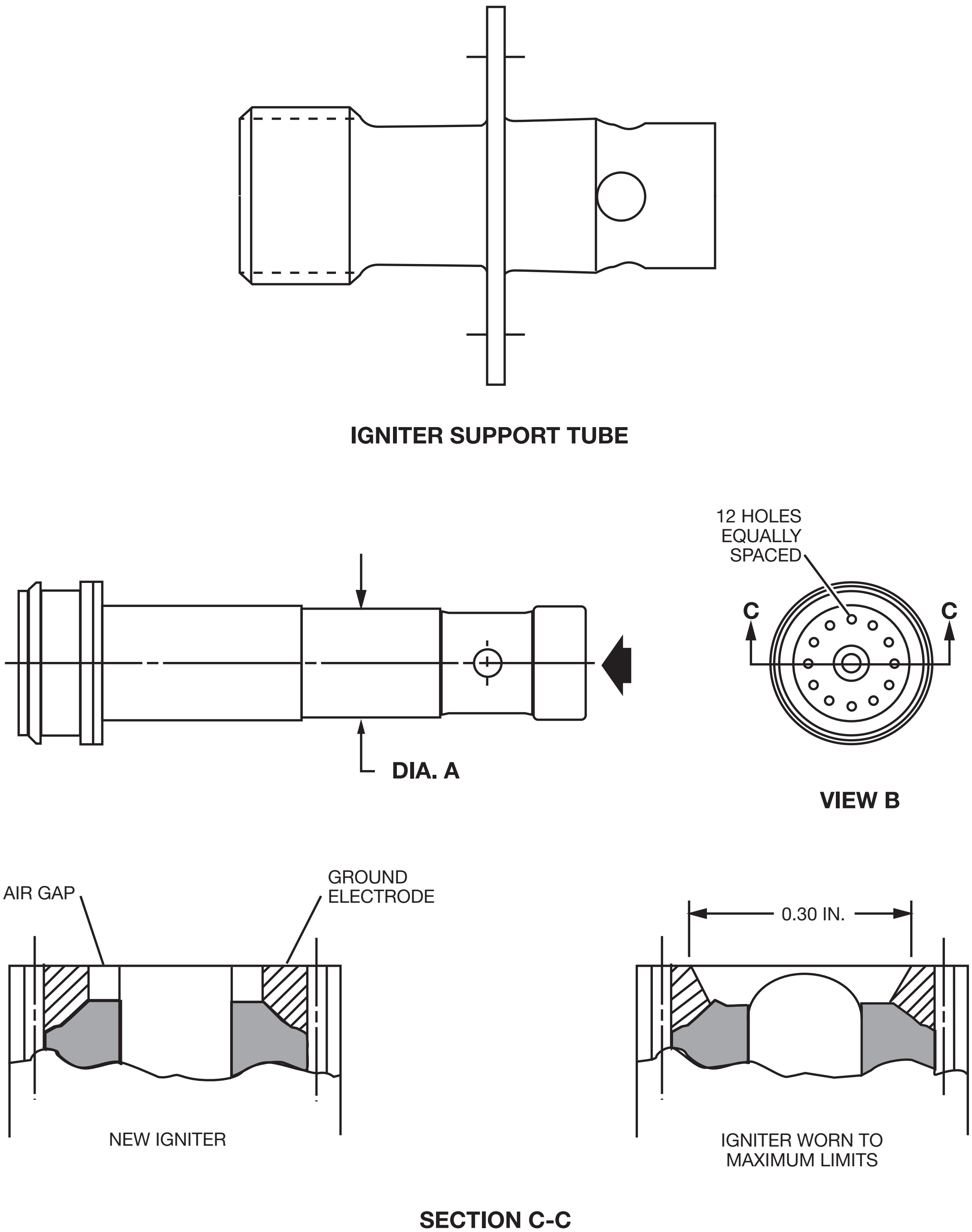


Figure 71-28. Igniter and Wear Limits

Igniters and Leads

The igniters are at 5 and 7 o'clock on the turbine plenum. The igniter leads are shielded, high-tension flexible leads connecting the ignition unit and igniters. Two leads are used, one for each igniter Figure 71-28).

WARNING

Due to system lethal voltage, wait six minutes (or more) after switching ignition off before handling any ignition components.

CAUTION

If a spark igniter is dropped, internal damage (possibly not detectable by testing) can occur. The recommendation is to replace the spark igniter.

Controls and Indications

Ignition Indicating

Green ignition lights on the engine cluster gauge next to the ITT indicating tapes illuminate when 28 VDC is supplied to both circuits of the ignition unit.

NOTES

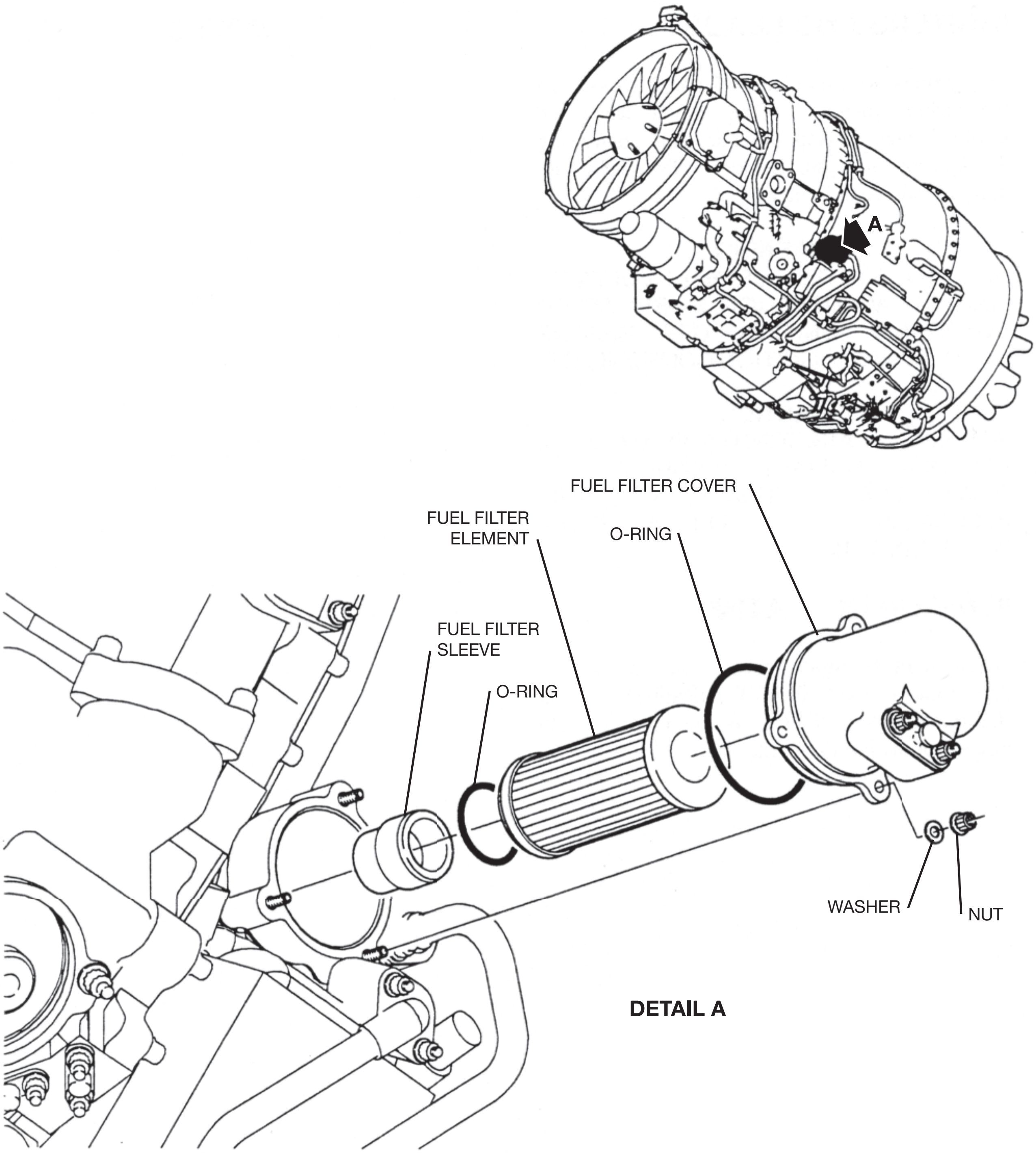


Figure 71-29. Fuel Filter and Housing Assembly

ENGINE FUEL SYSTEM

Description

The engine-mounted fuel control system on aircraft 5001 through 6000 includes the fuel control unit, fuel pump, and fuel flow divider valve. The fuel control unit, attached to the rear of the fuel pump, is driven by the fuel pump shaft. The engine-driven fuel pump is mounted on the rear face of the accessory gearbox. The fuel flow divider valve directs primary and secondary fuel flows to the fuel nozzles for combustion.

The engine-mounted fuel control system on aircraft 6001 and subsequent includes the fuel metering unit and fuel pump. The fuel metering unit, attached to the rear of the fuel pump, is driven by the fuel pump shaft. The engine-driven fuel pump is installed on the rear face of the accessory gearbox. This system uses a dual channel, Full Authority Digital Electronic Control (FADEC) to control the fuel metering unit. The FADEC uses Thrust Lever Angle (TLA) sensors in the throttle quadrant to find the TLA. The FADEC uses the data from the TLA sensors and detents in the throttle quadrant, to set the necessary fuel flow and engine thrust.

Fuel from the aircraft fuel system is pumped through the engine fuel system by the fuel pump, filtered, metered, and then delivered to the combustion chamber by the fuel manifold and atomized for efficient combustion by the fuel nozzles. The fuel control function is accomplished by the engine mounted fuel control unit or fuel metering unit.

Components

Engine Driven Fuel Pump

The engine driven, two-stage fuel pump is integral with the FCU. One stage is a regenerative-boost type and the other is a high-pressure gear type. The regenerative boost stage supplies unmetered fuel to the fuel/oil heat exchanger (FOHE), to the fuel filter, then to the high-pressure pump in the FCU. Fuel pressure at the outlet of the regenerative boost stage pump is approximately 60 to 100 psi greater than aircraft supply pressure. The high pressure gear type pump further increases the pressure to approximately 1,250 psi.

Fuel Filter

The fuel filter is a non-cleanable element that must be inspected every 150 hrs and replaced at every 900 ± 50 hours or every two years, (whichever comes first) (Figure 71-29). A fuel filter impending bypass switch is provided to illuminate FUEL FILTER BP L or R on the annunciator panel (XL/XLS) or FUEL FILTER BYPASS L-R CAS message (XLS+). Annunciation occurs if the differential pressure across the filter reaches 22 psid (XL/XLS) or 14 psid (XLS+). The fuel begins bypassing the filter as 28 psid (XL/XLS) or 26 psid (XLS+).

NOTES

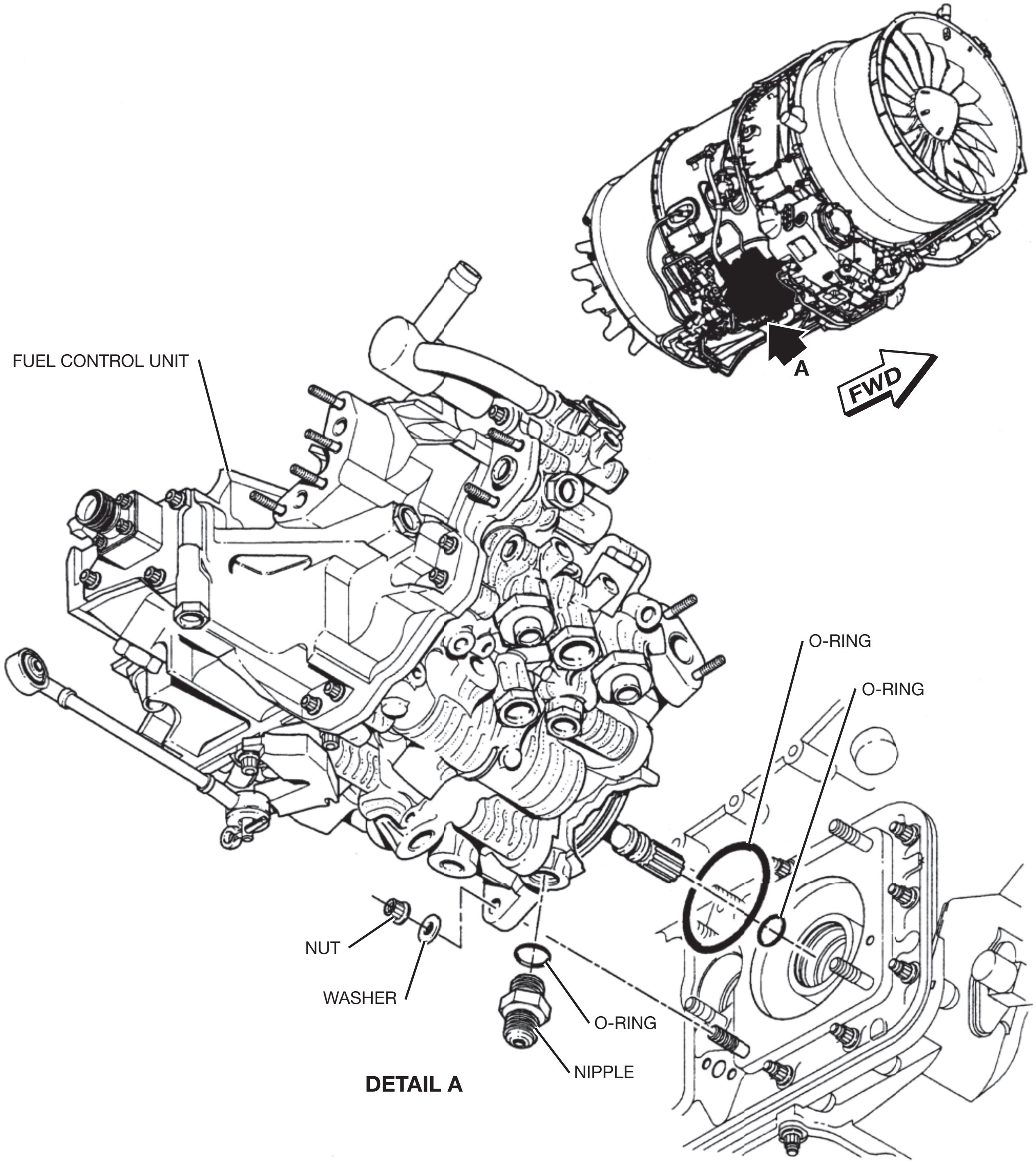


Figure 71-30. Fuel Control Unit

Fuel Control Unit (XL/XLS)

The FCU is a hydro-mechanical unit that controls the fuel supply to the engine (Figure 71-30). The FCU, when in MANUAL mode, computes and schedules fuel flow to the engine as a function of:

- Throttle lever angle (TLA)
- Fan inlet total temperature (T1/FCU)
- Compressor discharge servo pressure (P_3)

The FCU also provides high-pressure relief, ratio-unit adjustment for accel/decel fuel scheduling, and for fuel cut-off. The FCU also provides motive-flow fuel pressure, for aircraft ejector pump operation.

There is a torque motor on the FCU which receives scheduling information from the electronic engine control (EEC), anytime that the engine operates in AUTO mode. In AUTO, the EEC performs fuel scheduling.

FCU Adjustments

The only two adjustments which can be performed in the field are: the flight idle adjustment and the ratio unit adjustment for accel/decel speeds.

NOTES

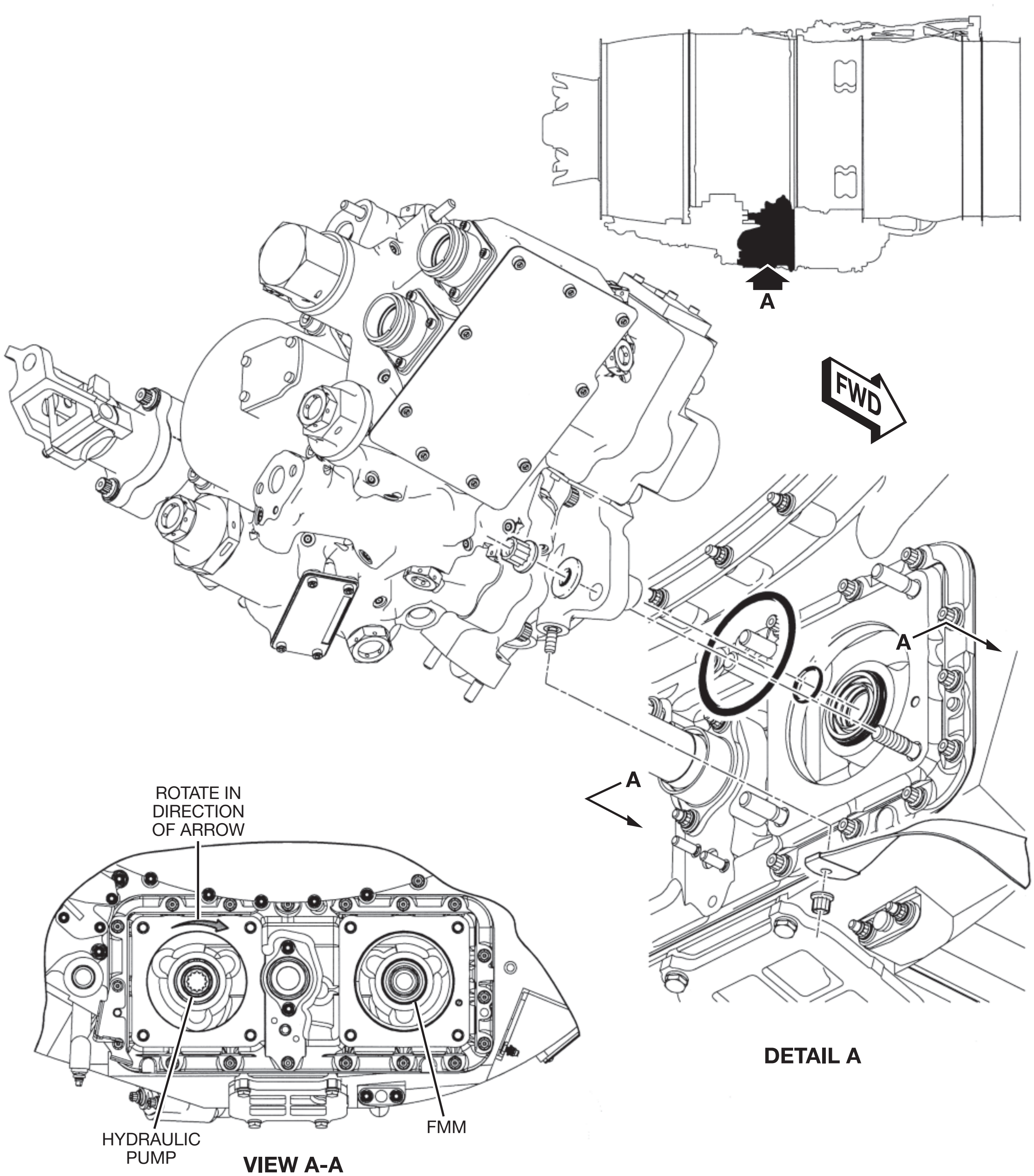


Figure 71-31. Fuel Metering Unit

Fuel Metering Unit (FMU) (XLS+)

The FMU performs the following major functions:

- Pressurization of fuel supply
- Regulation of burn fuel flow
- Division of primary and secondary flow
- Engine shutdown in normal and shaft shear circumstances
- Supply of motive fuel flow for airframe usage
- Prevention of discharge of fuel after engine shutdown

The FMU contains six major elements: the fuel pump, the permanent magnet alternator, the fuel metering system, manifold equalization system, motive flow system, the ecology system, and shaft shear protection valve (Figure 71-31).

NOTES

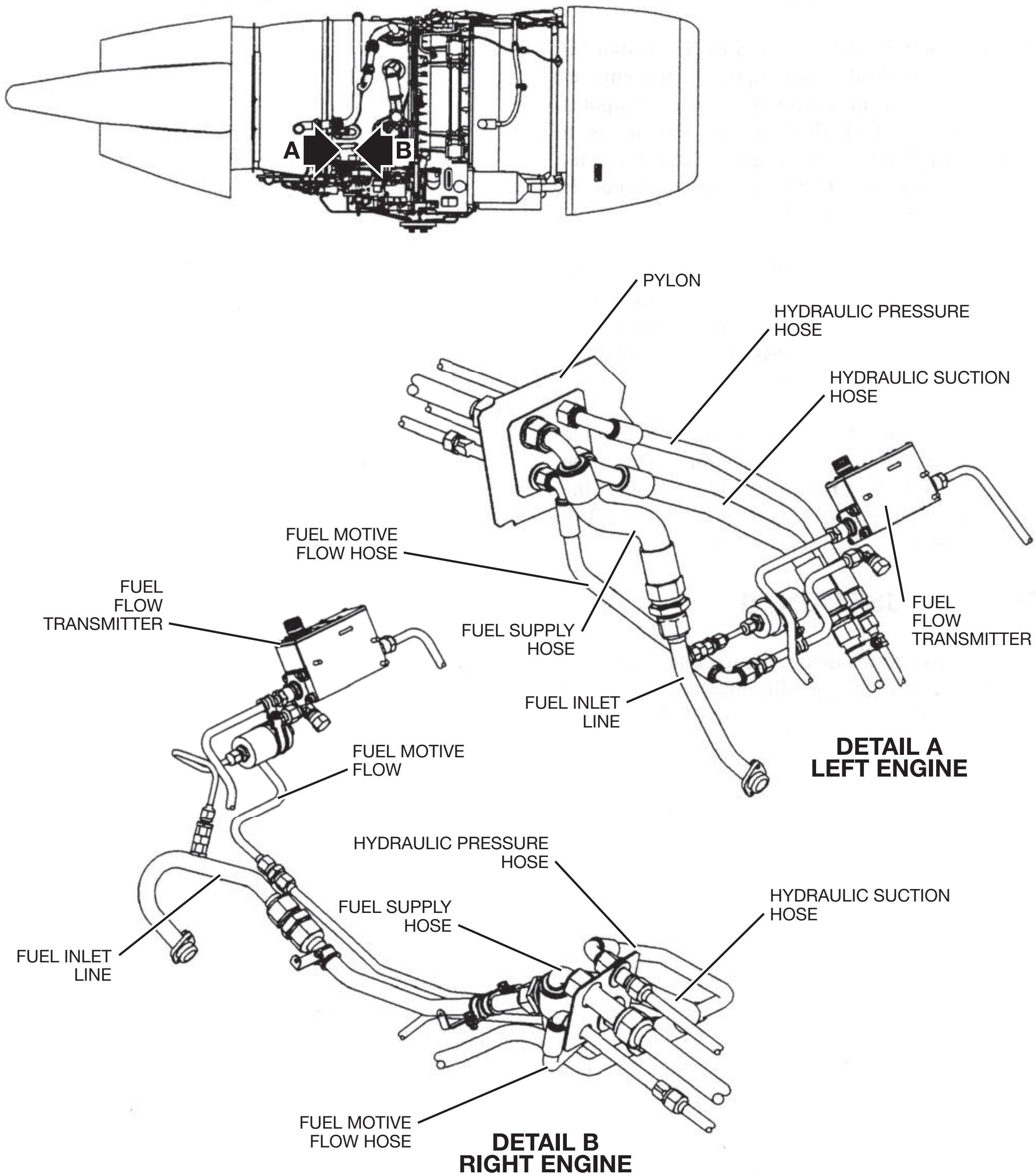


Figure 71-32. Fuel Flow Indicating Components

Fuel Flow Indication

Aircraft 5001 through 5268

The internally lighted, fuel flow/fuel quantity indicator is located on the center instrument panel (Figure 71-32) and is a dual channel fuel indicator.

The upper portion of the indicator has two digital displays, with the left display indicating fuel flow for the left engine and the right display indicating fuel flow for the right engine. The calibrated range of the fuel flow display is 0 to 1,990 pounds per hour. When the rotary TEST switch (SC060) is positioned to ANNU, both fuel flow displays will show "1888". When the rotary TEST switch is positioned to OFF, both displays will show actual fuel flow. When you remove input power from the indicator, the left and right digital displays are blank.

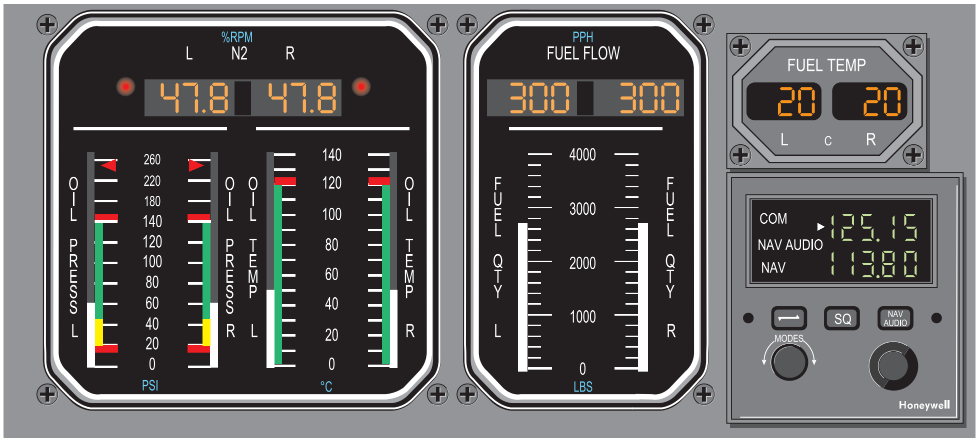
The lower portion of the indicator has two vertical tape displays, with the left display providing an analog indication of the left engine's fuel quantity and the right display providing an analog indication of the right engine's fuel quantity. The fixed vertical scale for the fuel quantity is 0 to 4,000 pounds. When you remove input power from the indicator, an OFF warning flag appears in each vertical tape display window.

Aircraft 5269 through 6000

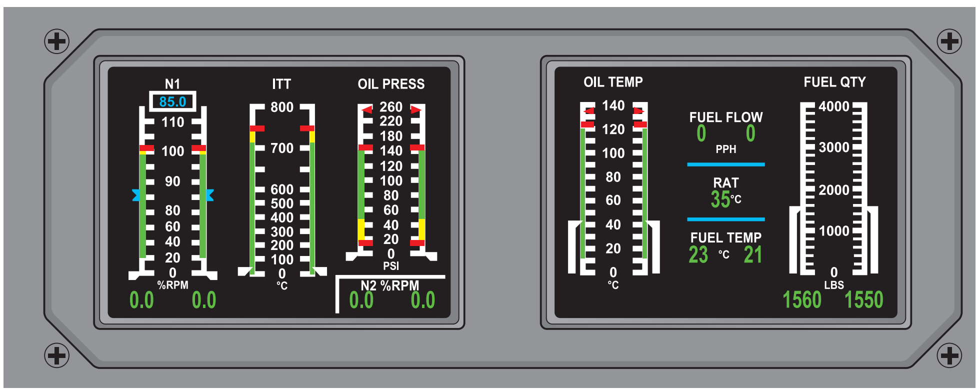
The fuel flow/fuel quantity is shown on the engine indicating system AMLCD and is located on the center instrument panel (Figure 71-32).

The AMLCD has two displays. The right display shows fuel flow for the left and right engine. The AMLCD comes on automatically when you start the engines and receives its input signals from components and sensors installed on the engines and in the fuel tanks. When you remove input power from the AMLCD, the fuel flowlfuel quantity is not shown.

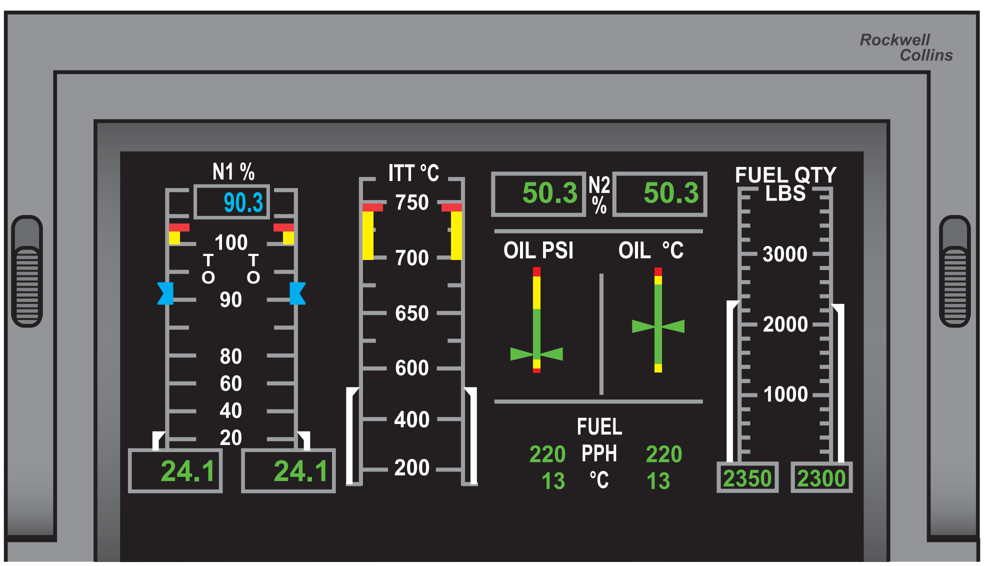
NOTES



XL (SNs 5001–5268)



XL (5269 AND SUBSEQUENT)
XLS (5501 THROUGH 6000)



XLS+ (6001 AND SUBSEQUENT)

Figure 71-33. Engine Insruments

Aircraft 6001 and Subsequent

The fuel flow/fuel quantity is shown on the EICAS start page. The EICAS start page can be seen on the pilot or copilot MFD (Figure 71-33).

The EICAS start page is shown automatically on the pilot MFD when you put the BATT switch to the ON position.

NOTES

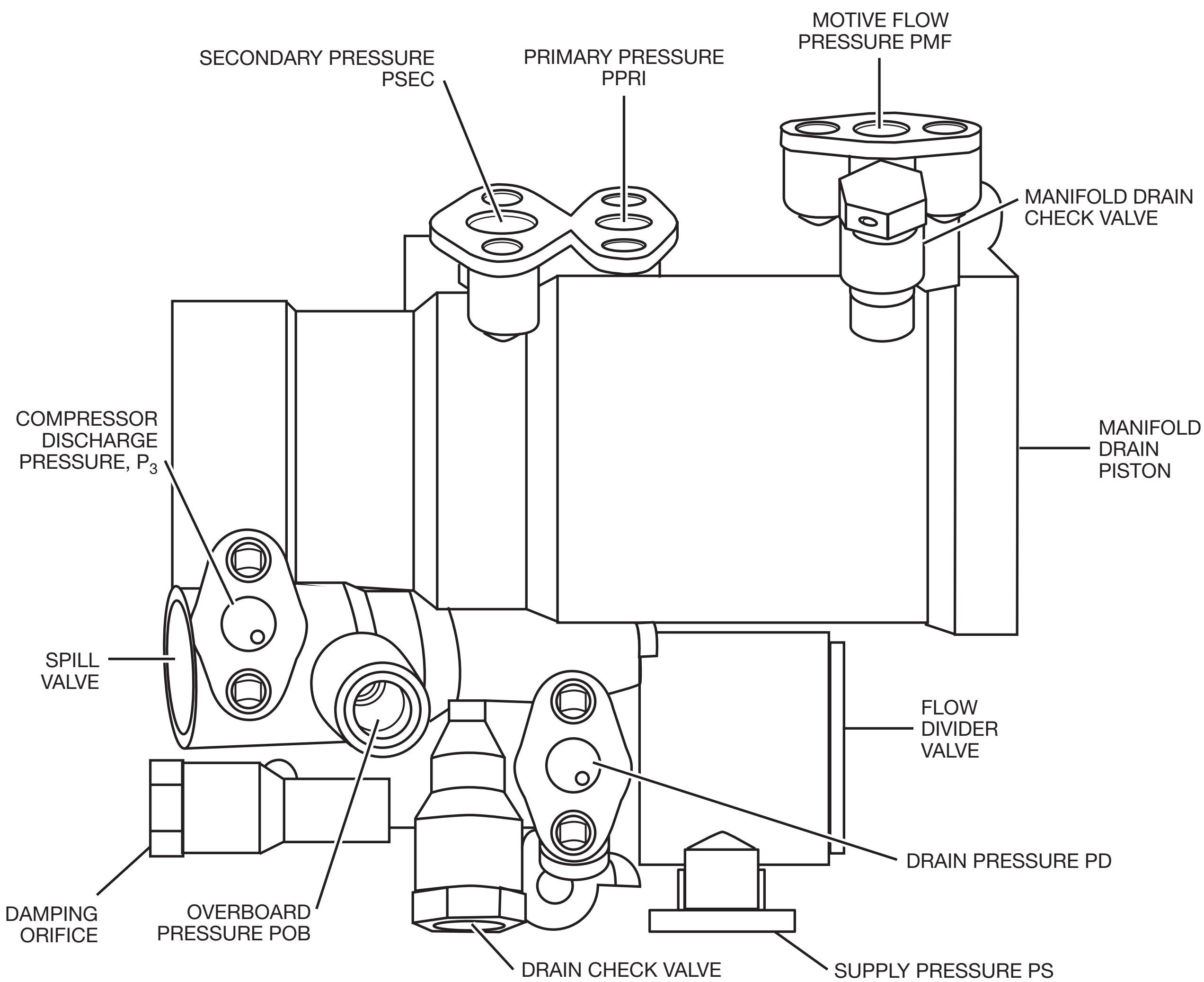


Figure 71-34. Flow Divider Valve

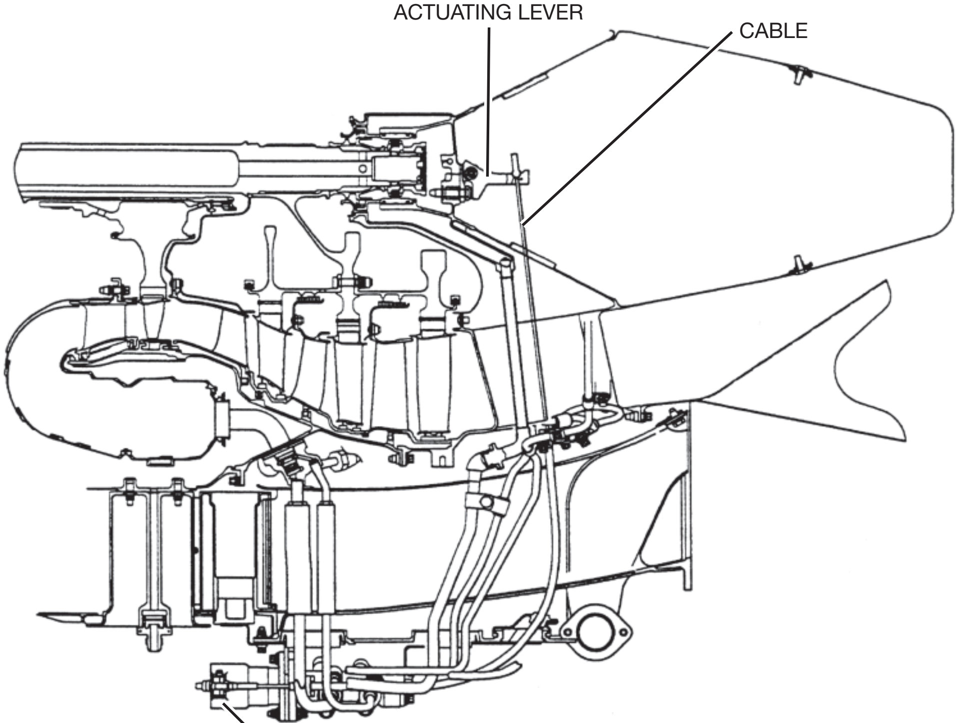
Flow Divider Valve

The flow divider valve divides fuel from the FCU between the primary and secondary manifolds (Figure 71-34). It also provides regulation of spill fuel flow during engine start, based on FCU schedule and P3 compressor discharge pressure. These functions are provided by the FMU on the PW545C engine.

Initially during start, the flow divider provides fuel flow to only the primary manifold, primes the secondary, and "spills" the rest of the fuel back to the FCU. As P3 air pressure reaches approximately 30 psi, the spill valve closes and fuel is then also supplied to the secondary manifold. P3 air pressure reaches 30 psi around 26 to 28% N_2. Both primary and secondary manifolds then supply fuel to the combustion chamber throughout all engine speeds. When the throttle lever is moved back to "cut-off, the flow divider valve directs all fuel to the spill valve.

A holding reservoir for holding fuel (from the manifold upon shutdown of the engine) is incorporated into the flow divider valve. The reservoir is made up of a cylinder with a springloaded piston inside. During engine operation, motive-flow fuel pressure compresses the spring and holds it compressed until shutdown. At shutdown, motive-flow fuel pressure goes to zero; thus allowing the spring to push the piston back. And, at the same time, it draws the fuel back out of the manifolds and stores it until the next engine start. Upon the next engine start, as motive-flow fuel pressure increases, the stored fuel is then pushed back out and is used for engine combustion. The reservoir holds up to 3 aborted starts before it is full.

NOTES



545A/B

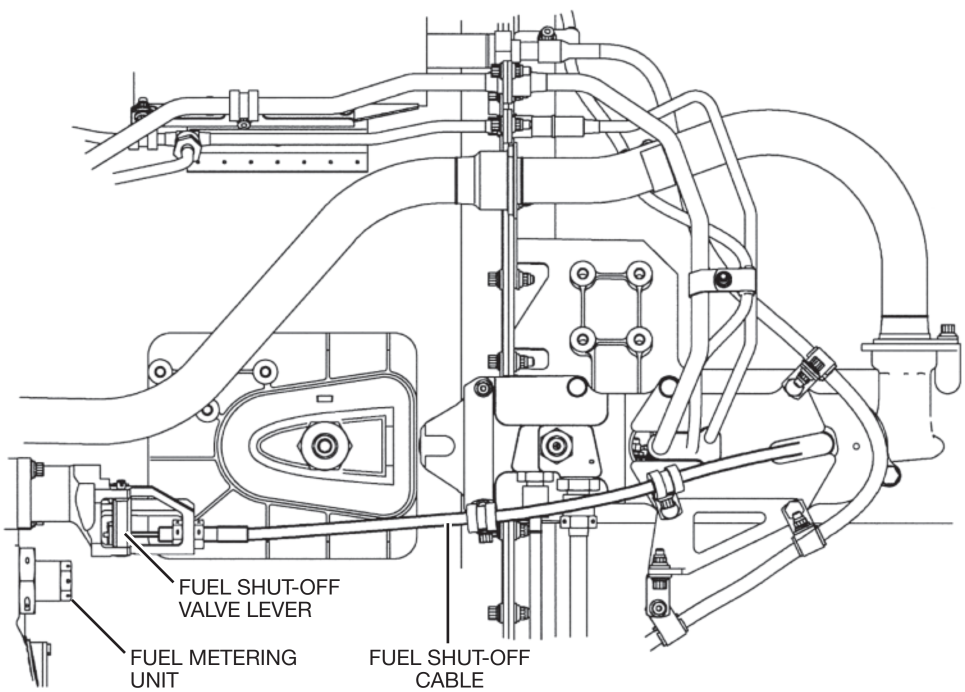


Figure 71-35. Emergency Fuel Shut-Off System

Emergency Fuel Shut-Off

The emergency fuel shut-off valve is between the flow divider and the fuel manifold assembly (Figure 71-35). Both the primary and secondary manifold fuel lines are routed through the shut-off valve independently. There is a lever assembly inside the exhaust cone at the aft end of the LP rotor shaft. From the lever assembly, a cable is routed down to a tripper assembly on the emergency fuel shut-off valve. The shut-off valve shuts off fuel going to the combustion chamber if for any reason the LP rotor shaft shifts aft, during engine operation. The engine would then shut down to prevent continued operation. On the 545C engine these functions are provided by the fuel shut-off valve piston assembly located inside the FMU.

NOTES

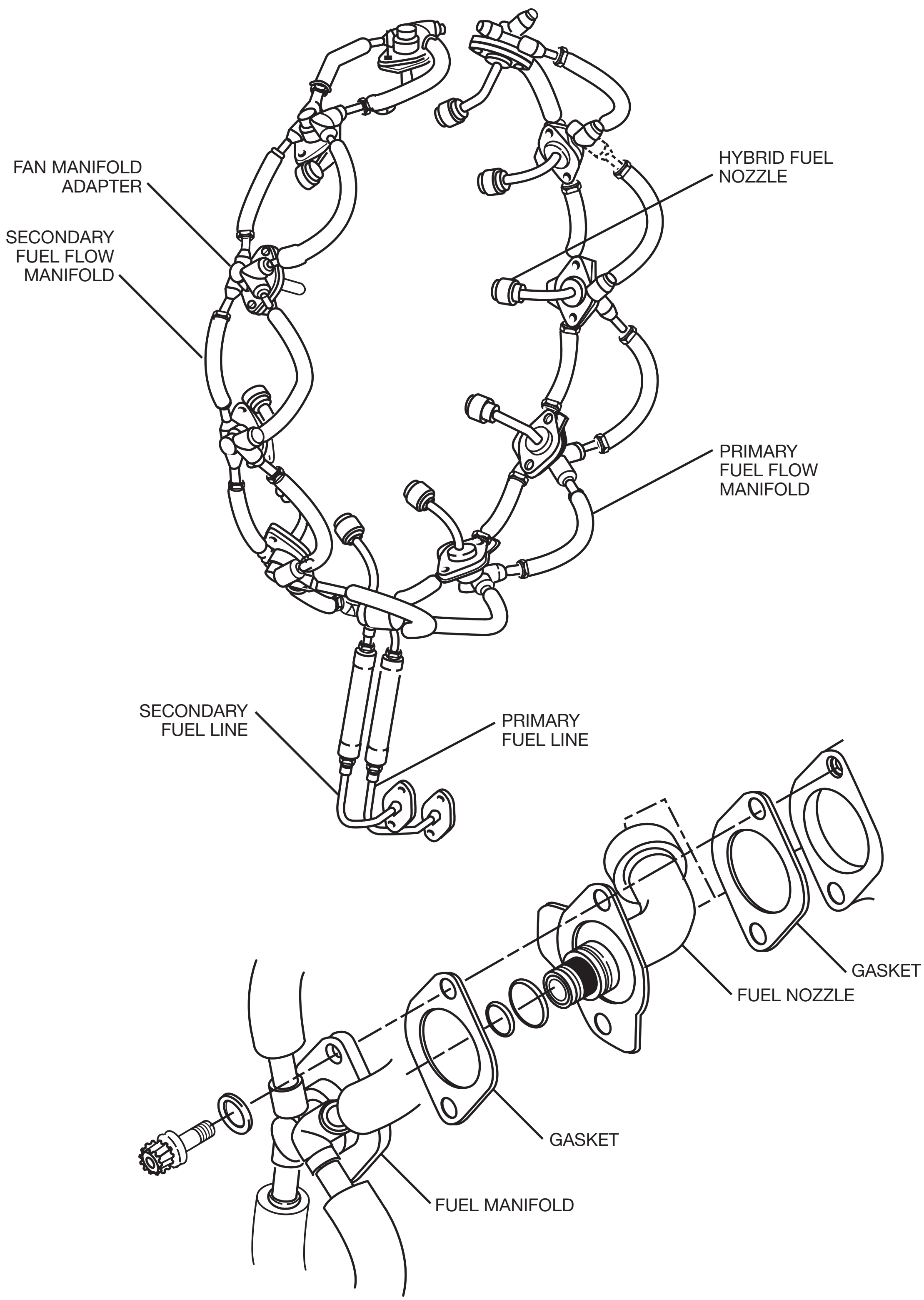


Figure 71-36. Fuel Manifold Assembly

Fuel Manifold

The fuel manifold distributes fuel to respective fuel nozzles inside the combustion chamber (Figure 71-36). Two sets of flexible hoses (one primary and the other secondary) connect the 11 fuel manifold adapters to each other. The flexible hoses are a crimped-end design and are permanently attached to each of the manifold adapters.

NOTES

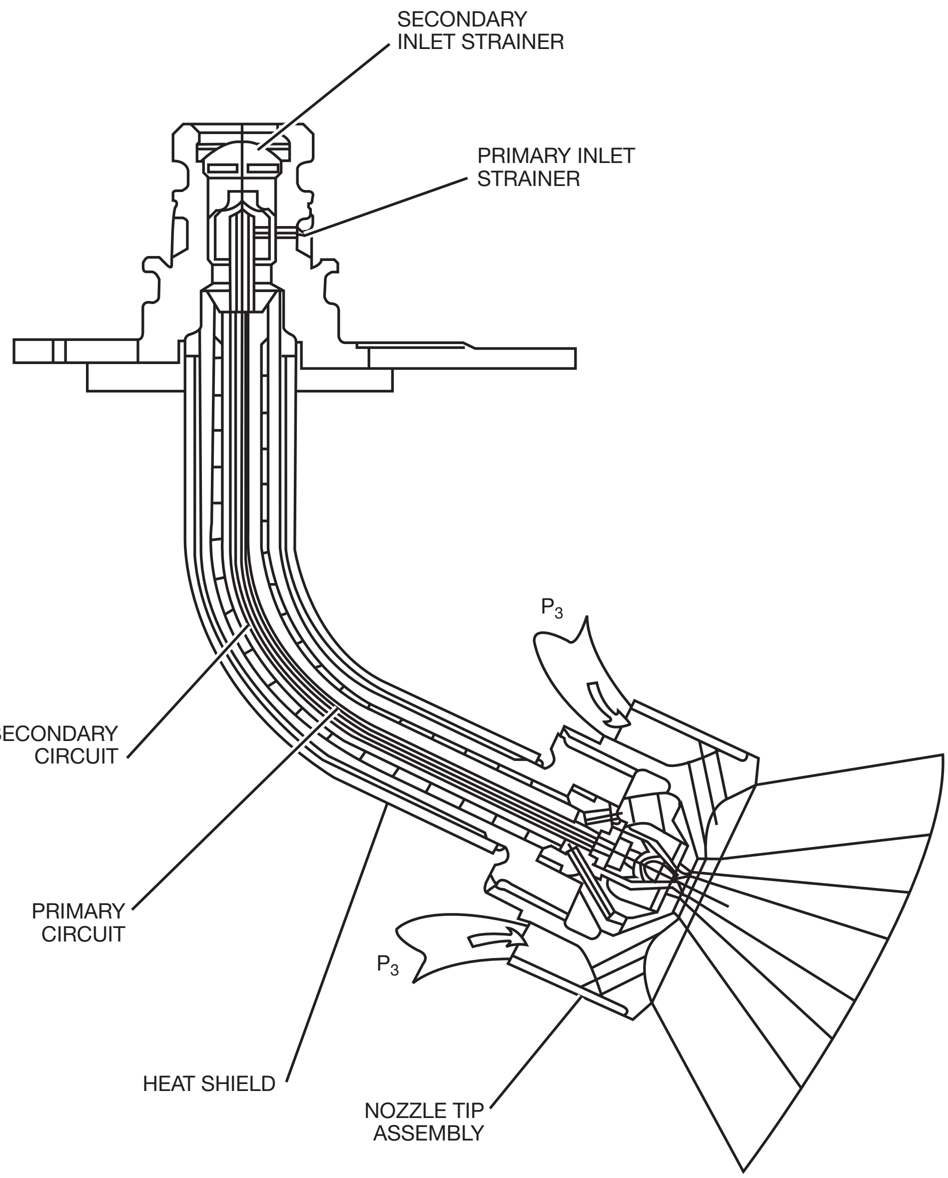


Figure 71-37. Fuel Nozzles

Fuel Nozzles

The fuel nozzle atomizes fuel within the combustion chamber (Figure 71-37).

Each nozzle consists of:

- Nozzle body
- Nozzle tip support/heat shield
- Spray tip assembly
- Primary and secondary orifices
- Primary and secondary strainers

The heat shield surrounds the tip support and protects inner parts from excessive heat during engine operation. The primary and secondary strainers are inside the fuel nozzle to prevent contaminants from entering the nozzle spray tip assembly.

Primary Fuel Atomization

Fuel is supplied under pressure to the primary circuit of the fuel nozzle body. The fuel is then routed through a strainer and calibrated orifice, through internal passages and finally to the spray tip assembly. It then passes through the metered distributor and into the spin passage of the primary cone, where it emerges as a conical sheet of fuel. Finally the fuel hits high velocity swirling air, which shreds the fuel sheet and breaks it up into fine atomized fuel droplets.

Secondary Fuel Atomization

Secondary fuel is supplied under pressure to the secondary circuit where it is routed through a strainer, a calibrated orifice, through internal passages and finally to the spray tip assembly. From there, the fuel travels through the secondary fuel swirler passages where it is spun in a swirling motion. The secondary fuel then moves towards the exit annulus where it emerges as a conical sheet of fuel. The fuel sheet is then hit with high velocity swirling air which further shreds and breaks it up into fine droplets.

NOTES

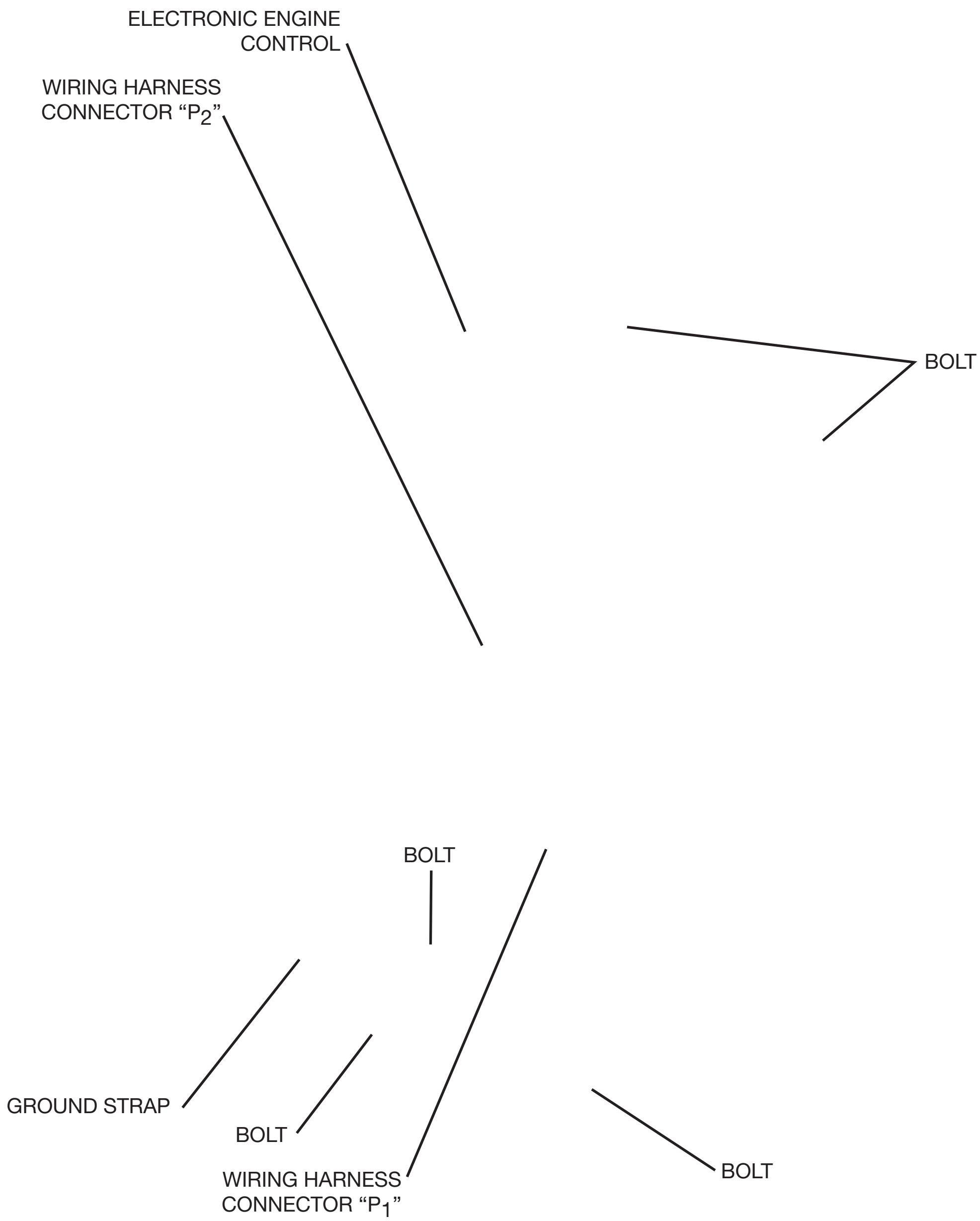


Figure 71-38. Electronic Engine Control (EEC)

ENGINE CONTROLS AND INDICATIONS

Description

545A/B engines provide two main modes of engine control. A left instrument panel mounted ENGINE COMPUTER switch allows selection of MAN (manual) or AUTO modes (Figure 71-38).

In the AUTO mode, the engine is controlled by an electronic engine control (EEC) unit, which controls the engine low rotor (fan N_1) speed and thereby engine thrust in accordance with the detented throttle lever position and prevailing ambient conditions.

In the AUTO mode, the EEC supplies detented throttle lever positions idle functions and selected engine synchronization.

In the case of an EEC major fault, the system automatically reverts to the manual (reversionary) mode. In the reversionary mode, the hydromechanical fuel control unit takes over full control of engine speed (high rotor N_2) in response to throttle lever position which moves the fuel control lever by mechanical linkage from the cockpit.

In manual mode the throttle position detents, ground idles and engine synchronization are turned off and an annunciator warning panel advisory light (white) illuminates, EEC MAN-L or R (Figure 71-39).

EEC MANUAL
Advisory—Electronic engine control is off-line (failed or selected off) and the engine is operating in manual mode. Throttle detents and engine sync are inoperative.

Figure 71-39. EEC Manual Annunciator (XL/XLS)

In the AUTO mode, input is supplied to EECs from throttle bellcrank RVDTs (rotary variable displacement transformers), EECs control steady state and transient operation of both engines. The system will modulate fuel flow rate to engine in response to input from airplane sensors and measurements of engine operating conditions. EECs automatically prevent engine N_2 from exceeding speed limits.

A fault indication panel is located inside and forward of the forward tailcone access door. It provides ECS FAULT (Engine Control System) LH and RH bite indicators, for maintenance personnel.

The detent throttle system consists of left and right throttle detent service air solenoid shut off valves, left and right pneumatic air cylinders, associated plumbing and left and right throttle detent lever assemblies. The electronic engine control (EEC) switches to Auto Mode during engine start at 20% N_2 . This is indicated when the EEC MANUAL L or R annunciator lights extinguish. At the same time, 28 volts is applied to the respective throttle detent service air solenoid shut off valve to open the valve. This ports service air from the cabin door seal supply line to extend the respective pneumatic air cylinder which engages its throttle detent assembly.

The detents remains engaged as long as the engine control remains in Auto Mode. If an engine reverts to manual Mode or if the Auto/Manual switch in the cockpit is selected to Manual mode, the EEC removes 28 volts from the detent bleed air solenoid shut off valve, closing the valve and removing the service air supply from the pneumatic air cylinder to disengage the respective detent lever assembly from the throttle system.

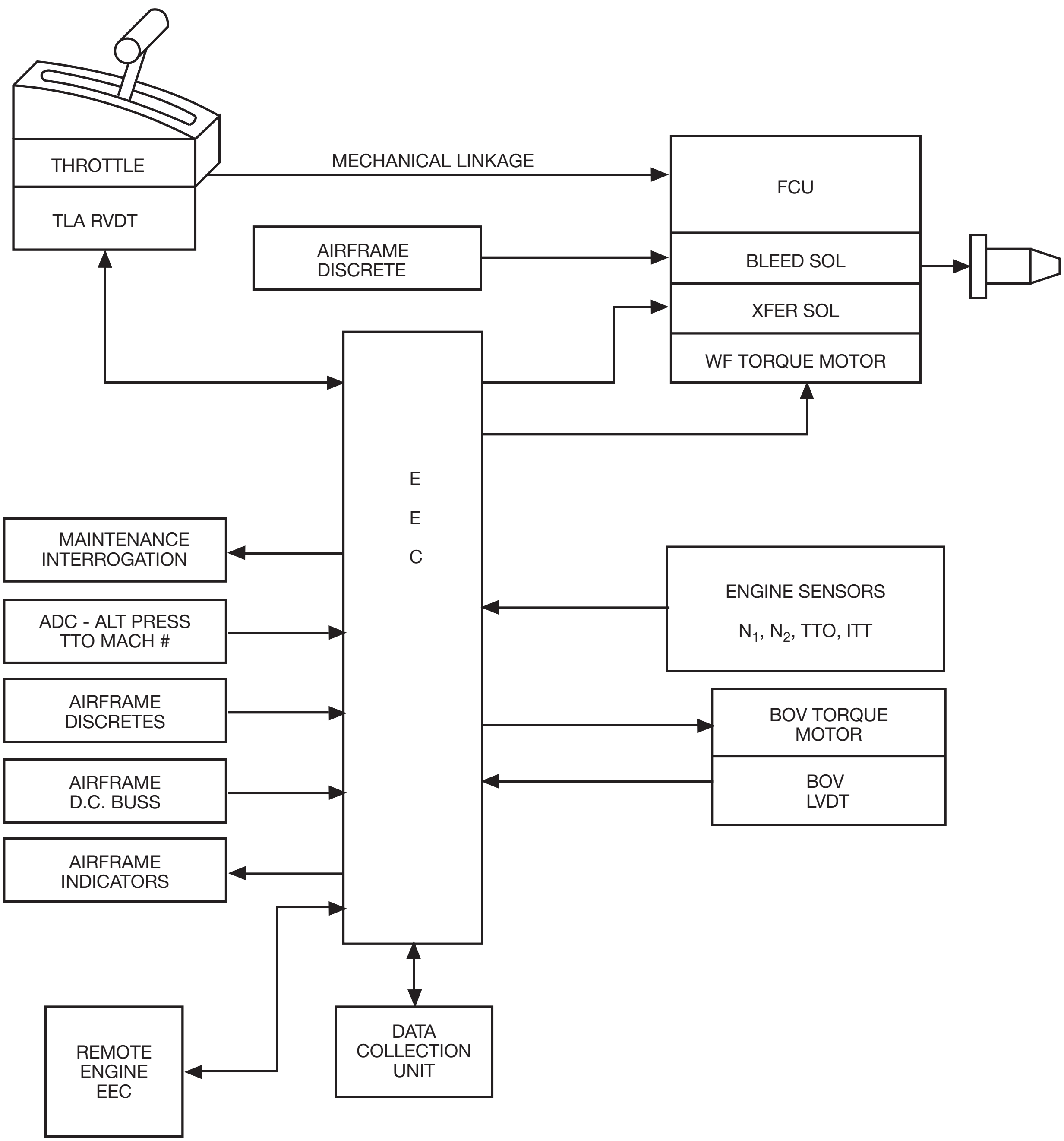


Figure 71-40. Engine Control System Schematic (XL/XLS)

Components

Electronic Engine Control (XL/XLS)

The EEC operates on a 28 volt DC input from the airframe. Continuous voltage is required to power the EEC and the independent N_2 overspeed system, contained in the same enclosure. The control system receives discrete inputs—frequency inputs and analog inputs from the local engine, the remote engine and the airframe (Figure 71-40). The frequency and inputs are checked for range, rate, interface failures, and in some cases are cross-checked against other parameters to determine if the input is valid. If the input is determined to be faulty, another input is selected to replace it when applicable. Discrete inputs are checked for validity where applicable. The EEC processes the following inputs:

- Fan inlet total temperature (TTO)
- Power lever angle position (TLA)
- Inter turbine temperature (ITT)
- N_1 rotor speed
- N_2 rotor speed
- Ambient Pressure (PAMB)
- ADC data via ARINC (pressure altitude, Mach. No., TTO)
- Several discrete inputs (bleed state, WOW, sync, and thrust reverser)

The EEC is a single-channel microprocessor-based controller. In the Auto mode the EEC provides the following functions, (in response to the TLA signal received from the RVDT on the throttle quadrant):

- Detented throttle, automatic thrust setting (N_1 governing)
- Idle governing (N_2 governing) at ground idle, flight idle and anti-ice idle
- Acceleration and deceleration limiting (N_2)
- N_1 and N_2 speed limiting (100% redline)
- Closed-loop bleed valve (BOV) control
- Engine diagnostic system functions (EDS.)
- Overspeed protection (N_2)
- N_1 and N_2 synchronization.

In the case of an EEC major fault, the system automatically reverts to the reversion mode. In the reversion mode, the FCU takes over full control of the engine speed, in response to the throttle position, which moves the FCU PLA by means of a mechanical linkage from the throttle in the cockpit. The following functions are provided:

- Pilot settable power setting (N_2 governing)
- Idle governing, (N_2 governing) at flight idle
- Acceleration and deceleration limiting (ratio unit control)
- N_2 speed limiting
- Closed-loop bleed valve (BOV) control
- Limited diagnostic system functions (EDS)

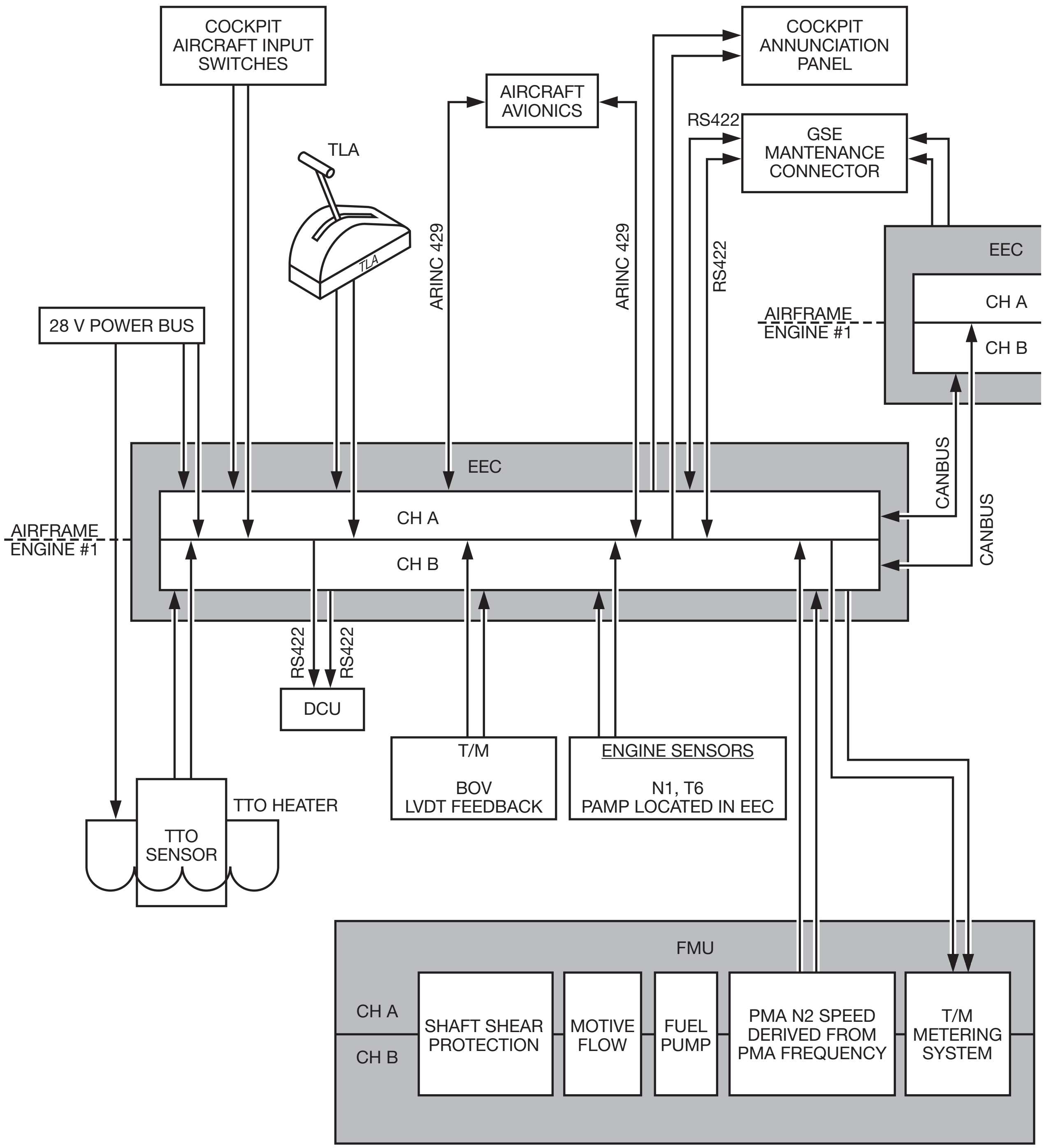


Figure 71-41. Electronic Engine Control-Schematic XLS+

XLS+ Electronic Engine Control

The PW545C is equipped with a dual channel electronic engine control (EEC). The EEC provides cockpit indication of engine faults and can indicate up to 3 possible states: All Clear, Channel Maintenance Advisory Indication (minor fault present), Channel Caution Indication (major fault present). The system is dispatchable in the first two states but is subject to time limited dispatch in the second state. The third state is a nondispatchable state.

The EEC controls the engine in accordance with pilot demands, ambient conditions and engine operating limits. The EEC is the LRU that contains the system electronics and is electrically connected via engine and airframe harnesses to input signals from the EEC engine sensors, as well as airframe discretes and the TLA signal. Using these inputs, the EEC modulates the fuel flow by means of a torque motor in the FMU and modulates the bleed valve by means of a torque motor in the Bleed OFF Valve (BOV), which provides position feedback via a LVDT.

The EEC is a single LRU containing both channels on two separate printed circuit boards. Each channel has two connectors, one for the interface with the engine and the second for the airframe. The two channels have several formats for digital communication. ARINC 429 is used for communication with the cockpit avionics.

A bi-directional link is provided in the hardware to allow communication between the avionics and the EEC. Two separate RS422 UART ports are provided for communication with the ground support equipment (GSE). Each channel has two CANBUS cross communication links for the exchange of data between individual EEC channels and between EECs for the transmission of data between engines for synchronization and for fault detection and accommodation purposes.

The EEC is able to transfer control from one channel to another in the event of a malfunction on that channel. Control of the engine is maintained in the presence of multiple faults through a hierarchical scheme that maintains the most fit channel in control. The channels are designated Channel A and Channel B.

Identical software is loaded into each channel and only one channel can be in control of the output devices at a time. The software in each channel reads the other channel's health status and determines which is the healthier channel to remain in control.

During each ground test start, the channel in control is switched to confirm that the standby channel is capable of control and is free from faults only detectable by having control of the engine. This process reduces the probability of dormant failures.

The airframe inputs to the EEC consist of analog throttle lever angle information, electrical power, Ambient Pressure from an Air Data Computer (ADC) and various hardwired discretes (Figure 71-41).

Outputs from the EEC system include: Torque motor drivers for the fuel metering unit (FMU) and Bleed OFF Valve (BOV), output drivers for the ignition system, the TTO heater, the cockpit annunciator panel, ARINC serial data to the aircraft avionics, CANBUS serial data between the local and remote engine RS422 serial data to communicate with the Ground Support Equipment for maintenance purposes.

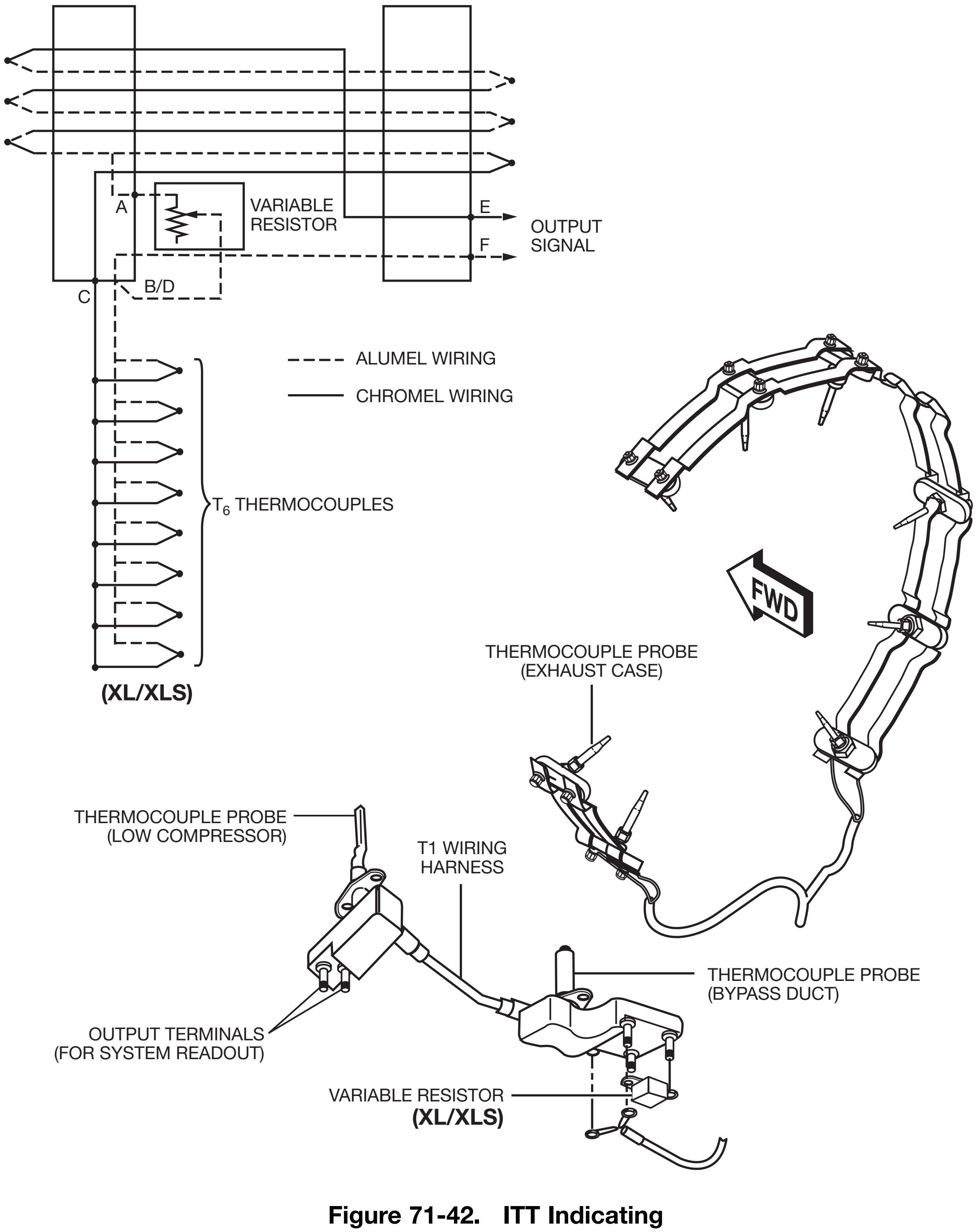


Figure 71-42. ITT Indicating

Special Considerations

Illumination of the Engine Control Fault L-R CAS message (Figure 71- 43) indicates a detected EEC system fault which causes degraded engine operation. In this condition, the engine is no longer dispatchable. Special operating procedures in this mode can include:

- Degraded operability margin (surgelaccel time).
- Inability to shut-down via the TLA directly.
- Inability to modulate thrust.
- Inability to modulate takeoff thrust.
- Inability to restart.
- Inability to achieve Ground Idle.
- Degraded ITT limiting during starts.
- Loss of ITT indication.

ENGINE CONTROL FAULT L-R			
Color	Inhibited By		Debounce
Amber	LOPI	TOPI	Standard
	POD	EFI	
		*SIPI	

This message is posted when a FADEC channel has failed. It is posted when FADEC 429 Label 271, bit 12 = 1 (Channel Fail Indication).

This message is inhibited for 20 seconds during initial DCU power up. This is to prevent nuisance indication due to the FADEC performing a power up test and activating this bit for 10 seconds.

* The message is also inhibited by an engine and/or APU start on the ground.

Figure 71-43. Engine Control Fault L-R CAS Message

Interturbine Temperature Indicating (XL/XLS)

The interturbine temperature (ITT) (T4.5) sensing system provides the pilot with an indication of the engine operating temperature (T4.5) (Figure 71-42). The temperature sensing system consists of integrated T1 and T6 systems from which a simulated interturbine temperature (T4.5) readout is computed.

The T6 system consists of:

- A braided wiring harness
- Four bus bars
- Eight individual thermocouple probes connected in parallel

The thermocouple probes are on the exhaust case and protrude into the gas path. The bus bars connect each probe to a twin-lead output, which provides connection to the T1 harness at the T1.4 thermocouple probe housing. For ease of identification, leads are color-coded with red for alumel, and yellow for chromel. To prevent inadvertent cross-connection, the lugs and terminal bolts are different diameters. Alumel is larger in diameter and slightly magnetic (negative polarity).

The T1 system consists of an insulated wiring harness with two temperature probes (T1 and T1.4). Each incorporates three thermocouples connected in series. When installed on the engine, the T1 probe protrudes into the inlet air stream in the fan case. The T1.4 probe protrudes into the bypass airflow through the bypass duct. The T1 probe housing body has two terminal studs, to provide aircraft connection for the system output signal.

By sensing the inlet (T1) and outlet (T1.4) temperatures of the bypass air stream, a rise in air temperature (through the bypass duct) can be established by subtracting T1 from T1.4. This is accomplished through the T1 system. Since each probe contains three thermocouples connected in series, the rise in temperature is multiplied by three to provide the required readout. The T6 thermocouple probes are connected in parallel through the wiring harness to provide an average exhaust temperature; and (with the T6 and T1 harnesses connected in series) a total readout of T6 plus three times the rise in bypass air temperature is provided as the output signal. A variable resistor (connected to the wiring harness on the T1.4 probe housing) is adjusted during final acceptance tests of the engine (to standardize the T4.5 readout range).

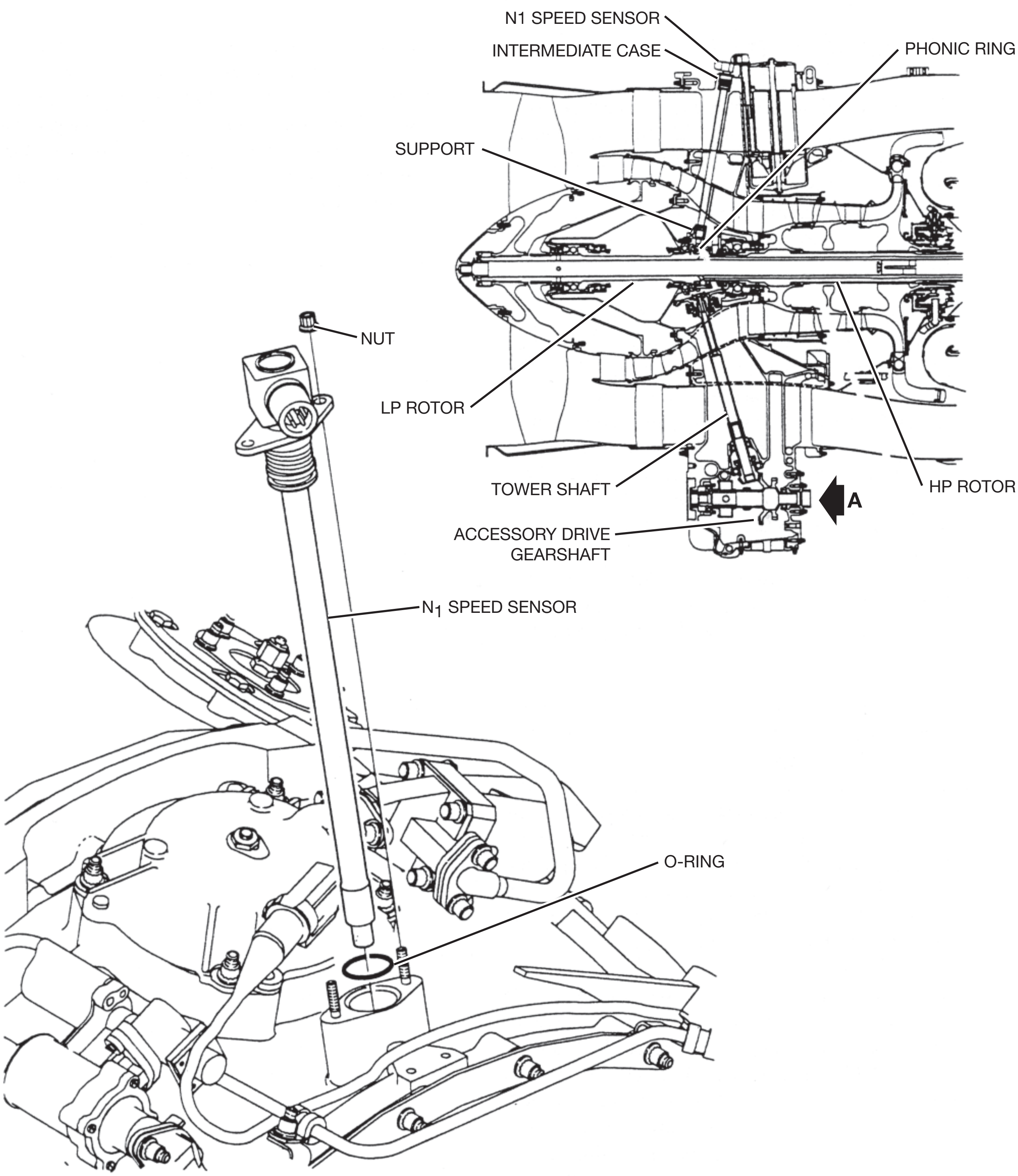


Figure 71-44. N_1 Speed Sensor

XLS+

The ITT (T4.5) sensing system provides the pilot with an indication of the engine operating temperature. Output from the T6 thermocouple is combined with other operating data to calculate the T4.5 in the FADEC.

The T6 system consists of a braided wiring harness, four bus bars and eight individual thermocouple probes connected in parallel. The thermocouple probes are installed on the exhaust case and protrude into the gas path. The bus bars connect each probe to a twin-lead output which provides connection to the electrical wiring harness at a terminal bracket on the bypass duct.

For ease of identification, leads are color coded with red for alumel and yellow for chromel. To prevent inadvertent cross-connection, the lugs and terminal bolts are of different diameters, alumel being larger in diameter and slightly magnetic (negative polarity).

Because the terminal junction connects the T6 harness (chromel-alumel) to the electrical wiring harness (copper), it also acts as a thermocouple and thereby alters the T6 indication. The cold junction in the back of the EEC connectors, which consists of a resistance temperature detector (RTD), senses the temperature around the junction and trims the T6 signal that is sent to the EEC.

XL/XLS Rotor Speed Sensing

Two speed sensors, one mounted at the 12 o'clock position on the intermediate case and one located on the accessory gearbox right-hand side, sense LP rotor (N_1) and HP rotor (N_2) speeds, respectively. Both N_1 and N_2 sensors consist of dual coils wound around a magnetically permeable rod (Figure 71-44).

A permanent magnet is bonded to one end of the sensor, the other end is located close to rotating gear teeth or phonic ring teeth. As the gear teeth or phonic ring sweep past the sensor, a magnetic flux is conducted out from the magnetically permeable rod. The resulting changing flux induces a voltage across each coil that is transmitted to the aircraft cockpit and bleed valve control.

N_1 Rotor Speed Sensing

The N_1 sensor is secured to a pad on top of the intermediate case (Figure 71-44). The sensor passes down through a hollow strut and the lower end of the sensor fits into a support bolted to the rear flange of the No. 2 bearing housing. The sensor is directed at a phonic ring on the LP shaft. The resulting voltage is transmitted to the cockpit instrumentation via the main control harness to display N_1 speed.

XLS+ N_1 Rotor Speed Sensing

One speed sensor (N_1) is mounted at the 12 o'clock position on the intermediate case, senses LP rotor speed. The N_1 sensor consists of dual coils wound around a magnetically permeable rod.

A permanent magnet is bonded to one end of the sensor, the other end is located close to rotating gear teeth or phonic ring teeth. As the gear teeth or phonic ring sweep past the sensor, a magnetic flux is conducted out from the magnetically permeable rod. The resulting changing flux induces a voltage across each coil that is transmitted to the EEC.

The N_1 sensor is secured to a pad on top of the intermediate case. The sensor passes down through a hollow strut and the lower end of the sensor fits into a support bolted to the rear flange of the No. 2 bearing housing. The sensor is directed at a phonic ring on the LP shaft. The resulting voltage is transmitted to the cockpit instrumentation via the main control harness to display N_1 speed.

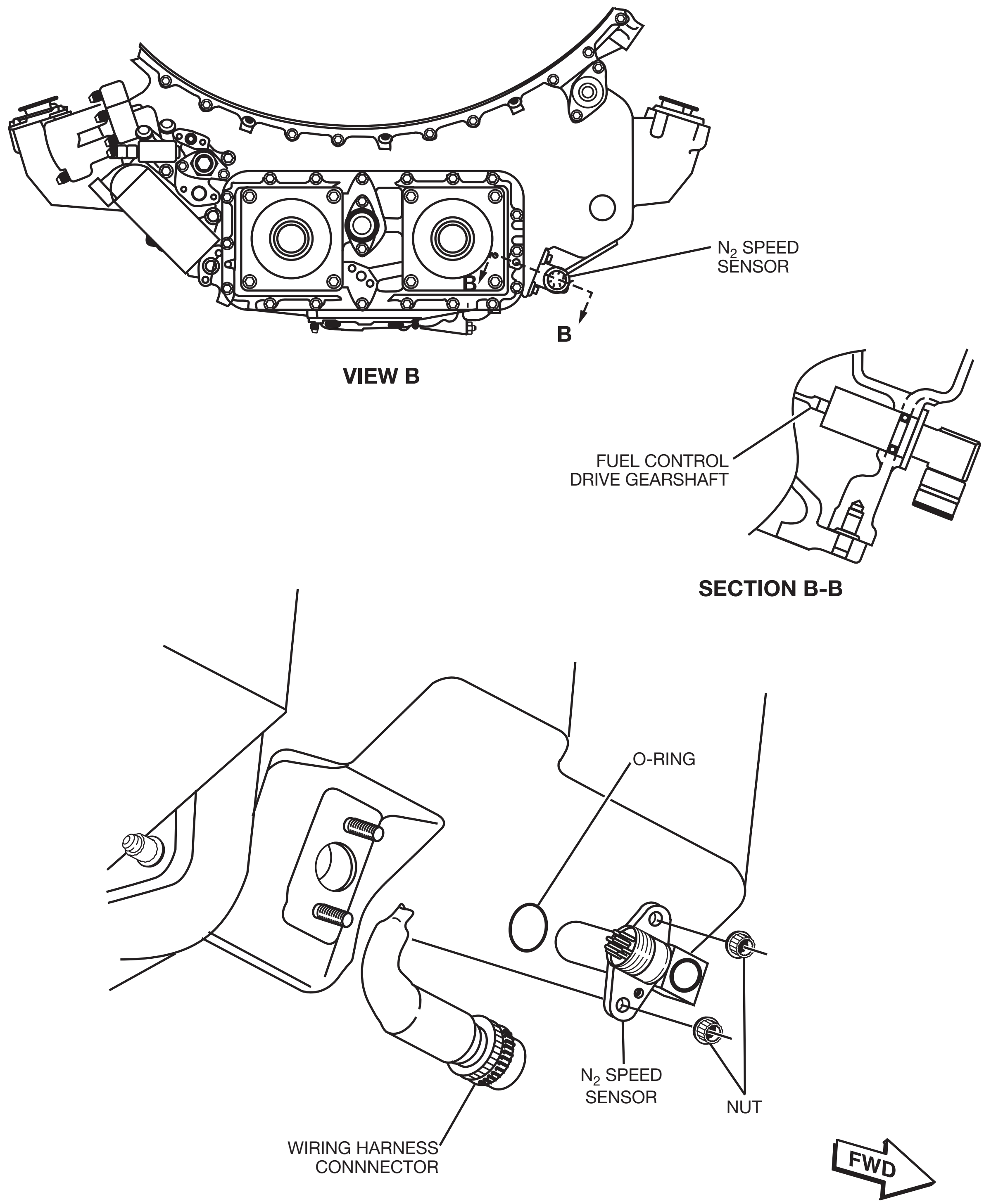


Figure 71-45. N_2 Speed Sensor (XL/XLS)

N_2 Rotor Speed Sensing

XL/XLS

The N_2 speed sensor is on the accessory gearbox right side (Figure 71-45). The sensor is directed at the teeth of the FCU drive gearshaft that is geared to the HP rotor via the accessory drive gearshaft and tower shaft. The resulting voltage is transmitted to the aircraft cockpitinstrumentation via the main control harness. The N_2 signal is also used by the EEC to compute correct bleed valve position.

XLS+

The HP rotor speed (N_2) is derived by the EEC from the frequency of the signal it receives from the PMA, which is mechanically connected to the HP rotor.

NOTES

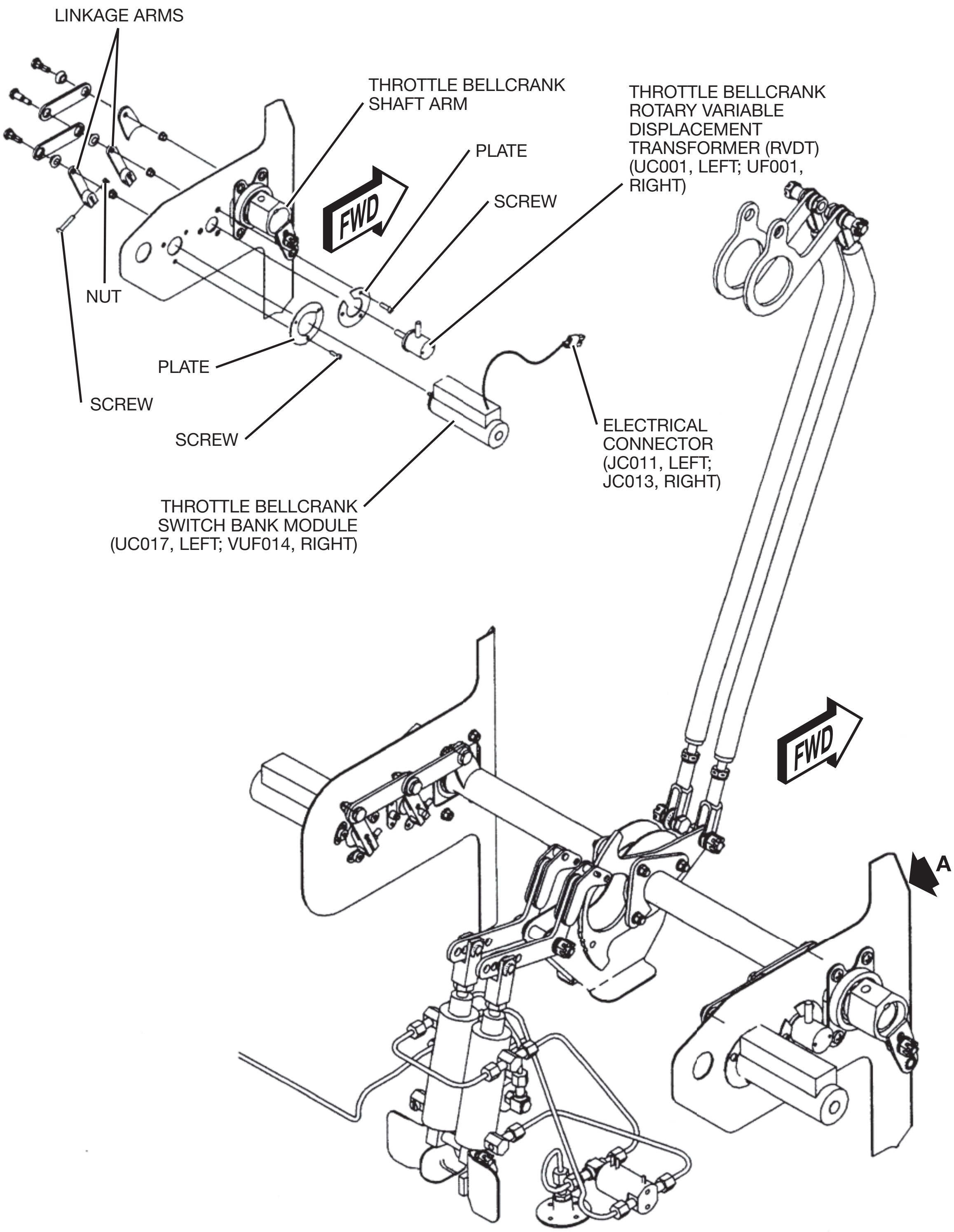


Figure 71-46. Throttle Bellcrank Assembly (Units 0001 through 0016)

Throttle Lever Inputs

XL/XLS

The throttle levers are installed in the control pedestal and are connected to the bellcrank assembly shaft by push-pull rod assemblies. Throttle cables attach to the throttle bellcrank shaft arm levers and are routed aft, under the cabin flooring. They attach to the aft control cables (in the aft cabin) and continue aft through the aft pressure bulkhead and then upward over the tailcone baggage compartment, aft, and connect to the aft engine control cables which route outboard through the pylons to the respective engine fuel control bellcrank linkage (Figure 71-46).

Operation

The throttle is manually operated to select desired engine power settings. Motion of the throttle lever in the throttle quadrant is transmitted to the respective bellcrank which rotates and repositions throttle bellcrank shaft arm, the RVDT and switch bank module arms. This movement is also transmitted to the respective engine fuel control by three push-pull cable assemblies to obtain the required direction of movement of the fuel control lever, from the fuel cutoff position to full throttle position.

Bellcrank Assembly

Components at the bellcrank assembly are:

- Throttle switch bank modules
- Throttle rotary variable differential transformers (RVDTs)
- Throttle detent pneumatic air cylinders
- Throttle detent bleed-air shutoff valves

Throttle Switches

Two sets of four switches are used for the throttle switches. One set under the pilot side floor and the other under the copilot floor. Each switch bank is then connected into the bellcrank assembly. Their single common shaft rotates as their respective throttle lever is repositioned. The switches then give throttle position to the following systems:

- Deck A .... Pressurization 62°
- Deck B .... Landing gear 45°
- Deck C .... Speedbrakes 54°
- Deck D .... No takeoff 54°
- Deck E .... Takeoff indicator 74°
- Deck F .... Climb indicator 66°
- Deck G .... Cruise indicator 58°

Throttle switches are adjusted, using an ARINC reader; and then are referenced to degrees of TLA as shown on the reader.

NOTES

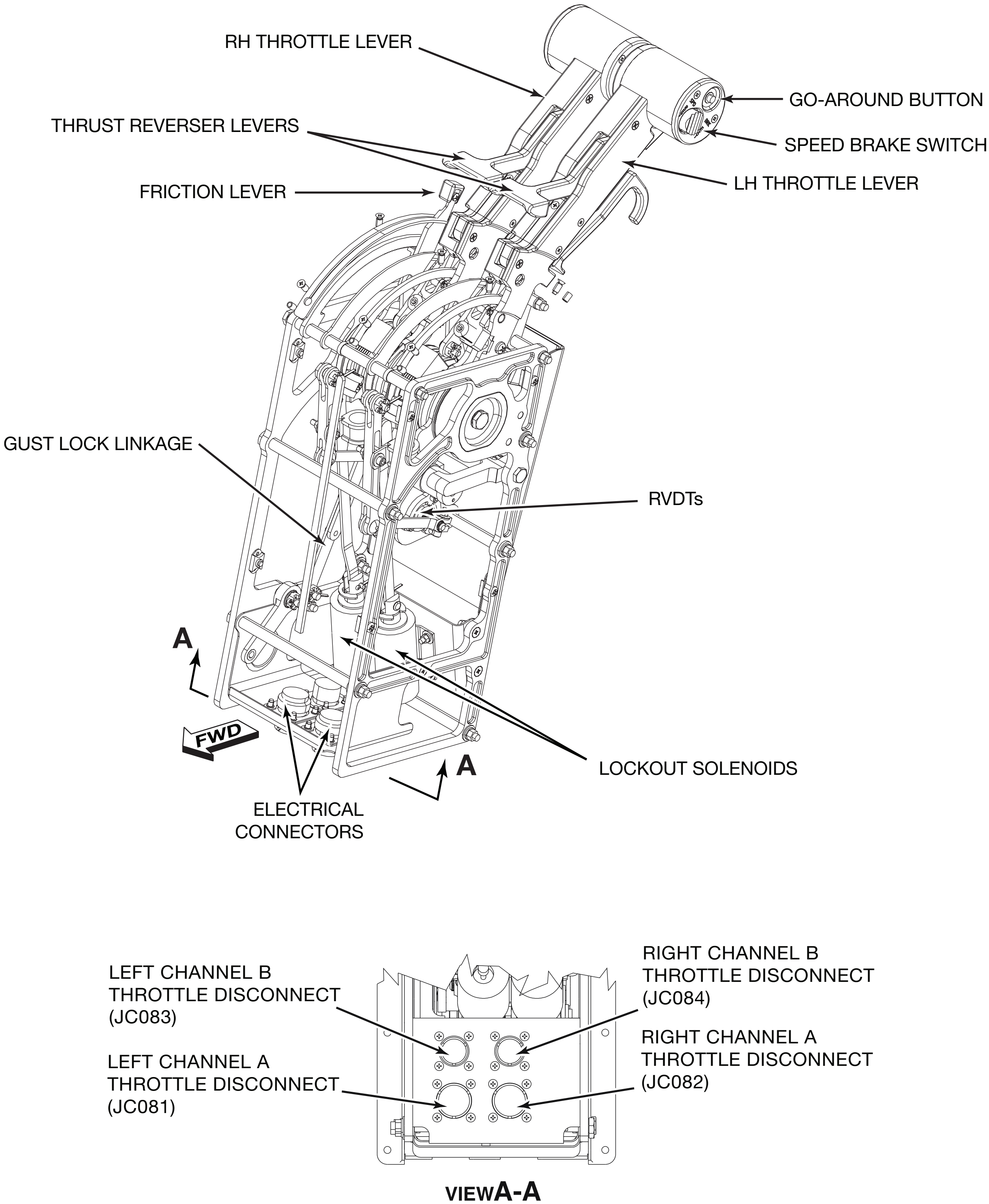


Figure 71-47. XLS+ Throttle Quadrant

Engine Power Controls

XLS+

The throttle levers are installed in the control pedestal and are not mechanically connected to the engine fuel monitoring unit. Engine thrust is governed by signals sent to the engine mounted dual channel full authority digital engine control (FADEC) from the RVDT connected to the throttle lever (Figure 71-47).

Operation

The crew moves the throttles to the correct position for the necessary engine thrust. The engine control system on these airplanes uses a dual channel, FADEC to control the engines low rotor speed (N_1). The FADEC uses a RVDT in the throttle quadrant to find the TLA. The FADEC uses the data from the RVDT and det

FADEC Controls

XLS+

The XLS+ has one dual channel FADEC installed on each engine. One channel is in control while the other is a backup. The channel in control changes each time that the engine is started or when a fault is found in the active channel.

The FADEC controls the high pressure rotor speed and schedules fuel flow during acceleration and deceleration with automatic compensation for ambient conditions.

The FADEC will not change to a channel that cannot operate. The pilot can reset faults on the FADEC with the FADEC RESET switch light. The FADEC RESET switch lights are installed on the left tilt panel.

The pilot can align the speed of the engines with the ENGINE SYNC NORM/OFF switch light. The ENGINE SYNC NORM/ OFF switch light is installed on the center pedestal, in the ENGINE switch panel. For information on troubleshooting the FADEC, contact Pratt & Whitney.

Operation

The primary function of the FADEC system is to control the thrust of the engines. The throttle lever angle (TLA) tells the FADEC which throttleposition the pilot selects. The FADECs control two dual-element rotary variable differential transformers (RVDTs) and supply electrical signals that agree with the angle of the throttle levers.

The FADEC monitors thrust settings and adjusts for ambient temperature and pressure altitude. It also monitors critical engine parameters and controls operational values. The FADEC gives a throttle feedback function, which moves the thrust to idle if a thrust reverser deployment signal is received during flight or when the TLA is above idle.

The anti-ice, bleed air, and thrust reverser system is related to the FADEC controlled parameters. The FADEC also controls the motive flow valve, ignitors and the bleed valve to provide compressor surge protection.

NOTES

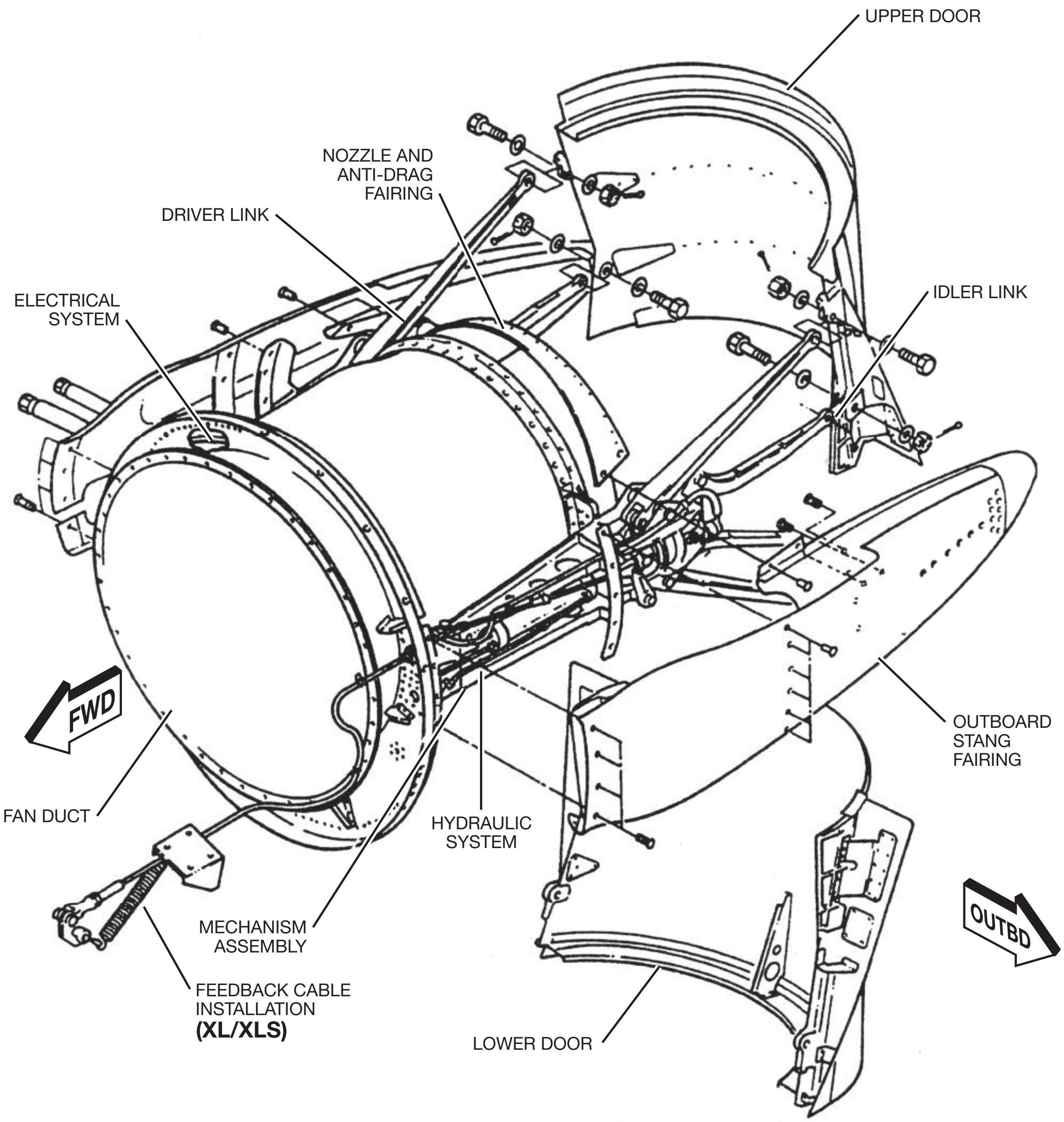


Figure 71-48. Thrust Reverser Components

THRUST REVERSERS

Description

The thrust reverser is a hydraulically-actuated four-bar linkage target-type reverser, on the aft end of each engine (over the fan duct assembly) (Figure 71-48). The fan duct assembly is a component of the thrust reverser assembly.

When deployed after landing and during roll out, the reverser doors actuate from the stowed position and join behind the exhaust nozzle cone to deflect engine exhaust forward, over and under the nacelles.

NOTES

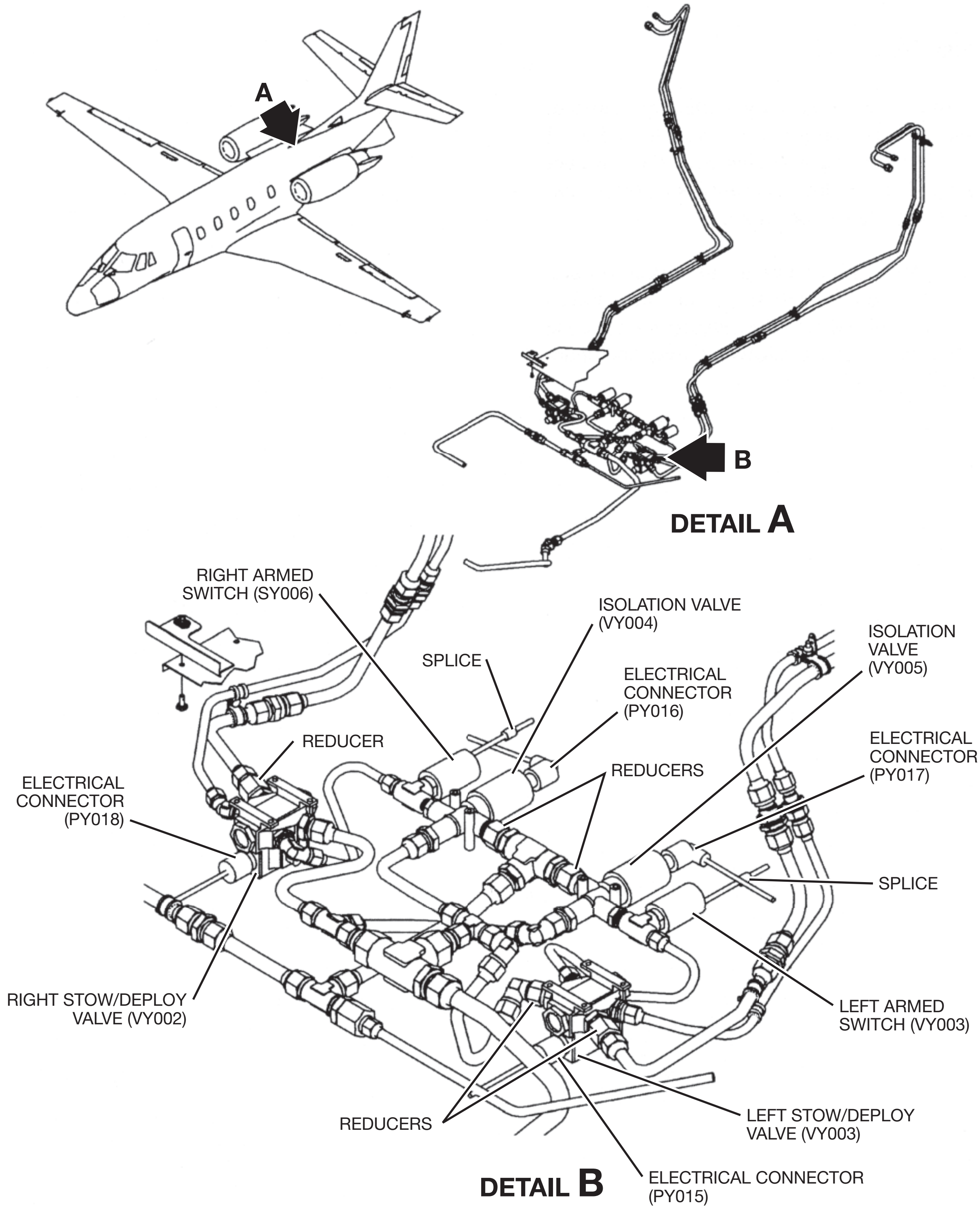


Figure 71-49. Thrust Reverser Hydraulic Panel XL (Sheet 1 of 2)

Components

Thrust Reverser Panel

The thrust reverser panel is aft of the hydraulic panel in the fuselage fairing (Figure 71-49).

The thrust reverser panel includes:

- Isolation valves
- Arm switches
- Control valves
- Restrictors
- Check valves in the return line

Isolation Valve

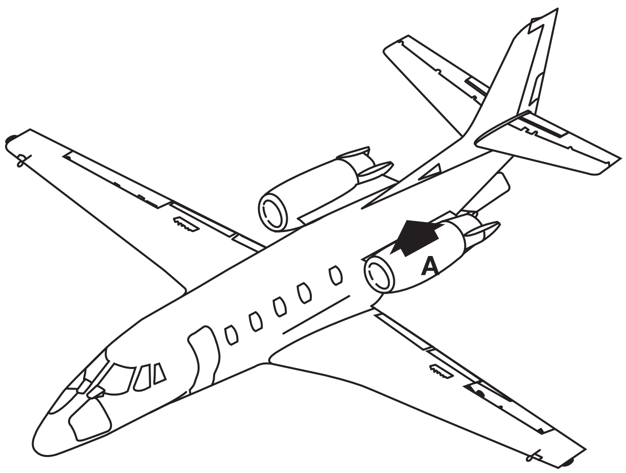
In normal operation, the airplane hydraulic system supplies a maximum of 1500 psig. The isolation valve functions to isolate the control valve and actuators from the airplane hydraulic system, while thrust reversers are not in operation. This is accomplished by blocking off the high-pressure inlet port. Pressure drop through the valve is less than 30 psi during normal operation.

In operation, the isolation valve solenoid is energized by moving the thrust reverser control to deploy, which induces 28 VDC on the solenoid. The solenoid remains energized during the deploy translation and remains in that state while the system is in the deploy mode. When the thrust reverser lever is moved to the stow position, power is applied to the isolation valve through the stow switches (during the stow cycle). Upon completion of the stow cycle, the stow switches open and remove power from the solenoid, which then isolates the system from the hydraulic supply pressure.

Arm Switch

The arm switch is connected by a tee, off the hydraulic high pressure line between the isolation valve and the control valve. The purpose of the arm switch is to sense a hydraulic pressure of 200 psig, closing an electrical switch that completes a ground circuit for the ARM lights (amber) in the fire tray. When the pressure drops to 100 psig, the switch opens.

NOTES



FILTER ASSY'S
2 PRESSURE
2 RETURN

MAIN MANIFOLD ASSY

L AND R T/R
MANIFOLDS

DETAIL A (XLS/XLS+)

Figure 71-49. Thrust Reverser Hydraulic Panel (Sheet 2 of 2)

Control Valve

The control valve is a hydraulic four-way three-position spool valve with two independently energized solenoids. The control valve directs hydraulic pressure to the actuator deploy or stow ports as each mode is selected. When one port is pressurized, the other port is vented to the low-pressure hydraulic return. Both solenoids are never energized at the same time. The control valves are on the thrust reverser panel.

Restrictor

There are two restrictors in the hydraulic system, between the stow ports (on the actuators) and the control valves. The two restrictors that control hydraulic flow in both directions. The restrictors prevent the actuator piston from overrunning the hydraulic supply flow, at airplane forward deploy speeds up to 120 KIAS.

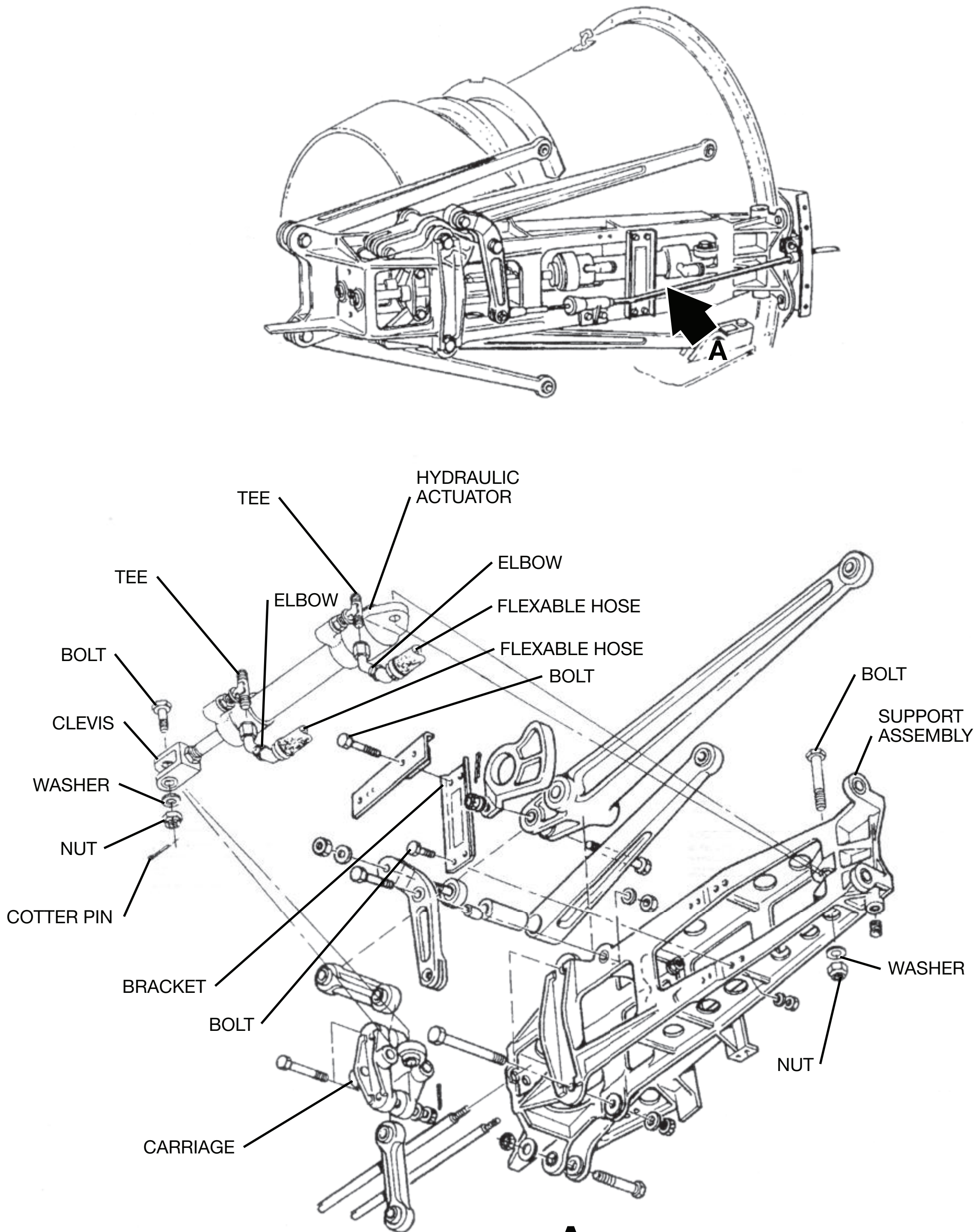
Check Valve

There is a check valve just before the tee-fitting in the return lines, from the isolation and control valves. There is one check valve installed for each thrust reverser system. The check valves allow free hydraulic fluid flow back to the airplane system, with no flow in the reverse direction. The check valves are rated for 3.5 GPM flow rate.

NOTE

The XLS/XLS+ utilize a hydraulic manifold in place of these components. Operation is the same.

NOTES



DETAIL A

Figure 71-50. Mechanism Assembly

Thrust Reverser Assembly

Fan Duct Assembly

The fan duct assembly is a component of the thrust reverser assembly. The fan duct assembly attaches to the rear flange of the engine bypass duct and incorporates an attaching exhaust nozzle cone. The fan duct assembly provides structural mounting for the mechanism assembly.

Mechanism Assembly

The mechanism assembly consists of an actuator-driven carriage which moves forward along guide rods in a support assembly, causing driver links to extend and drive the reverser doors, and idler links to the deploy position. Reversing the direction of the carriage causes the driver links to retract, and the reverser doors to stow and lock (Figure 71-50).

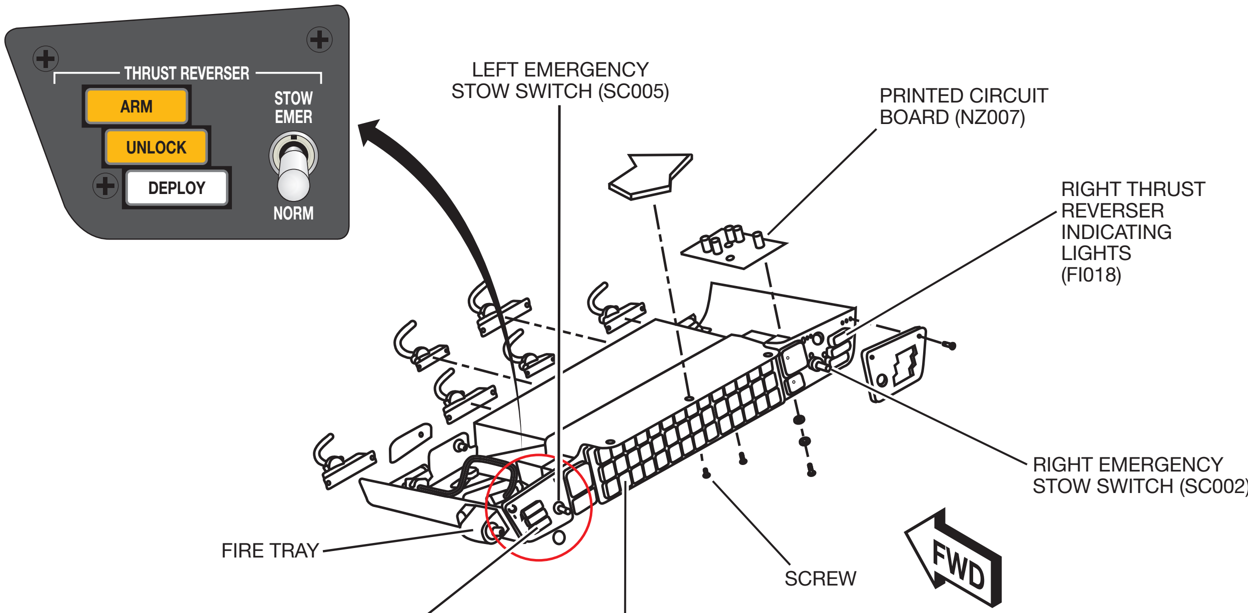
Forward movement of the carriage moves the over-center links from the “locked” to the “unlocked” position. The overcenter links in turn, exert force on the driver links, causing the driver links to pivot about bushings in the support assembly. This raises the leading edge of the reverser doors into the airstream flowing over and under the nacelle. The driver links continue to move the reverser doors, which in turn pivot the idler link, until the reverser doors are fully deployed behind the exhaust stream of the engine.

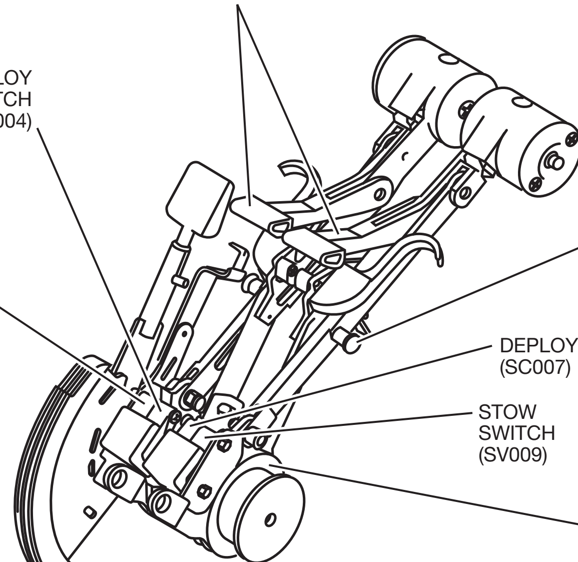
Aft movement of the carriage causes the overcenter links to exert force on the driver links which in turn pivot about bushings in the support assembly. This causes the reverser doors to move from the deployed to the stowed position. When the reverser doors are fully stowed, the carriage continues to drive the overcenter links until the link’s pivot on the end of the driver links into an “overcenter” or “locked” position.

Actuators

The actuators transform hydraulic energy into mechanical power to open and close the thrust reverser doors, through mechanical linkage. Each thrust reverser system has two actuators in a support beam structure on the inboard and outboard sides of the engine exhaust. The actuators are mounted horizontally along each side of the exhaust in a single-point pinned mounting. Two hydraulic lines are connected to ports which are used for high-pressure and return, depending on the mode of operation selected in the cockpit. Actuators inputs are from the control valves. Normal operating pressure for the actuators is 1500 psig.

NOTES





NOTE:
ROLLER PIN ALIGNS WITH THE SLOT IN THE CAM PLATE; SOLENOIDS ACTUATE, PULLING CAM AND PROVIDING ROLLER PIN FREE PASSAGE IN SLOT. THRUST REVERSER LEVER IS FREE TO OPERATE.

Figure 71-51. Thrust Reverser Control and Indicating

Controls and Indications

Electrical Control System

The electrical control system provides two functions. Hydraulic flow to the actuators is controlled through a series of switches and solenoid valves, which serves as an interlock to prevent inadvertent actuation of the system. Flow to the actuators is started by closing/opening of the thrust reverser switches in the cockpit (Figure 71-51). This connects the control system to the 28 VDC bus through circuit breakers. In normal operation, 28 VDC is supplied to the isolation/control valves through the emergency stow switch contacts. The deploy solenoid in the control valve is grounded by the squat switch being "closed". The stow limit switches provide the path for 28 VDC to the isolation and stow solenoid of the control valve. This causes the reverser to stow.

Electrical Indicating System

Each thrust reverser has a three-annunciator indicating system for the flight crew to monitor. The annunciator system is actuated by switch functions that complete a ground for their respective light. The annunciators illuminate during the "deploy" cycle and to extinguish during the "stow" cycle. The actuation and sequence of these annunciators is described under Operation. If the thrust reverser lever is inadvertently placed in the "idle reverse" detent position during flight, the MASTER WARNING light flashes and the ARM and HYD PRESS ON annunciators illuminate (Figure 71-52).

ARM
Illumination indicates pressure is available to the thrust reverser (pressure is sensed past the isolation valve). Illumination is normal on ground during TR operation, but abnormal inflight. Illumination inflight causes the red MASTER WARNING lights to flash.

UNLOCK
Illumination indicates the thrust reverser is unlocked. Illumination is normal on ground during TR operation, but abnormal inflight. Illumination inflight causes the red MASTER WARNING lights to flash.

DEPLOY
Illumination of the white light indicates the thrust reverser is deployed. Illumination is normal on ground during TR operation, but abnormal inflight.

Figure 71-52. T/R ANNUNCIATORS

NOTES

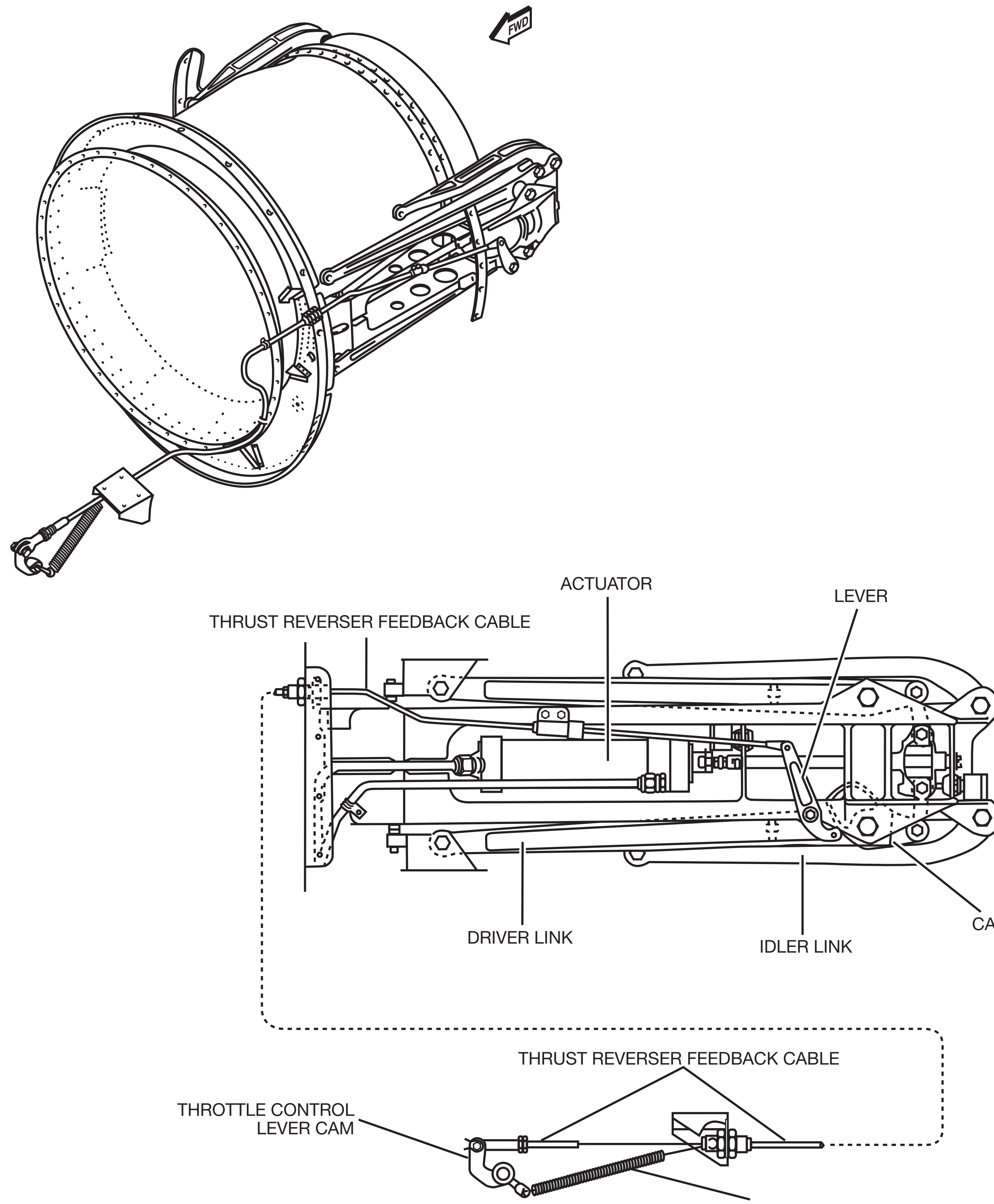


Figure 71-53. Thrust Reverser Feedback System (XL/XLS)

Thrust Reverser Feedback System

XL/XLS+

A feedback system is provided to make sure that the engine thrust is restrained at an idle level during inadvertent translation of the thrust reverser system (Figure 71-53). During normal operation, this is ensured by use of the thrust reverser control for throttle "up" and "down" operations. The feedback system ensures that if inadvertent deployment or stow occur, the engine power level is reduced to idle during the thrust reverser translation stage.

The feedback system is made up of a positive action mechanical linkage, connected directly to the FCU. Position of the thrust reverser system is transmitted by a cam connected directly to a driver link. A roller cam follower rides on the cam, the other end of the cam follower is attached to a flexible cable. During thrust reverser translation, the flexible cable drives a spring-loaded throttle control lever cam, which follows the airplane throttle linkage to the IDLE position. This prevents the linkage from advancing the power setting until the deploy cycle is complete.

Feedback control is not normally needed in the landing phase of thrust reverser operation, since the engines are at "idle" position for operation of the thrust reverser levers. If a thrust reverser deploys inadvertently during takeoff or while in flight, the feedback system reduces the thrust on the affected engine to the "idle" position during the translation cycle of the thrust reverser.

The XLS+ eliminates the mechanical feedback system and utilizes discrete inputs from the stow and deploy limit switches to limit engine power during the deploy and stow operation of the thrust reversers.

NOTES

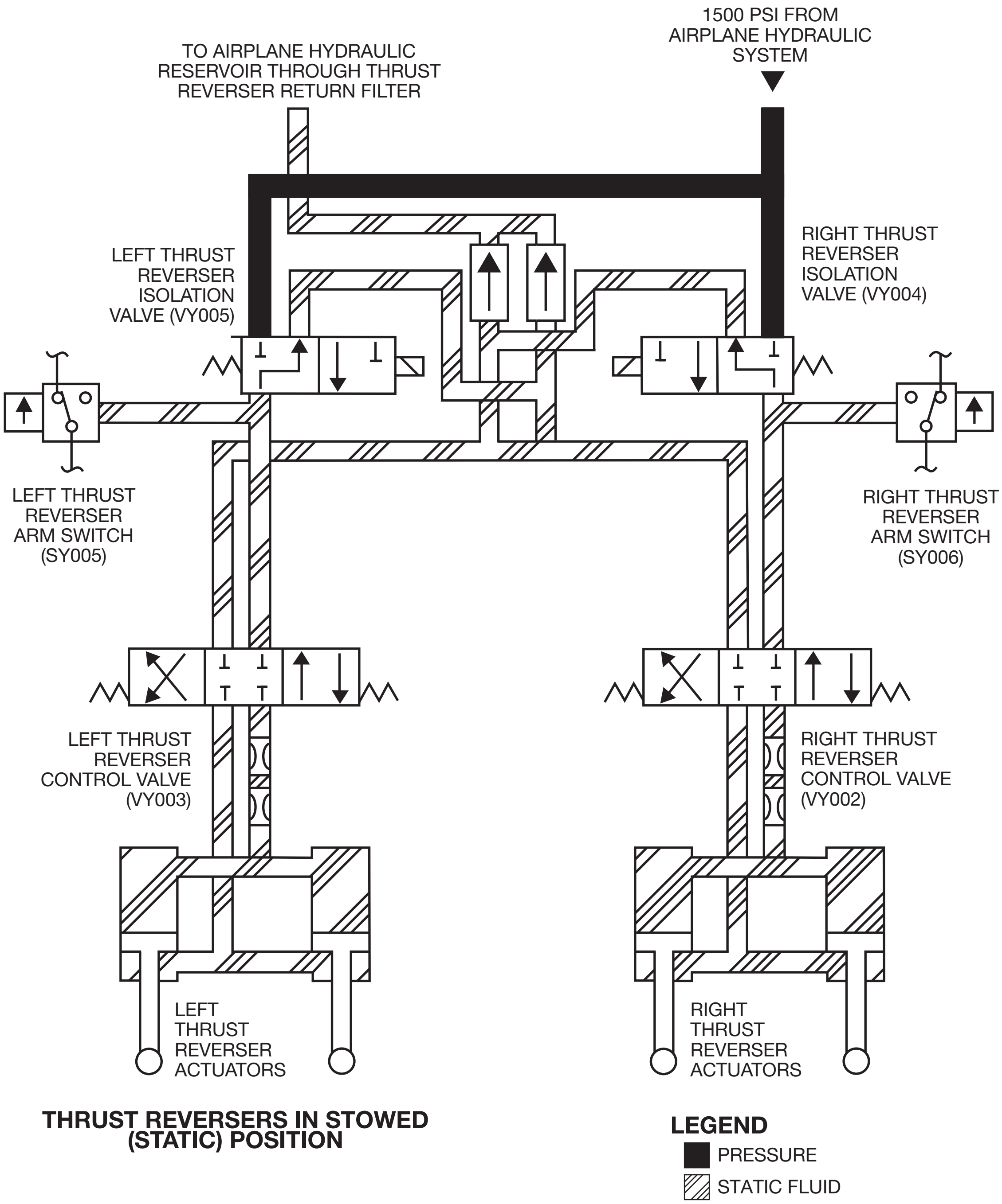


Figure 71-54. Thrust Reverser Hydraulic Diagram

Operation

Hydraulic Actuating System

Each airplane engine drives a hydraulic pump, connected by hydraulic lines through open center valves (Figure 71-54). The high-pressure line (1,500 psi) is routed to the thrust reverser isolation valves. From the isolation valves the hydraulic lines are routed to control valves. The arm switch is in the line between the isolation valve and the control valve. The control valves apply hydraulic pressure to either the deploy or the stow ports of the actuators. There are two restrictors between the stow ports of the actuator, and the stow port of the control valve, to limit the hydraulic flow in both directions. From the fuselage fairing area, the hydraulic lines run through the tail cone and then through the pylon, to the engine nacelle. The low-pressure (or return line) is always the other line, depending on which mode of operation is selected by the control valve. From the control valve, the return line bypasses the isolation valve, then returns to the airplane hydraulic system through a check valve. This prevents surge pressures from blocking the return line.

Sequence of Operation

Pulling back on the reverser levers (attached on the throttle levers) actuates the thrust reversers. This action directs a hydraulic supply from the airplane hydraulic system to the “deploy” port of the reverser actuator, while acting on a pressure switch in the system. The pressure switch transmits a signal which lights an ARM light (amber) in the cockpit.

Hydraulic pressure to the “deploy” port of the reverser actuator causes the actuator rod to retract and stows the reverser carriage forward. This unlocks the linkage mechanism and extends the mechanism driver links, driving the reverser doors toward the “deployed” position.

Forward movement of the reverser carriage releases a normally-closed limit-switch (stow switch), which transmits a signal to illuminate an amber UNLOCK annunciator in the cockpit.

Completion of the deploy cycle causes the mechanism idler links to depress the arm of a limit switch (deploy switch), in the aft end of the mechanism support. This transmits a signal causing a white DEPLOY annunciator to illuminate in the cockpit.

During the deploy cycle, the thrust reverser feedback system restricts the engine throttle linkage to the “idle” position. Upon completion of the deploy cycle, the throttle linkage is released. This allows the pilot to move the reverser levers further back, driving the throttle linkage from idle to any selected power setting (up to the limited reverse power) to provide the required degree of reverse thrust.

Upon completion of the thrust reverser cycle, the reverser levers are rotated back to the “stow” position. This reverses the mechanism actuation, returning the reverser doors to the “stowed” and “locked’ position. During the stow cycle, the DEPLOY, UNLOCK and ARM annunciators in the cockpit extinguish, in that sequence.

NOTES

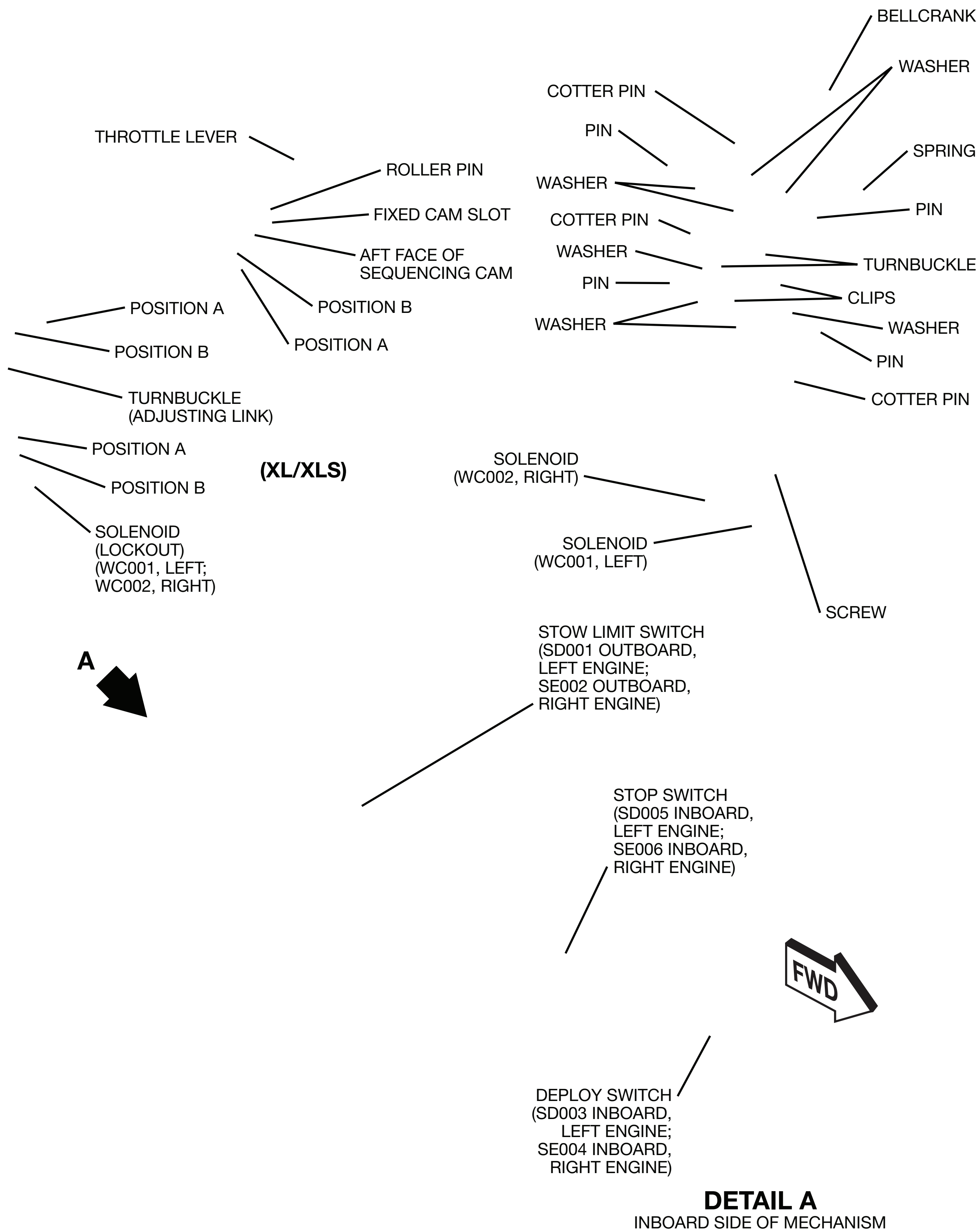


Figure 71-55. Thrust Reverser Electrical Components

Normal Deploy Cycle

Initial aft movement of a thrust reverser control lever to the "deploy" position, actuates a switch and the following functions occur (Figure 71-55):

- The airplane hydraulic open center bypass valve actuates, to pressurize the airplane system. The HYD PRESS annunciator illuminates.
- The hydraulic isolation valve is energized to allow hydraulic pressure to flow to the thrust reverser system. The ARM annunciator illuminates to indicate thrust reverser hydraulic pressure.
- The deploy solenoid on the thrust reverser control valve is energized. The UNLOCK annunciator illuminates as soon as the thrust reverser doors are unlocked and deployment begins.

When the doors are fully deployed, the following conditions exist:

- The DEPLOY, UNLOCK and ARM annunciators are illuminated.
- The HYD PRESS annunciator is illuminated on the annunciator panel.
- The thrust reverser lockout solenoid is energized to permit lever movement aft. Power to operate the solenoid is supplied by a transistorized circuit on a printed circuit board in the thrust reverser junction box.

At the time that power is initially supplied to the solenoid, 28 VDC ensures operation. However, continued operation at 28 VDC would overheat/burn out the solenoid. The initial 28 VDC tapers off to approximately 6 VDC to hold the solenoid energized when the thrust reverser is deployed.

Normal Stow Cycle

In the normal stow cycle, the thrust reverser control lever is moved to the forward position. This returns the control switch to the "stow" position and the following functions occur:

- The stow solenoid on the thrust reverser control valve is energized. The DEPLOY annunciator extinguishes when the thrust reverser starts to move to the "stow" position.
- When the thrust reverser reaches the stowed position and locks:
  - The UNLOCK annunciator goes out
  - The thrust reverser hydraulic system depressurizes
  - The ARM annunciator goes out
  - The airplane HYD PRESS annunciator on the annunciator panel goes out

Emergency Stow

The emergency stow switches are on the glareshield fire warning switch panel for stowing of the thrust reverser, in case they are inadvertently deployed in flight. This emergency system bypasses the normal system and provides an override command signal to "stow" from electrical power (through the opposite thrust reverser circuit breaker).

Electrical Sequence

Operation of the thrust reverser (for each engine) is controlled by a thrust reverser lever, mounted on the engine throttle control lever. The throttle lever must be in the "idle" position before the thrust reverser lever can be moved. Initially. the thrust reverser is moved to the "deploy" position. The thrust reverser lever cannot be moved beyond the deploy position (to increase engine thrust) until the thrust reverser is fully deployed. The switch on the thrust reverser must be operated to actuate a solenoid lock on the throttle lever.

Both reverser systems deploy only when either one or both landing gear squat switches are actuated.

OPERATION

BEFORE STARTING ENGINES

1. BATT Switch—ON
2. GEN Switches—ON (OFF if GPU start)
3. PARK BRAKE—SET
4. FLOOD Lights—FULL BRIGHT (night only)
5. Exterior Lights—AS REQUIRED
   a. GND REC—ON
   b. NAV Lights—ON (during night operations)
6. Annunciators—CHECKED

STARTING ENGINES

NOTE

Either engine may be started first. If the door is secured before battery start initiation, it is recommended that the left engine be started first. Spool up is slightly faster due to less line loss, since the battery is on the left side of the tail cone compartment. Due to foreign object ingestion hazard, the left engine must not be running during boarding or deplaning. If last minute boarding is anticipated, start the right engine first.

If the aircraft has been cold-soaked at temperatures below –10°C (14°F) and the engines have not been preheated, external power or warming the battery to –10°C (14°F) or warmer is recommended. This temperature can be checked with the battery temperature gauge. Proper battery warm-up may require extended application of heat to the battery.

Engine—START

1. Momentarily depressing an ENGINE START button causes:
   - The button and engine instrument floodlights to illuminate
   - Activates the fuel boost pump and the associated FUEL BOOST annunciator light
   - Commences engine rotation
2. At 8% N_2—THROTTLE TO IDLE. At 8-10% turbine RPM, lifting the cutoff latch and advancing the throttle to idle activates ignition and the associated igniter light, and initiates fuel flow.
3. Abort “start” if there no ITT rise within 10 seconds.

NOTE

Temperatures during ground start must not exceed 720°C. Temperatures exceeding this value must be investigated in accordance with the *Engine Maintenance Manual*.

4. Abort start if ITT rapidly approaches 720°C.
5. Abort start if there is no indication of N_1 rotation by 25% N_2.

Engine Instruments—CHECK NORMAL

1. Check engine instruments within limits. Check that starter has disengaged and that all annunciator lights are out except GND IDLE.
2. Make sure that fuel, oil, generator, and hydraulic annunciators are extinguished.
3. Make sure that the LO FUEL PRESS, FUEL BOOST and LO FUEL LEVEL annunciators are extinguished.

After light-off occurs (at approximately 42 to 46% turbine rpm), the starter relay opens, terminates ignition and fuel boost, and extinguishes the start button and instrument floodlights. During a battery start with the GEN switch ON, the generator comes online, extinguishing the GEN OFF annunciator (at approximately 42 percent turbine rpm).

With external power in use, the GEN switches can be "off" until starting is complete. It may not be possible to bring the generators online until the external power unit is removed. In any case, electrical equipment must not be turned on until both GEN OFF lights have extinguished.

NOTE

If automatic start sequencing does not terminate, the boost pump, ignition and associated lights remain "on". The starter, however, discontinues cranking due to speed sensing which governs at approximately 42 percent N_2. Depressing the STARTER DISENGAGE button terminates the automatic start sequence. This button is illuminated any time the PANEL LIGHT CONTROL master switch is ON.

An overvoltage protection system is provided during use of an external power unit (EPU). The control unit monitors the EPU voltage and deenergizes the external power relay if the voltage goes above 32–34 volts. External power cannot be reapplied to the airplane after an overvoltage condition has occurred, until the EPU power has been cycled "off" and back "on" with the power output supplying the nominal 28.5 VDC.

Other Engine—START

1 Repeat procedures in item 1.

CAUTION

The operating engine must be at idle for a cross generator start.

2. Engine Annunciators—EXTINGUISHED (except GND IDLE).
3. GPU—DISCONNECTED (if used).
4. GEN Switch—ON/CHECK DC AMPS/VOLTS.
   a. Left generator—Off, right generator—GEN, check left generator voltage, check right generator AMPS.
   b. Left generator—GEN, right generator—Off, check left generator AMPS, check right generator voltage.
   c. Left generator—GEN, right generator—GEN, check left generator AMPS, check right generator AMPS, check system voltage.

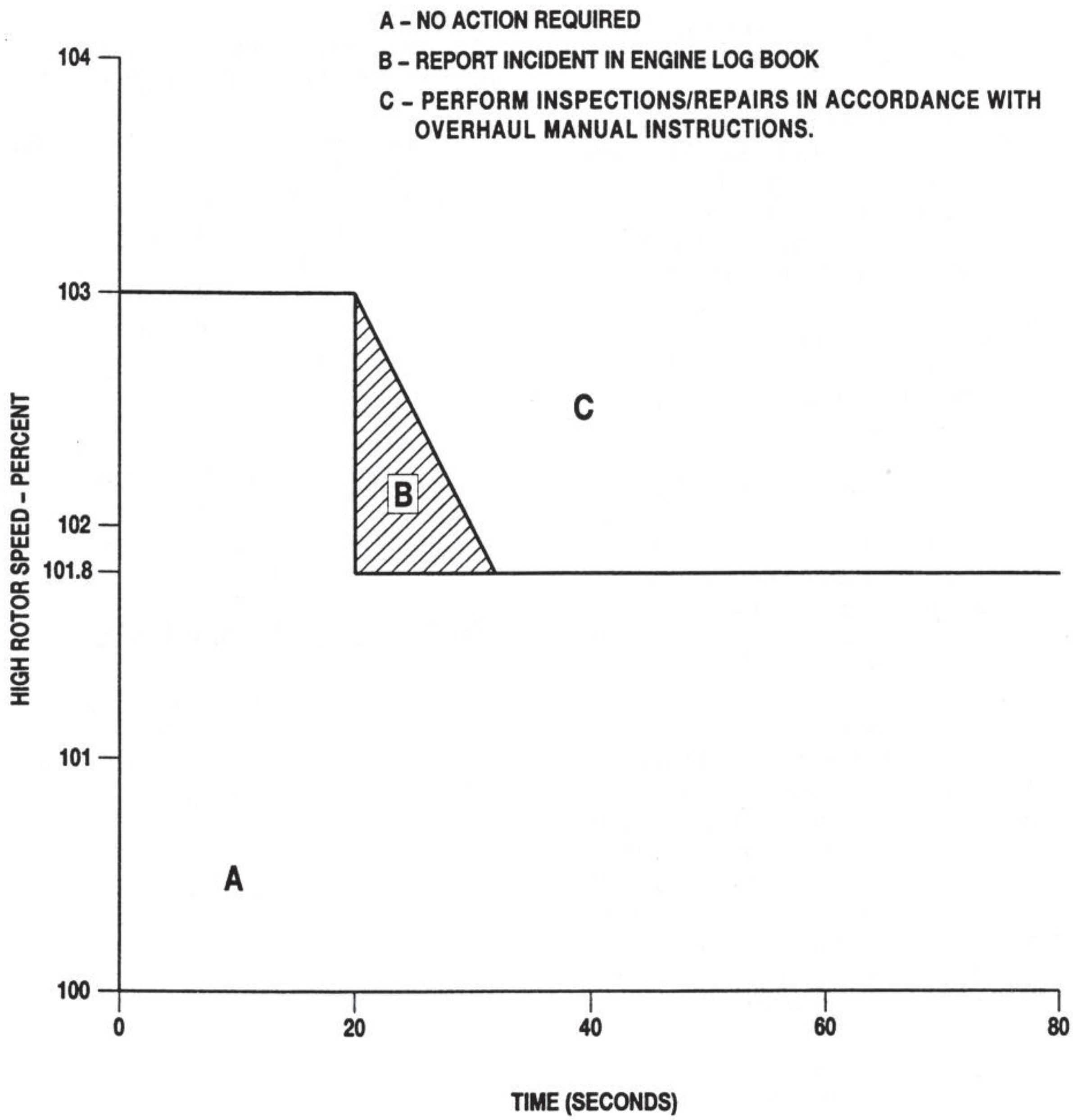


Figure 71-56. HP Rotor Overspeed Limits

LIMITATIONS

The following pages contain graphs of the limitations on ITT temperature, N_1, and N_2. Also included in the graphs are corrective actions, which must be performed for any such exceedance (Figure 71-56 through 71-61).

NOTES

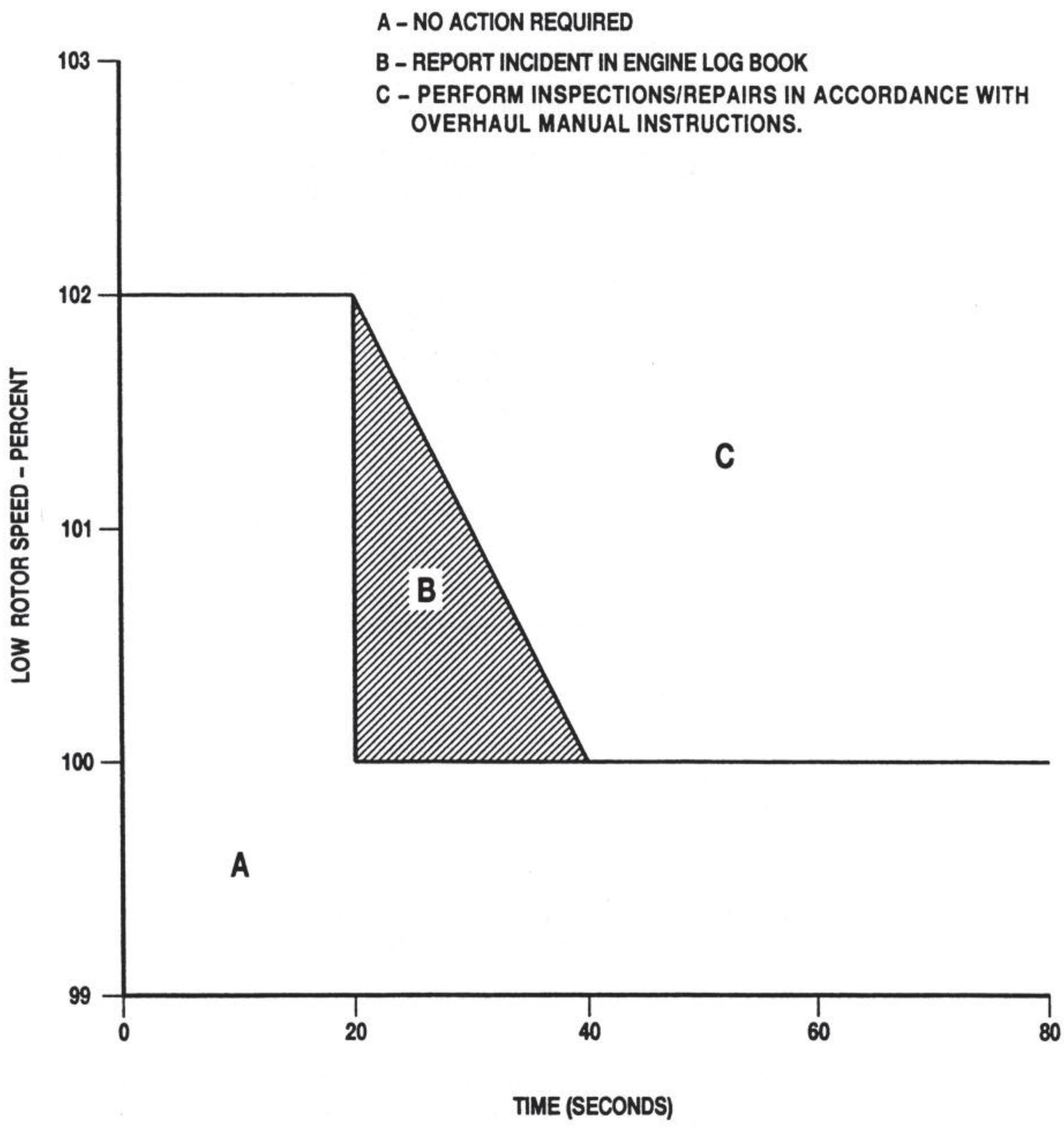


Figure 71-57. LP Rotor Overspeed Limits

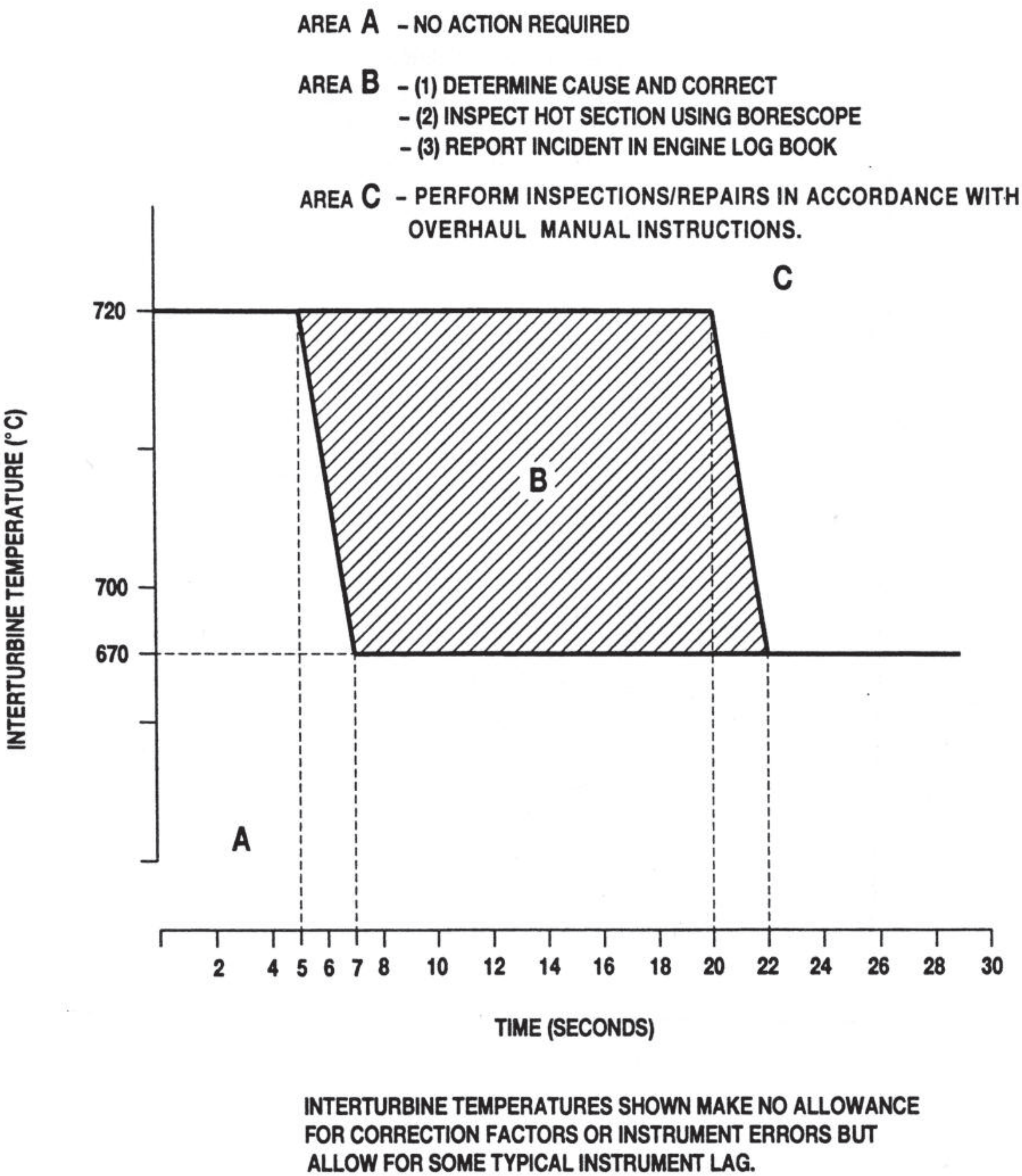


Figure 71-58. 545A Overtemperature Limits (Starting)

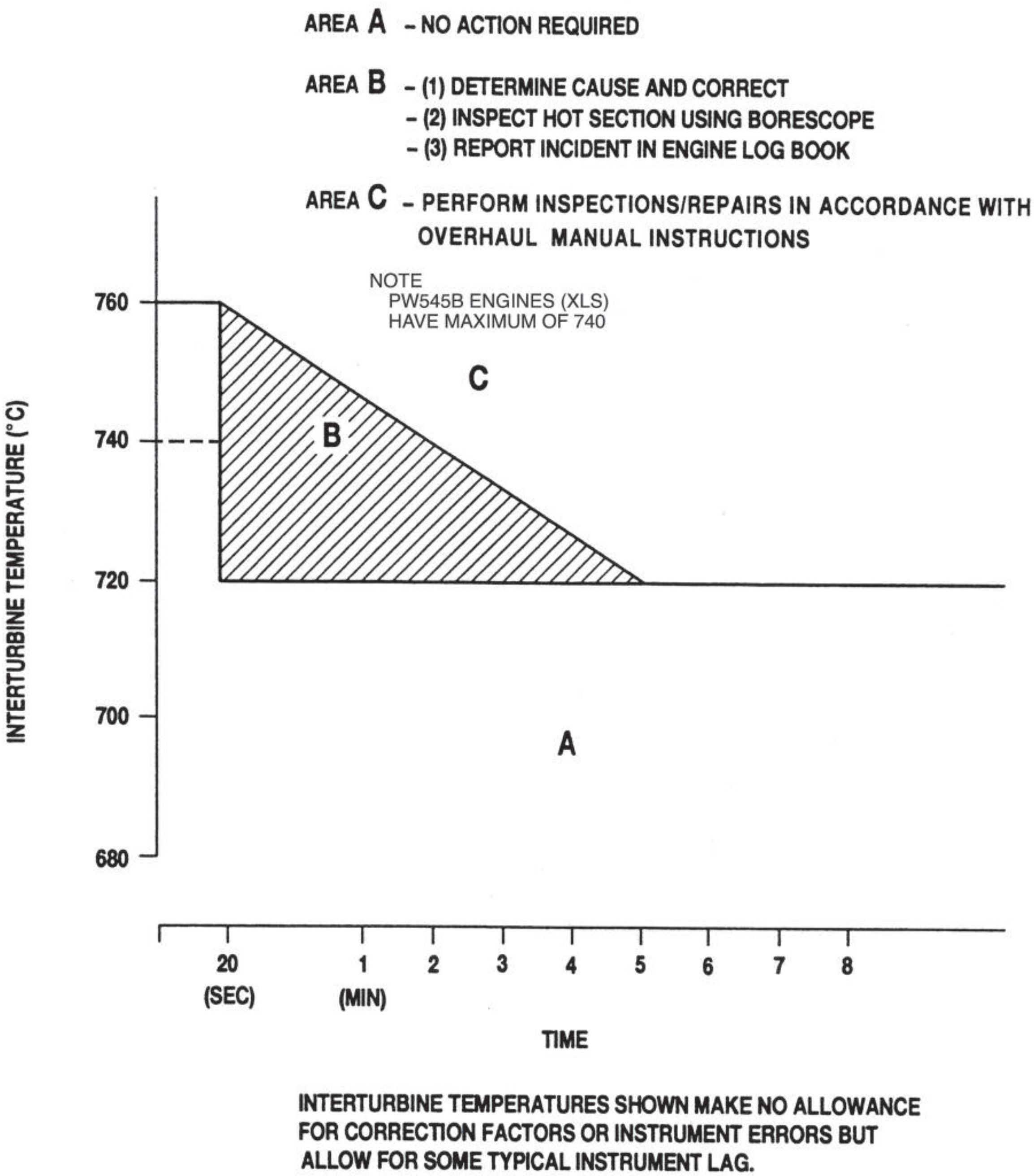


Figure 71-59. 545A Overtemperature Limits (Operating)

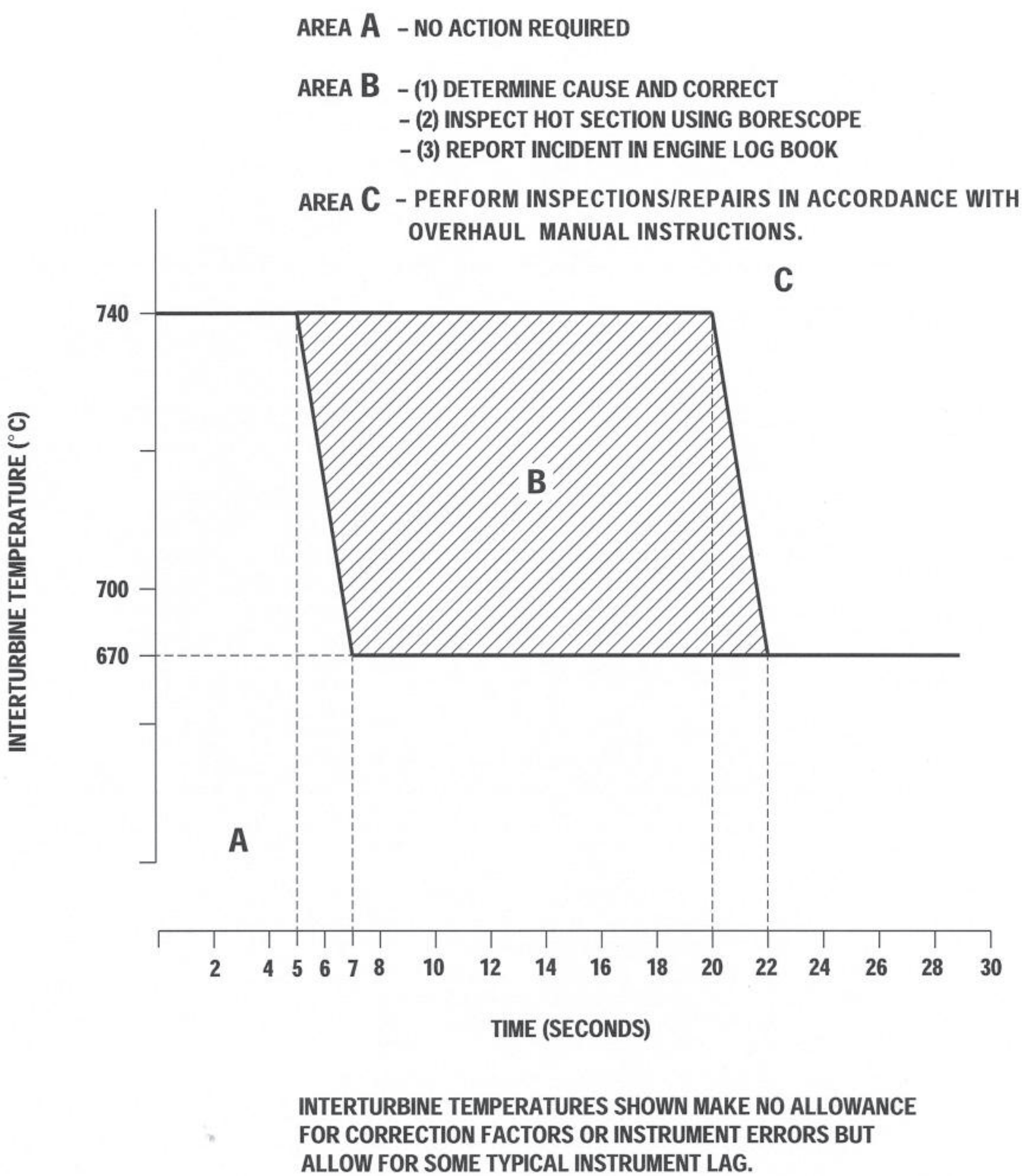
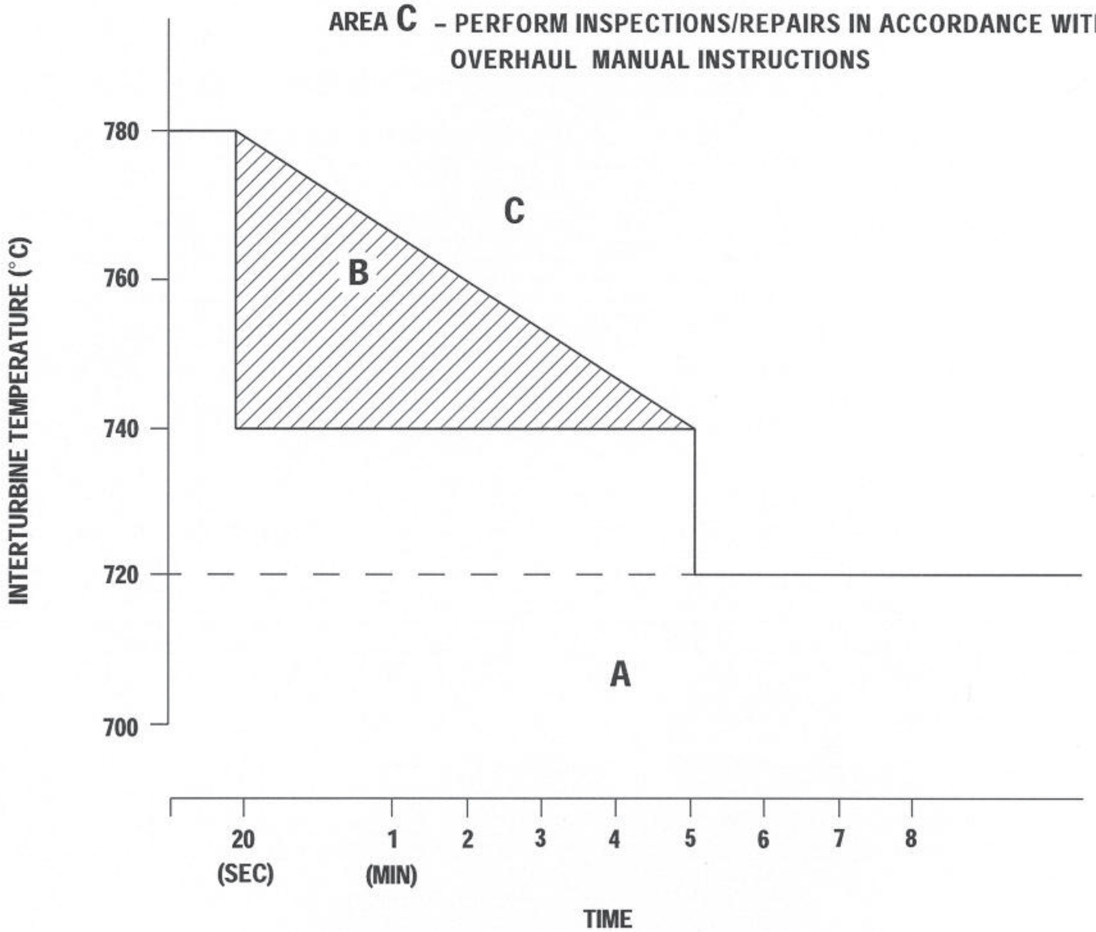


Figure 71-60. PW545B/C Overtemperature Limits—Starting Conditions Only

AREA A – NO ACTION REQUIRED

AREA B – (1) DETERMINE CAUSE AND CORRECT
– (2) INSPECT HOT SECTION USING BORESCOPE
– (3) REPORT INCIDENT IN ENGINE LOG BOOK

AREA C – PERFORM INSPECTIONS/REPAIRS IN ACCORDANCE WITH OVERHAUL MANUAL INSTRUCTIONS



INTERTURBINE TEMPERATURES SHOWN MAKE NO ALLOWANCE FOR CORRECTION FACTORS OR INSTRUMENT ERRORS BUT ALLOW FOR SOME TYPICAL INSTRUMENT LAG.

Figure 71-61. PW545B/C Overtemperature Limits—All Conditions Except Starting

APPROVED OILS

Type II Oils

- Aero Shell Turbine Oil 500
- Royco Turbine Oil 500
- Mobil Jet Oil II (Type II)
- Castrol 5000
- Exxon/Esso 2380 Turbo Oil

Third Generation Oils

- Mobil Jet Oil 254
- Aero Shell Turbine Oil 560

Engine Oil System Maintenance

- Check the oil level quantity at every preflight/daily and aircraft minor inspection (150 hrs). Oil level is always checked on the outboard sight glass when the engine is on the aircraft (Table 71-2).
- Change oil at every 900 ± 50 hours or every two years, whichever comes first.
- Remove and check the oil filter element for foreign matter or damage at every aircraft minor inspection (150 hrs). The oil filter is not cleanable.
- Replace the oil filter element with a new a element at every 900 ± 50 hours or every two years, whichever comes first.
- Check the aircraft cockpit oil impending bypass indicator lamp at every preflight/daily and aircraft minor inspection (150 hrs).
- Remove the chip collector and check for metallic debris at every aircraft minor inspection (150 hrs).
- Check the accessory gearbox pad seals for oil leaks at every aircraft minor inspection (150 hrs) (leak rates exceeding 5 cc/hr from any pad seal is unacceptable).

Table 71-3. OIL SYSTEM SPECIFICATIONS AND LIMITATIONS

Oil Tank Capacity	4.47 qts (5.8 L)
Normal Oil Pressure	45–145 psid @ 60% N_2
Min Transient Press.	0 psid for 20 sec 20 psid for 120 sec
Max Transient Press.	250 psid for 120 sec
Max Oil Consumption	1 Qt/10 hrs or 0.2 lbs/hr
Max Oil Temperature	121°C
Min Oil Temperature @ Idle @ 3767 lbs of Thrust @ 3804 lbs of Thrust	 –40°C 10°C 10°C

NOTES

PW545A ENGINE CALIBRATION RUN DATA SHEET

TAKEN FROM INSTALLED CALIBRATED AIRCRAFT EQUIPMENT

DATE:	A/C REG. NO.	ENGINE S/N:	ENGINE POSITION:
ENGINE TOTAL TIME SINCE NEW:		WIND DIRECTION/VELOCITY:	
OUTSIDE AIR TEMP: 0= √0=		PRESS. ALT:	
MAX N1 FOR AMBIENT CONDITIONS PER A/C FLIGHT MANUALS: %N1			

OBSERVED ENGINE PARAMETER (A/C INDICATOR)	SYNC. OFF / BLEED AIR OFF / ANTI-ICE OFF					
	NORMAL T/O	T/O - 2%	T/O - 4%	T/O - 6%	T/O - 10%	GROUND IDLE
N1						
N2						
(T4.5)						
FUEL FLOW						
OIL PRESS						
OIL TEMP						
CORRECTED ENGINE PARAMETER CALCULATION						
N1						
N2						
T4.5						

RUN DOWN TIME	
N1: (SECONDS)	N2: (SECONDS)

Figure 71-62. Engine Calibration—Run Data Sheet

ENGINE CALIBRATION AND PERFORMANCE CHECK

The initial engine ground calibration run establishes the installed N_1 rotor speed, N_2 rotor speed, and temperature relationship after installation of a new or overhauled engine (Figures 71-62 and Table 71-3). The curves plotted during the initial calibration run become the MASTER curves. Values plotted during future ground calibration runs must then be compared to the original MASTER curve in order to assess engine condition.

The performance check is used to determine engine condition and diagnose performance problems. This allows repair or replacement of an engine before potential problems become extensive and costly.

NOTES

Table 71-4. TEMPERATURE CORRECTION FACTOR SAMPLE TABLE

T(AMB) °C	0	√0	D
14	.9965	.9983	.95
15	1.0000	1.0000	0.00
16	1.0035	1.0014	–0.94
17	1.0069	1.0035	–1.88

PW545A ENGINE CALIBRATION AND PERFORMANCE CHECK CURVE, N1 vs. N2

DATE:	A/C REG. NO.	ENGINE S/N:	ENGINE POSITION:
ENGINE TOTAL TIME SINCE NEW:		ENGINE TOTAL TIME SINCE OVHL:	

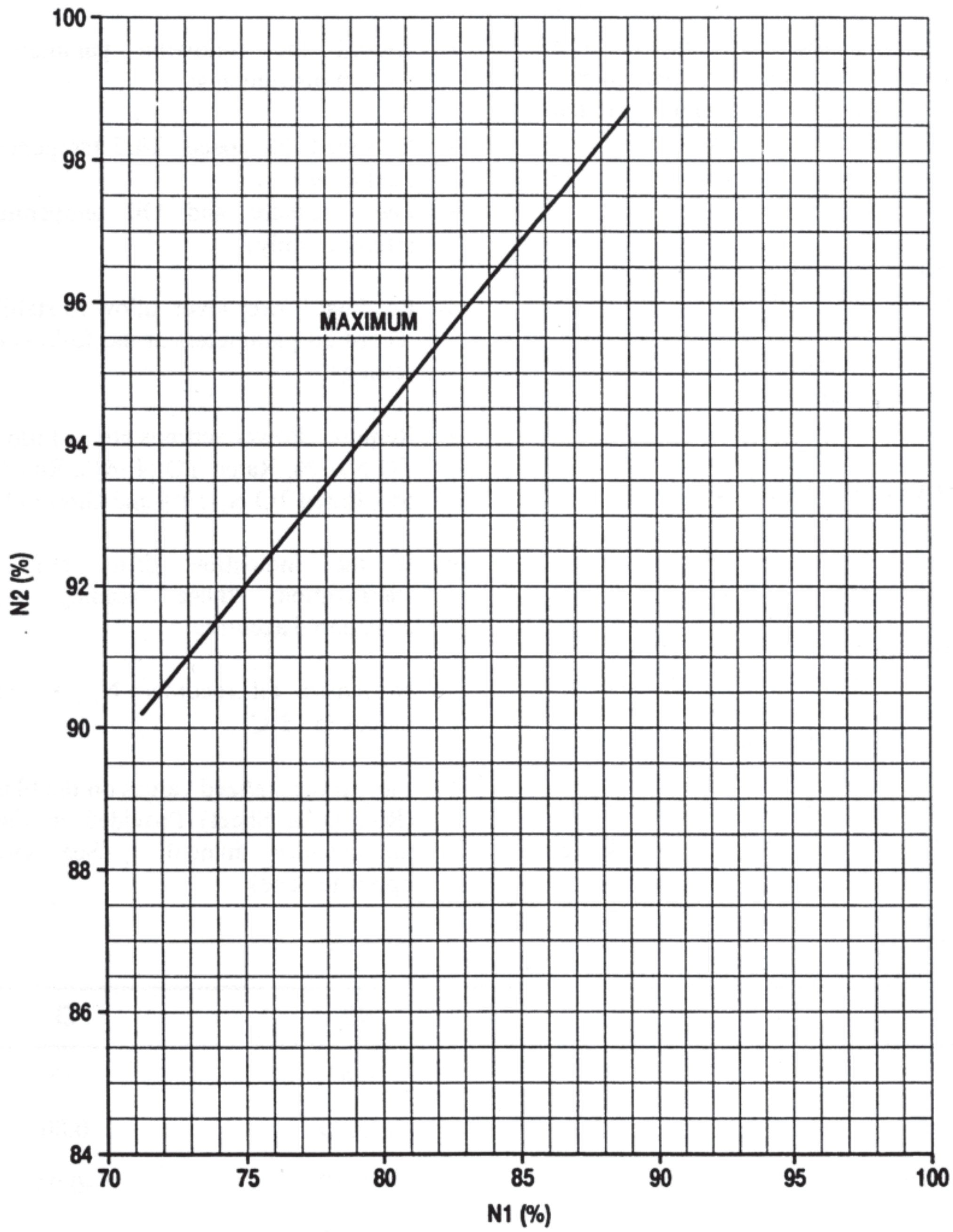


Figure 71-63. N_1 vs N_2 Performance Check Curve

Calibration MASTER Curves

The following procedure is used to produce the initial calibration curves (Figure 71-63 and 71-64):

1. Ensure that instrumentation is functioning correctly and has been accurately calibrated.
2. Position aircraft into the wind.
3. Refer to *Aircraft Flight Manual (AFM)* to obtain the normal rated take-off N_1% value for the particular day OAT.
4. Start the engine.
5. Make sure that all accessory loads and air bleeds are off during the test.
6. Advance the power lever to the “Normal Take Off” detent position and allow the engine to stabilize for at least 3 minutes.
7. Record the following parameters from cockpit instruments:
   - N_1 speed, N_2 speed, T4.5 temperature, and fuel flow
   - Fuel pressure and oil temperature (for reference only)
8. Readjust the power lever. Allow it to stabilize, and record all parameters at the following power settings. The required power settings are as follows:
   - Rated TO N_1-2%
   - Rated TO N_1-4%
   - Rated TO N_1-6%
   - Rated TO N_1-10%
   - Ground idle
9. If the indication drifts slightly after stabilization, these readings can be considered accurate.
10. Normalize all recorded N_1, N_2, and T4.5 values to 15°C.
11. Plot all normalized values on the blank graph (Run Data Sheet) Provided in the engine maintenance manual. (see Figure 71-51).

NOTES

PW545A ENGINE CALIBRATION AND PERFORMANCE CHECK CURVE, N1 vs. T4.5

DATE:	A/C REG. NO.	ENGINE S/N:	ENGINE POSITION:
ENGINE TOTAL TIME SINCE NEW:		ENGINE TOTAL TIME SINCE OVHL:	

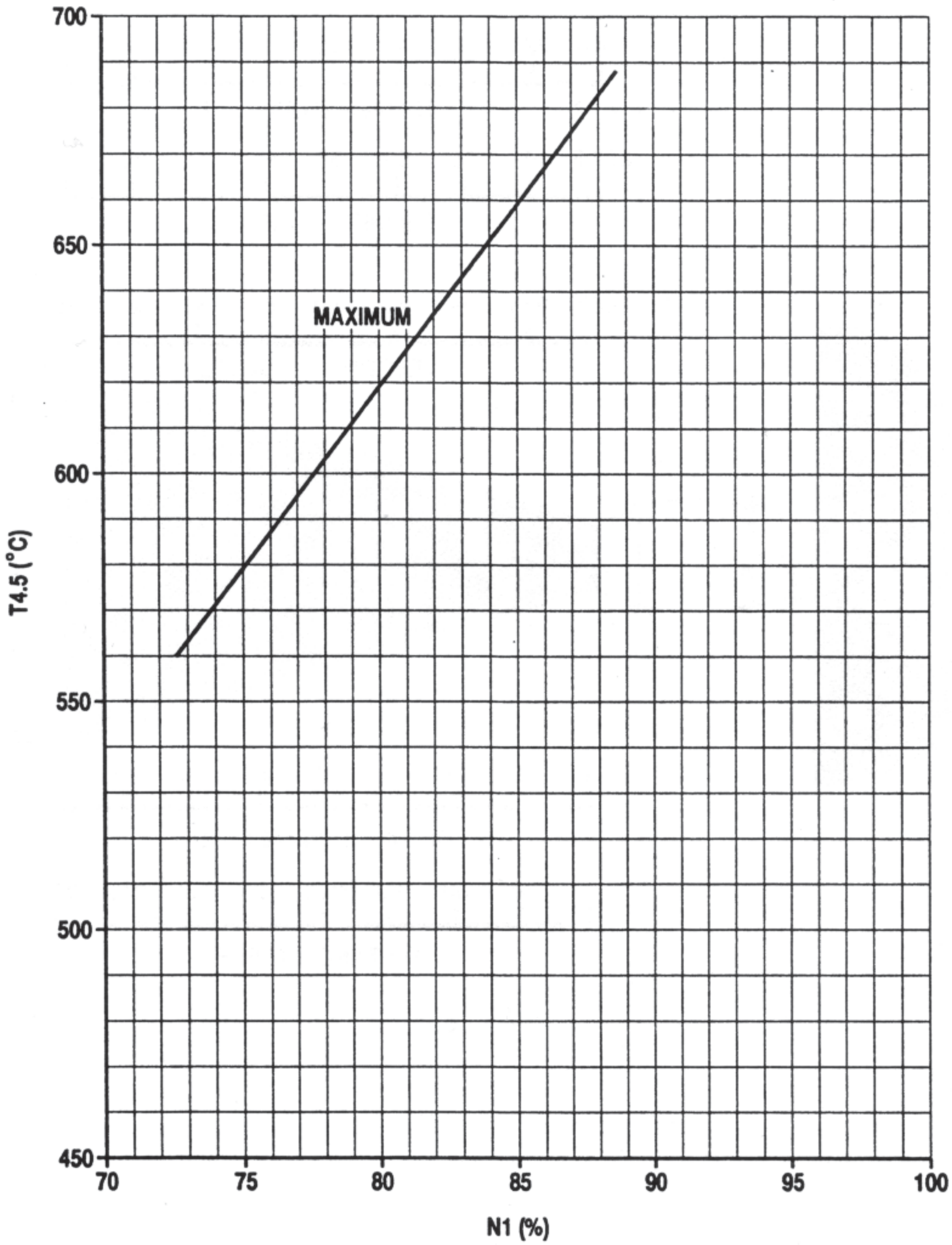


Figure 71-64. N_1 vs T4.5 Performance Check Curve

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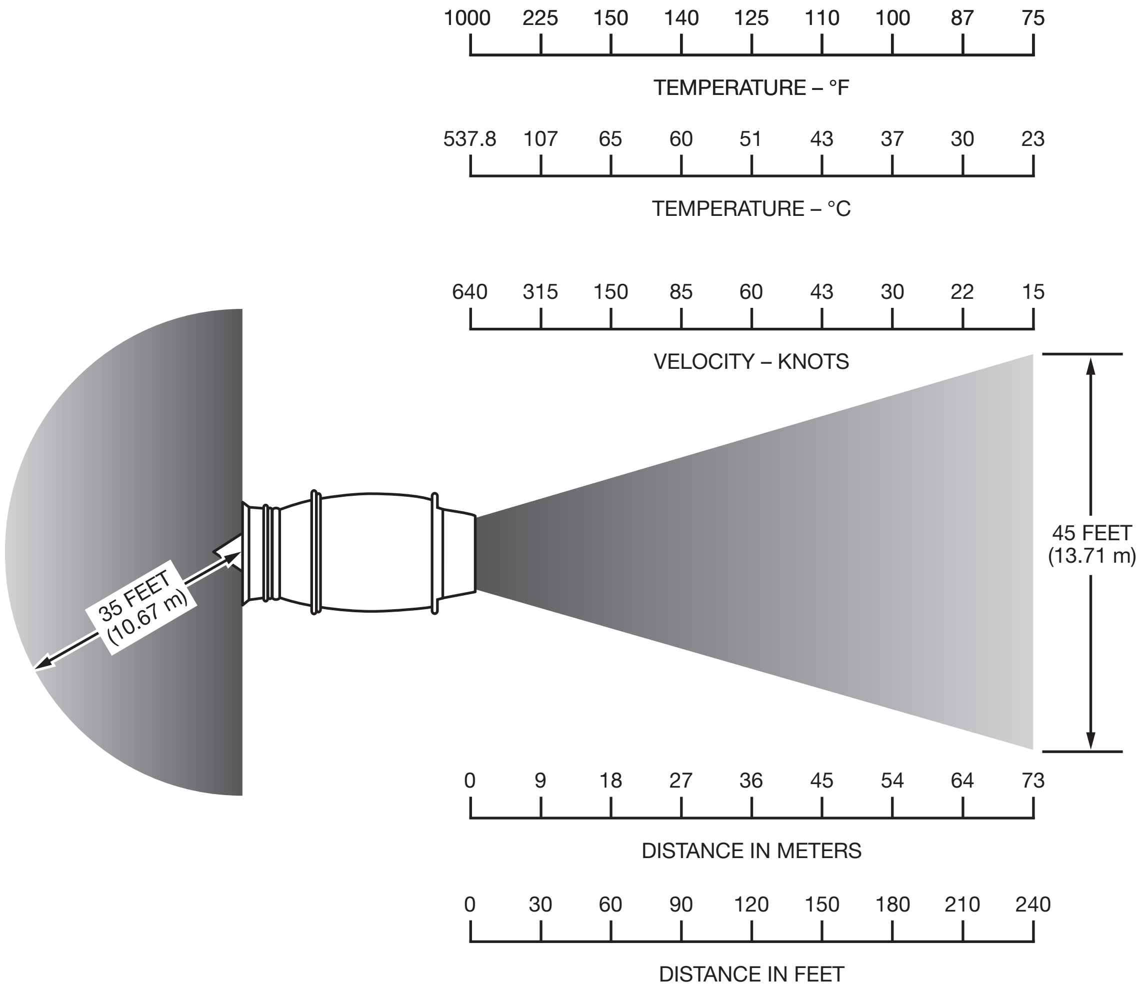


Figure 71-65. Engine Danger Areas

ENGINE GROUND SAFETY PRECAUTIONS

Great care must be taken when handling or working on turbine powered aircraft, to avoid injury to personnel, damage to property, or damage to the engine. Areas of extreme danger are the air intake and exhaust jet wake.

Air Intake

CAUTION

The air intake is capable of generating sufficient suction to pull a person into the intake ducting. The potential danger of this suction cannot be overemphasized. In an idling engine it may be sufficient to cause ingestion of eyeglasses, small tools, rags, and small objects in general (Figure 71-65).

Exhaust Jet Wake Area

CAUTION

The exhaust jet wake must be approached with extreme caution as extensive damage can be incurred from the high temperature and high velocity gases. At high engine speeds, the jet wake may propel loose dirt, sizeable stones, sand, and debris over a considerable distance. The high temperatures in the wake are sufficient to deteriorate asphalt; for this reason, concrete is recommended for run-up areas.

CAUTION

Occasionally, when starting a jet engine, an accumulation of fuel in the exhaust tail pipe can be blown out as long streams of flame. Personnel must observe proper fire precautions and move all flammable material to a safe distance.

CAUTION

Exposure to gases in the jet wake must be avoided. Failure to do so may result in respiratory irritations and a burning sensation in the eyes. Particular care must be taken to avoid exposure to gases in confined spaces where the gas concentration may be high.

Cool Down

CAUTION

After engine operation, work that requires contact with the exhaust tailpipe must not be carried out for at least one-half hour. Heat resistant gloves must be worn, if work in this area is required immediately after shutdown.

Jet Fuel and Lubricating Oil

CAUTION

All jet fuels and lubricating oils have and injurious effect on the skin. Precautions must be taken to avoid contact as much as possible.

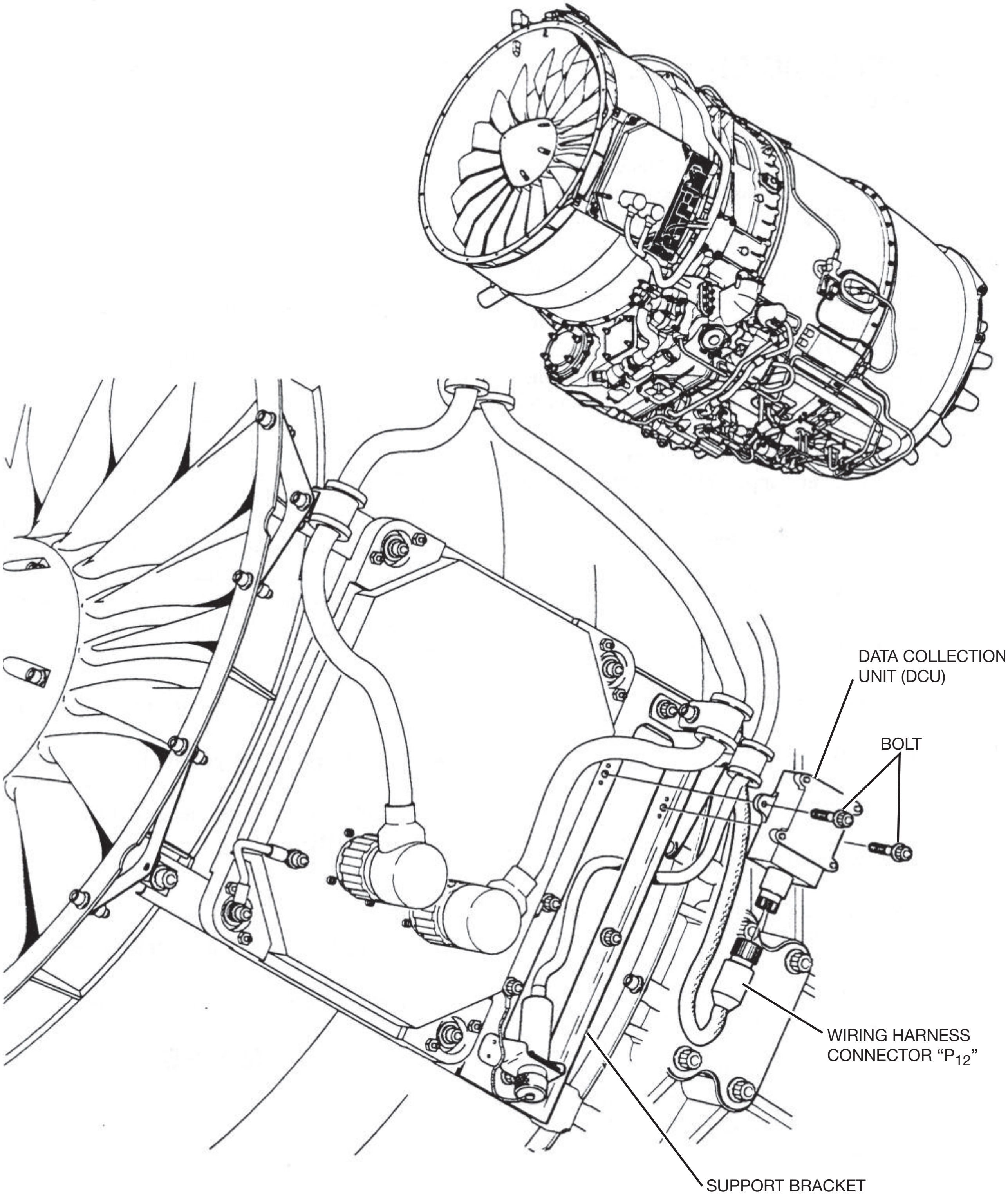


Figure 71-66. Data Collection Unit (DCU)

DIAGNOSTICS

ENGINE DIAGNOSTIC SYSTEM

The engine diagnostic system (EDS) provides troubleshooting tools to resolve engine and airframe related EEC system problems. The EEC is the primary element in the EDS. The EEC generates and calculates data. The processed fault and life cycle data is transmitted across a UART link and stored in the data collection unit (DCU) (Figure 71-66).

Data Collection Unit

The DCU functions as a memory bank in which the EEC can record trend data, fault codes, and exceedances. This information can then be downloaded to a laptop computer using a Pratt and Whitney ground based software (GBSLite) package. If a fault or exceedance has occurred within the engine control system (ECS), then a manually resettable ECS FAULT bite indicator (in the accessory compartment) gives an indication. In order to download the information from the DCU, the EEC must be installed and initialized. Two points for download are available. One point is a connector on the engine, just below the DCU. The other is a connector at the aft end of the pedestal, on the same side as the respective engine. There are two connectors for data download in the pedestal: one for each engine.

The DCU also stores engine and component information. All engine serialized parts are logged into the DCU as well as the engine serial number. Aircraft registration, unit number, and location of the engine on the aircraft are also stored. All of these parameters can be updated using the GBS and a laptop.

CAUTION

Do not move DCUs from one engine to another. Exchanging DCUs from one engine to another causes the incorrect trims to be loaded on the engine. If this occurs, remove the DCU and reinitialize.

NOTES

TERMS AND ABBREVIATIONS

Table APP-1. TERMS AND ABBREVIATIONS

ABBREVIATION	DESCRIPTION
ASCB	Avionics standard communications bus
AGRAS	Air/ground radiotelephone automated service
AHRS	Attitude heading reference system
ADC	Air data computer
AP	Autopilot
AC	Alternating current
AM	Amplitude modulation
ANT	Antenna
AWG	American wire gauge
ADF	Automatic direction finding
AOA	Angle-of-attack
ANN	Annunciator
AVC	Automatic volume control
AMP	Amplifier
BIT	Built-in test
BFO	Beat frequency oscillator
bps	Bits-per-second
Baud	Baudot
CPLR	Coupler
CDU	Control display unit
CAP	Capacitor
CRT	Cathode ray tube
CLK	Clock
DTU	Data transfer unit
DC	Direct current
DH	Decision height
dB	Decibel
DG	Directional gyro

Table APP-1. TERMS AND ABBREVIATIONS (Cont)

ABBREVIATION	DESCRIPTION
DBLR	Doubler
DEMOD	Demodulate
DADC	Digital air data computer
DC	Display computer
DMU	Data monitor unit
DME	Distance measuring equipment
EM	Electromagnetic
ESD	Electrostatic sensitive device or electrostatic discharge
EDS	Electronic display system
EFIS	Electronic flight instrument system
EADI	Electronic attitude direction indicator
EHSI	Electronic horizontal situation indicator
F (FREQ)	Frequency
FM	Frequency modulation
FMS	Flight management system
GND	Ground
GA	Gauge
Hz	Hertz
HF	High frequency
HDG	Heading
HSI	Horizontal situation indicator
IC	Integrated circuit
IMPD	Impedance
IRS	Inertial reference system
INS	Inertial navigation system
IAC	Integrated avionics computer
K	Kilo
LED	Light emitting diode
LF	Low frequency

Table APP-1. TERMS AND ABBREVIATIONS (Cont)

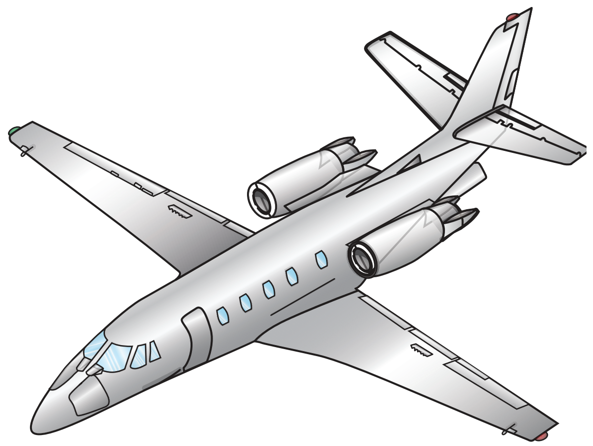
ABBREVIATION	DESCRIPTION
LP	Low pass
LRU	Line replaceable unit
M	Mega
u	Micro
mA	Milliamperes
m	Milli
MIC	Microphone
MADC	Micro air data computer
MODEM	Modulator demodulator
MSU	Mode selector unit
MDA	Minimum decision altitude
NEG	Negative
NO	Normally open
NPN	Negative positive negative
NOM	Nominal
NC	Normally closed
NAV	Navigation
NMU	Navigational management unit
NDB	Navigational data base
OSC	Oscillator
PFD	Primary flight display
PFS	Primary flight software
PM	Pulse modulation
PCB	Printed circuit board
PK	Peak
PLL	Phase locked loop
PTT	Push to talk
POST	Power on self-test
PNP	Positive negative positive
POS	Positive
PREAMP	Preamplifier

Table APP-1. TERMS AND ABBREVIATIONS (Cont)

ABBREVIATION	DESCRIPTION
POT	Potentiometer
PRF	Pulse recurrence frequency
PRI	Primary
PWR	Power
PROM	Programmable read-only memory
RCVR	Receiver
RAD	Radian
RADAR	Radio direction and ranging
RS	Radio Standard
RSC	Radio-set control
RTU	Radio tuning unit
RF	Radio frequency
RMU	Radio management unit
RPU	Receiver processor unit
RSB	Radio standard bus
RVT	Resistive variable transducer
RLY	Relay
RFR	Radio frequency radiation
SW, S/W	Switch
SHLD	Shield
SPKR	Speaker
SERVO	Servomechanism
SRU	Service replaceable unit
SIG	Signal
TTL	Transistor transistor logic
TXMTR	Transmitter
TCAS	Traffic collision alerting system
XPNDR	Transponder
Q	Transistor
TERM	Terminal
UHF	Ultra high frequency

Table APP-1. TERMS AND ABBREVIATIONS (Cont)

ABBREVIATION	DESCRIPTION
UTC	Universal time coordinated
VAC	Volts AC
VOR	VHF omni range
VHF	Very high frequency
VLF	Very low frequency
V	Volts
VNAV	Vertical navigation
VG	Vertical gyro
WOW	Weight on wheels
WOG	Weight on ground



CITATION 560XL/XLS/XLS+ MAINTENANCE SCHEMATIC MANUAL

REVISION 0.2

Courses for the 560XL/XLS/XLS+ Series are taught at the following FlightSafety learning center:

Wichita Cessna Maintenance Learning Center
1962 Midfield Road
Wichita, KS 67209
(316) 220-3250
(800) 491-9796
FAX (316) 220-3275

INSERT LATEST REVISED PAGES, DESTROY SUPERSEDED PAGES

LIST OF EFFECTIVE PAGES

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Recision .....0.2 ....October 203

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Page No.	*Revision No.
Cover	0.1
Copyright	0.1
i	0.1
ii—iv	0
v—vi	0.1
21-1—21-5	0.1
21-6	0.2
24-1—24-5	0.1
24-6	0.2
24-7—24-11	0.1
24-1—24-11	0.1
24-12—24-17	0.2
24-18—24-20	0.1
24-21	0.2
24-22	0.1
24-23	0.2
24-24—24-26	0.1
24-27	0.2
24-28—24-30	0.1
24-31	0.2
24-32—24-34	0.1
24-35	0.2
24-36	0.1
24-37	0.2
24-38	0.1
24-39	0.2
24-40	0.1
27-1—27-8	0.1
28-1—28-6	0.1
32-1—32-10	0.1
34-1—34-4	0.1
36-1—36-4	0.1
49-1—49-4	0.1
71-1—78-4	0.1

*Zero in this column indicates an original page.

NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Pilot Manuals and Maintenance Manuals. It is to be used for familiarization and training purposes only.

At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

ILLUSTRATIONS

ILLUSTRATIONS

ILLUSTRATIONS

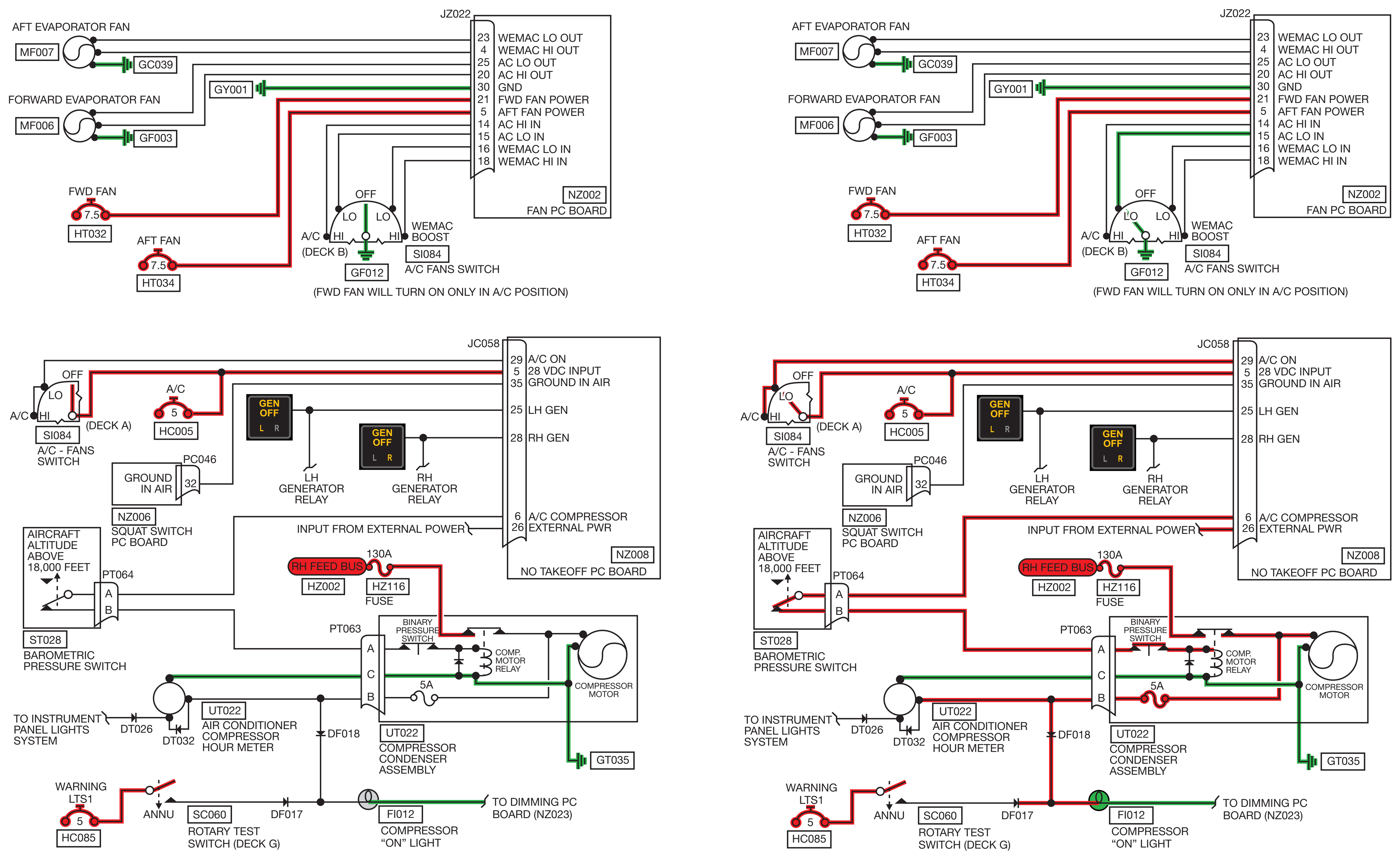


Figure 21-1. Vapor Cycle Air Conditioning - Off Selected and Low Selected with Aircraft on Ground with External Power Connected

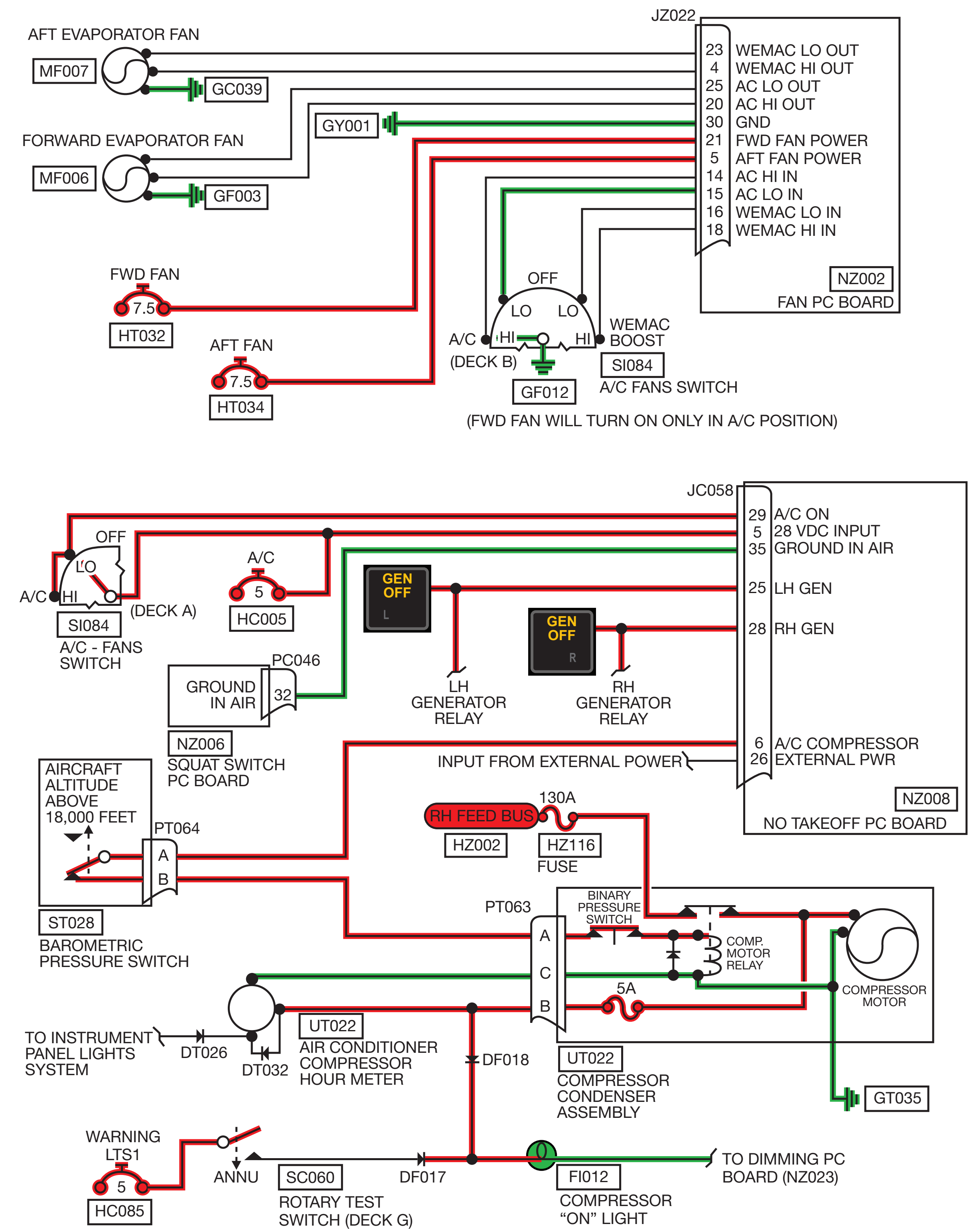


Figure 21-2. Vapor Cycle Air Conditioning - High Selected with Aircraft in Flight with Both Generators Online

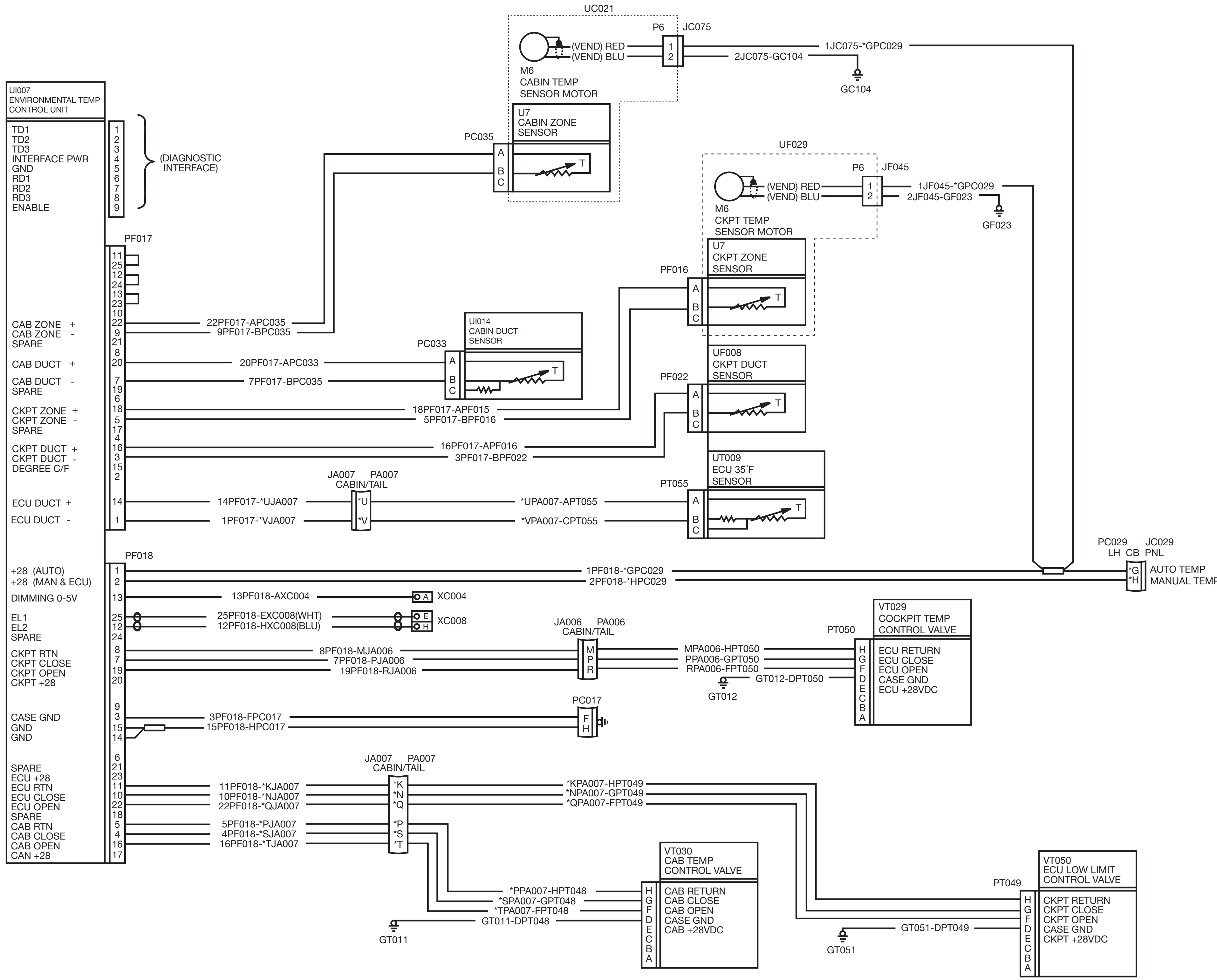
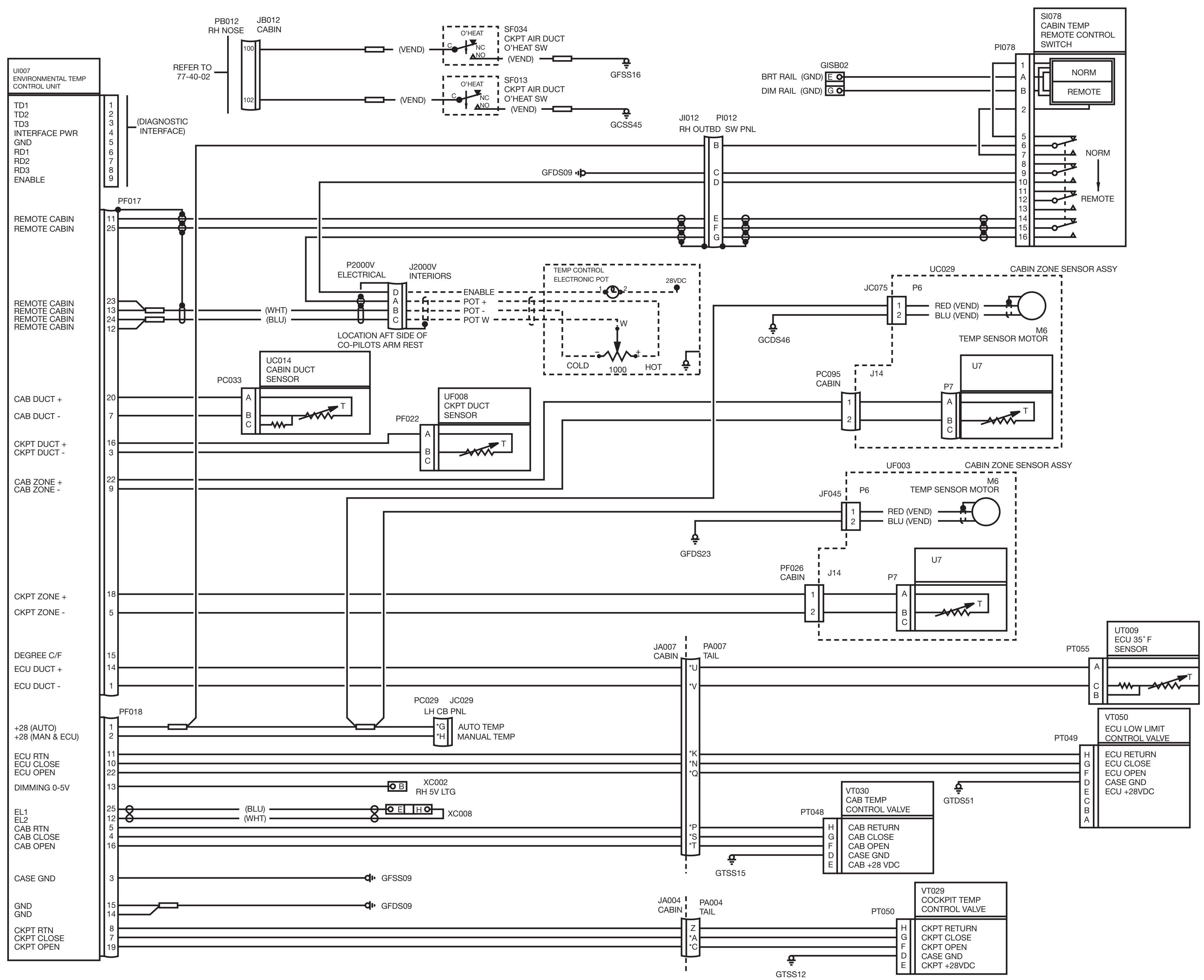


Figure 21-3. Temperature Control System - Units 560-5188 and Subsequent and Prior Aircraft with SB560XL-21-09 Incorporated



NOTES:
1. ALL WIRE 22AWG UNLESS OTHERWISE NOTED.

2.REFER TO 91-10-02 FOR GROUND INFORMATION.

Figure 21-4. Temperature Control System - Units 6001 and Subsequent

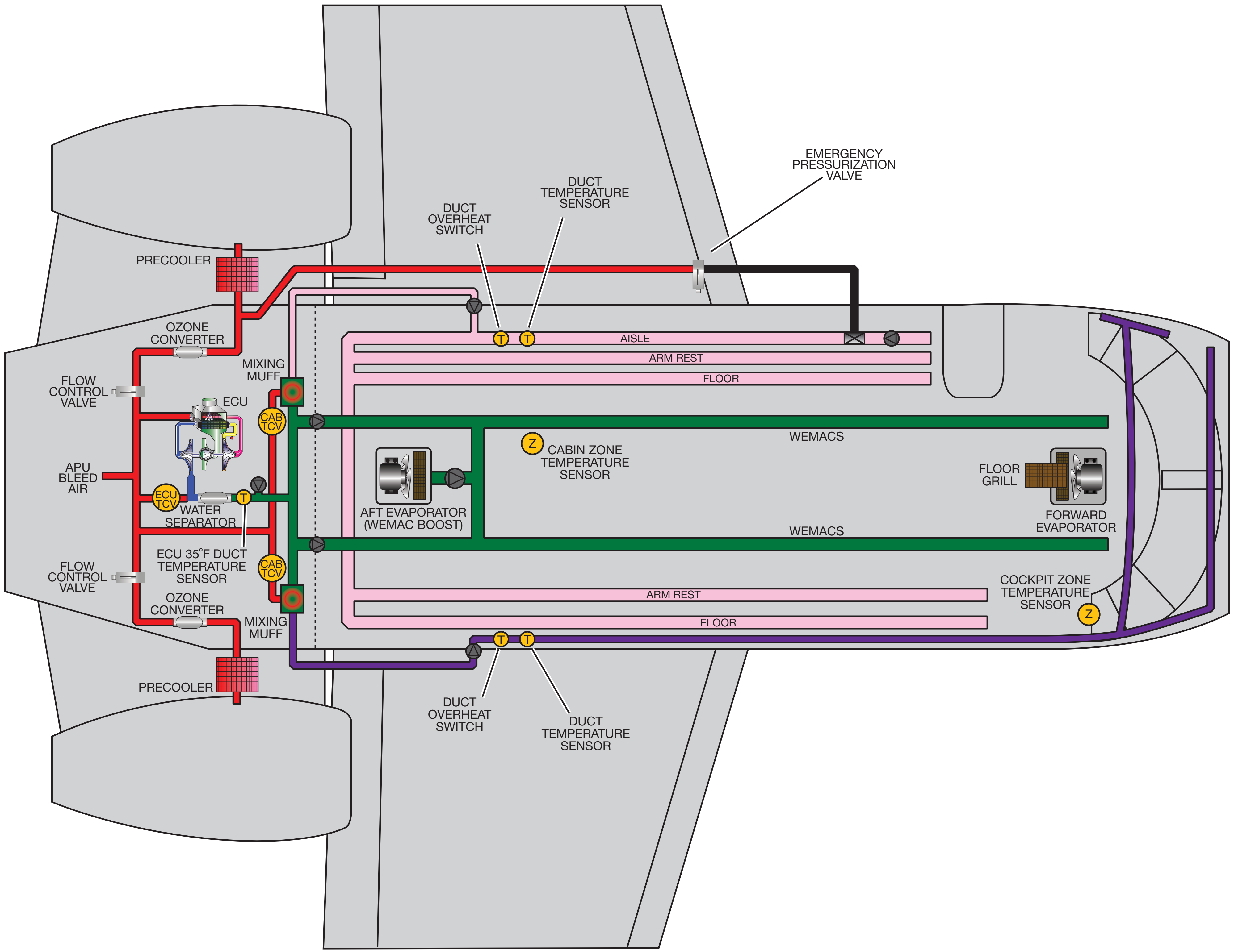
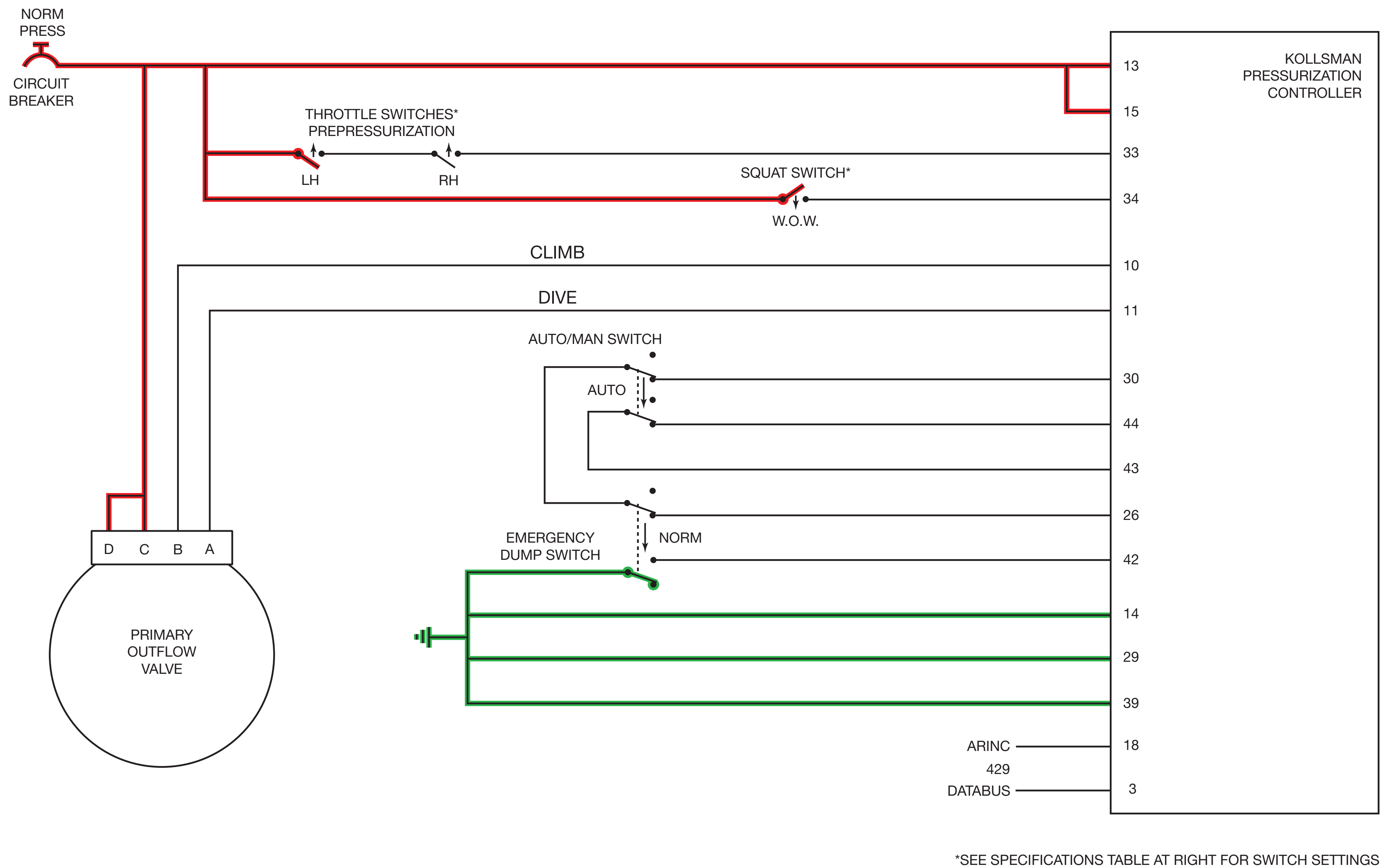


Figure 21-5. Air Distribution Schematic



CESSNA AIRCRAFT KOLLSMAN SYSTEM SPECIFICATIONS TABLE
CJ,CJ1,CJ2,CJ3 SQUAT SWITCH: RH PREPRESSURIZATION: ABOVE 85% N_2
ENCORE, ENCORE+ ,BRAVO SQUAT SWITCH: LH PREPRESSURIZATION: ABOVE 85% N_2
EXCEL, XLS, XLS+ SQUAT SWITCH: LH PREPRESSURIZATION: ABOVE 62° TLA

Figure 21-6. Kollsman Pressurization System

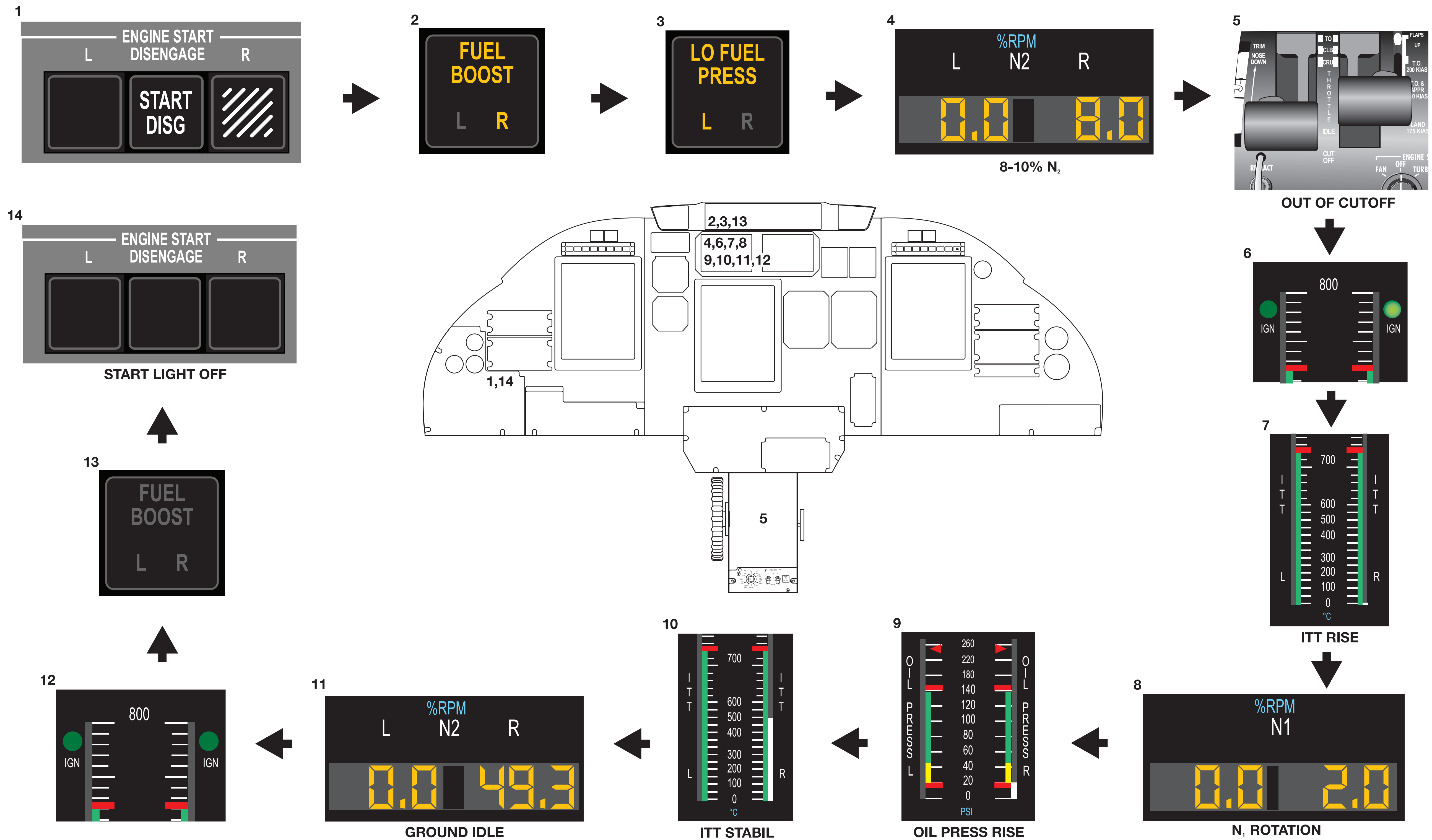


Figure 24-1. XL Start Procedures - 5001-5268

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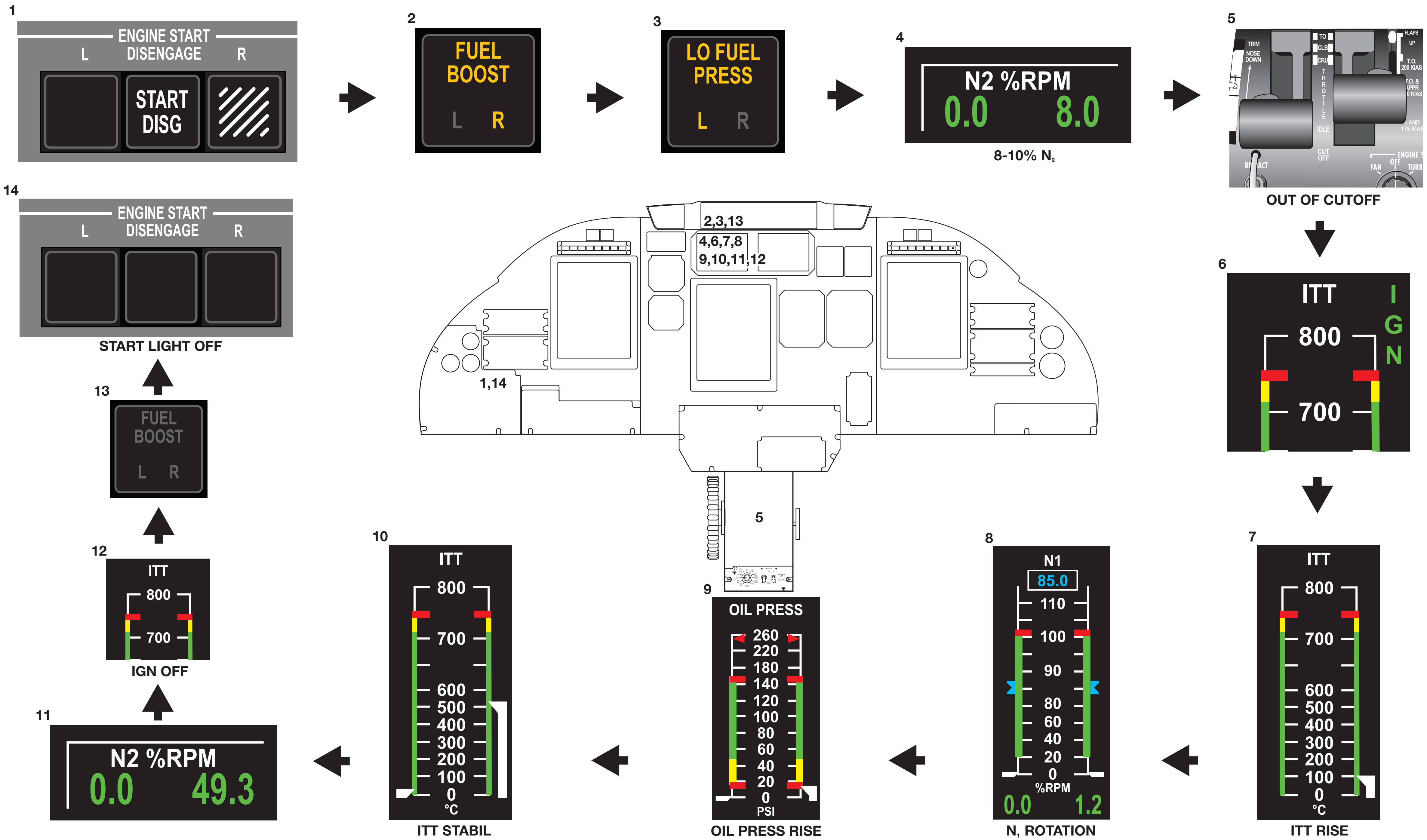


Figure 24-2. XL/XLS Start Procedures 5269 and Subsequent

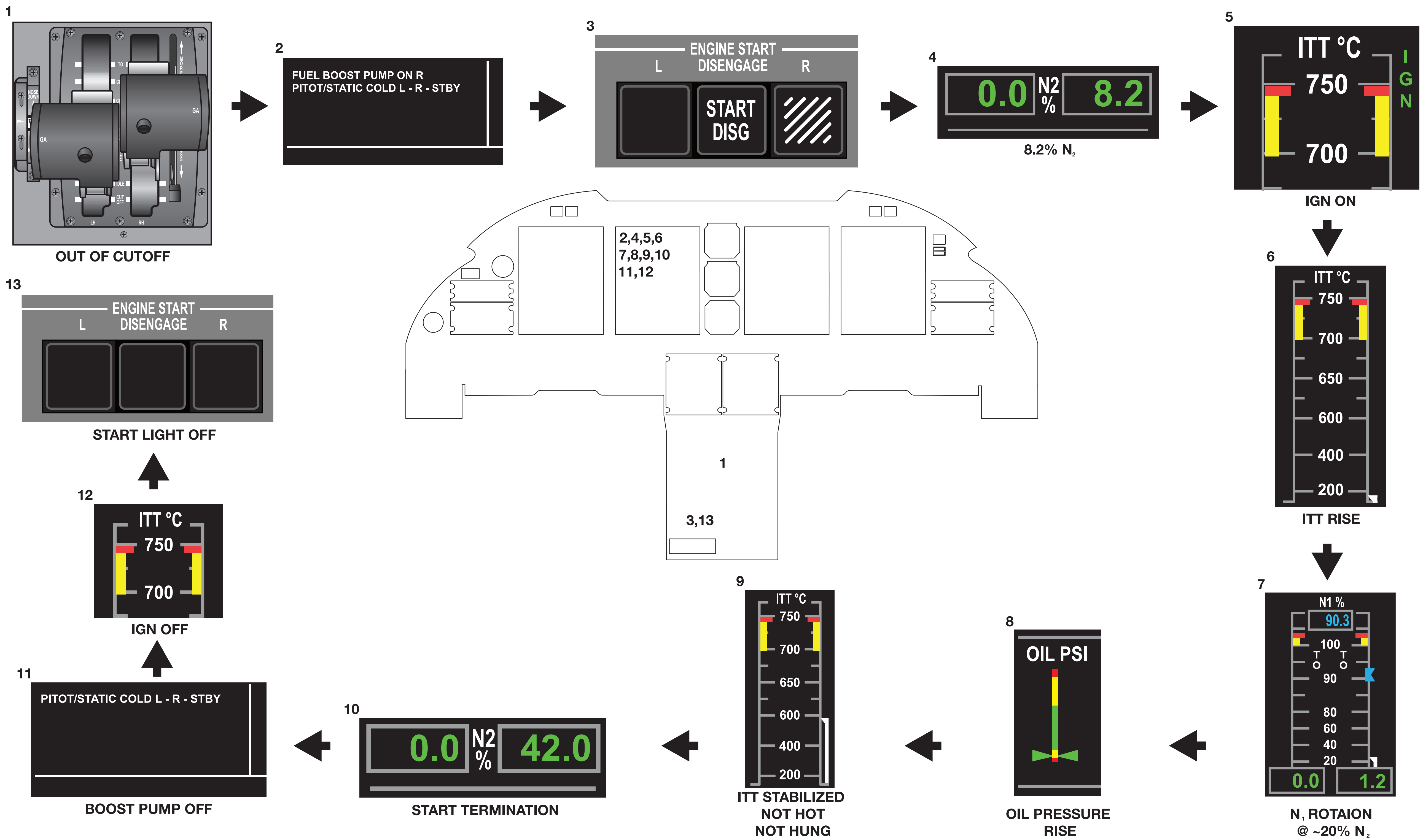


Figure 24-3. XLS+ Start Procedure “A”

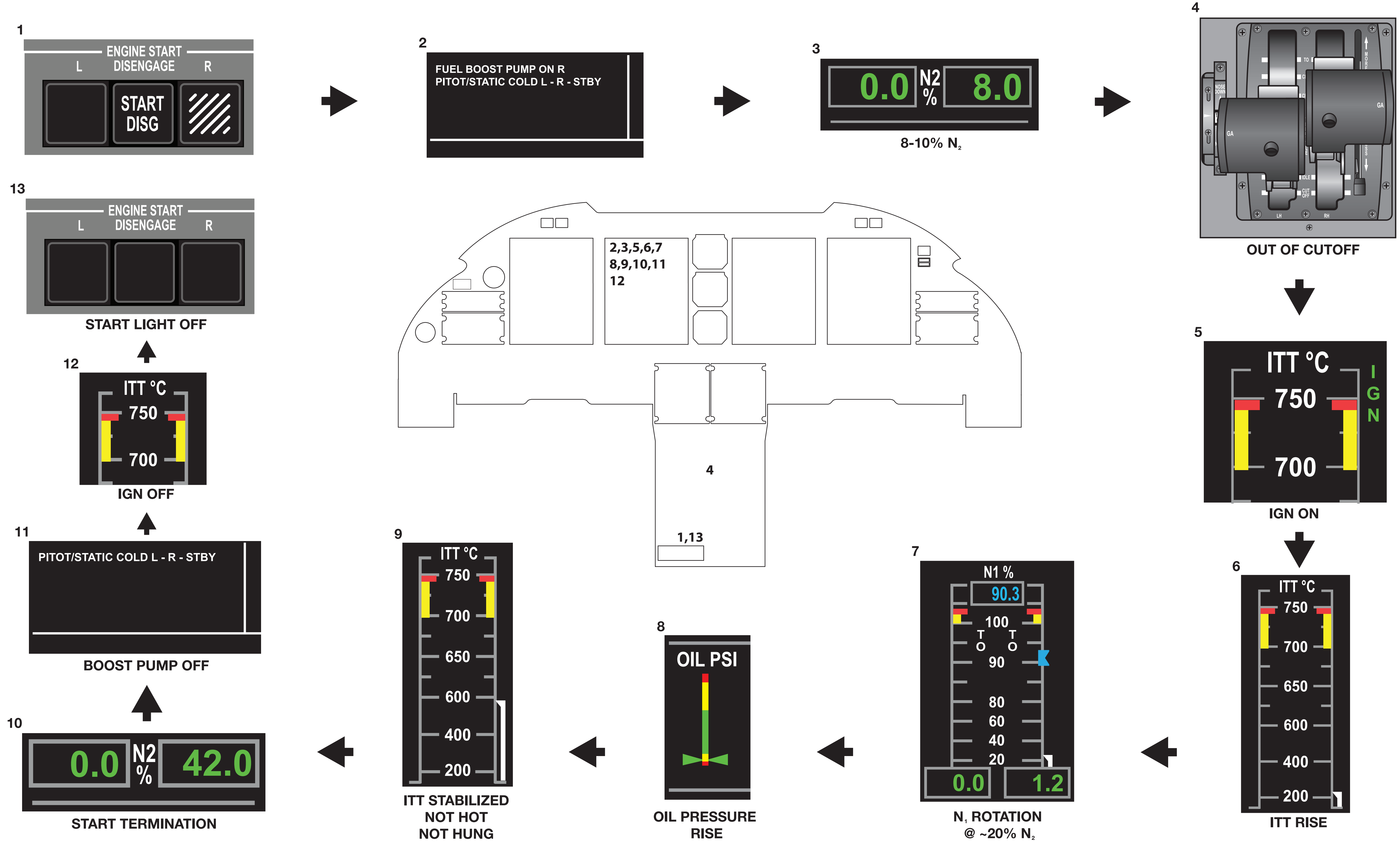
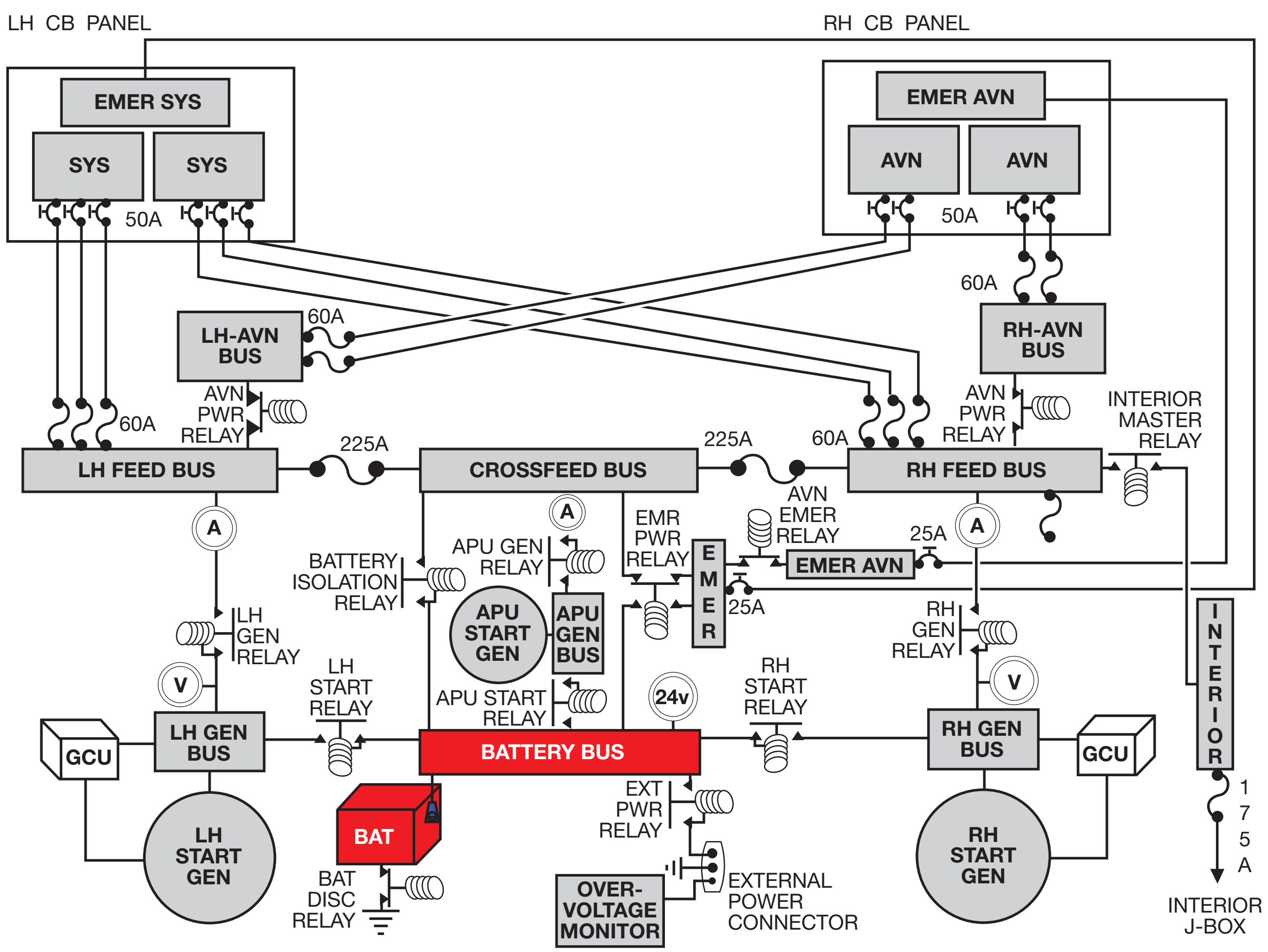
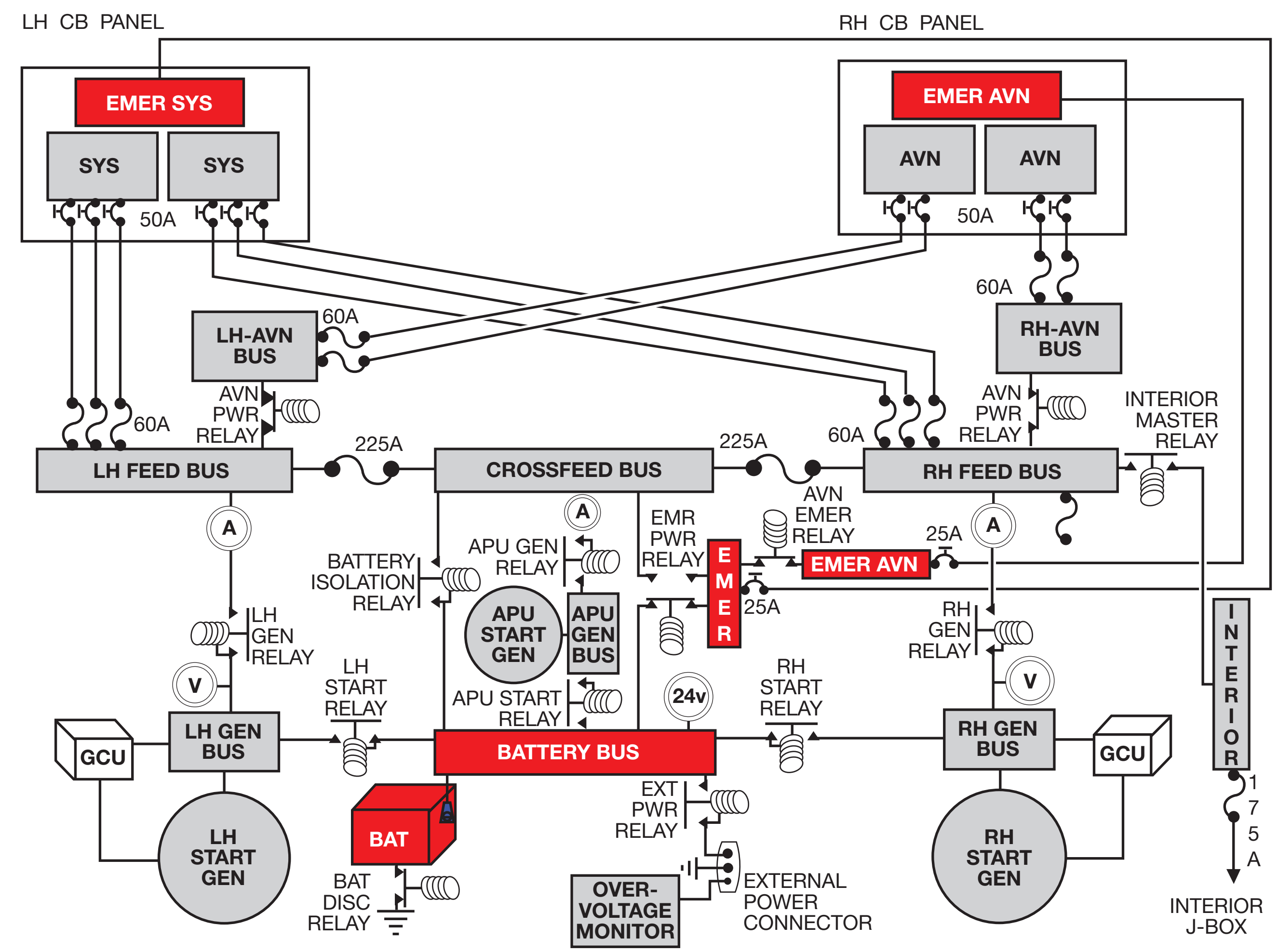


Figure 24-4. XLS+ Start Procedure “B”





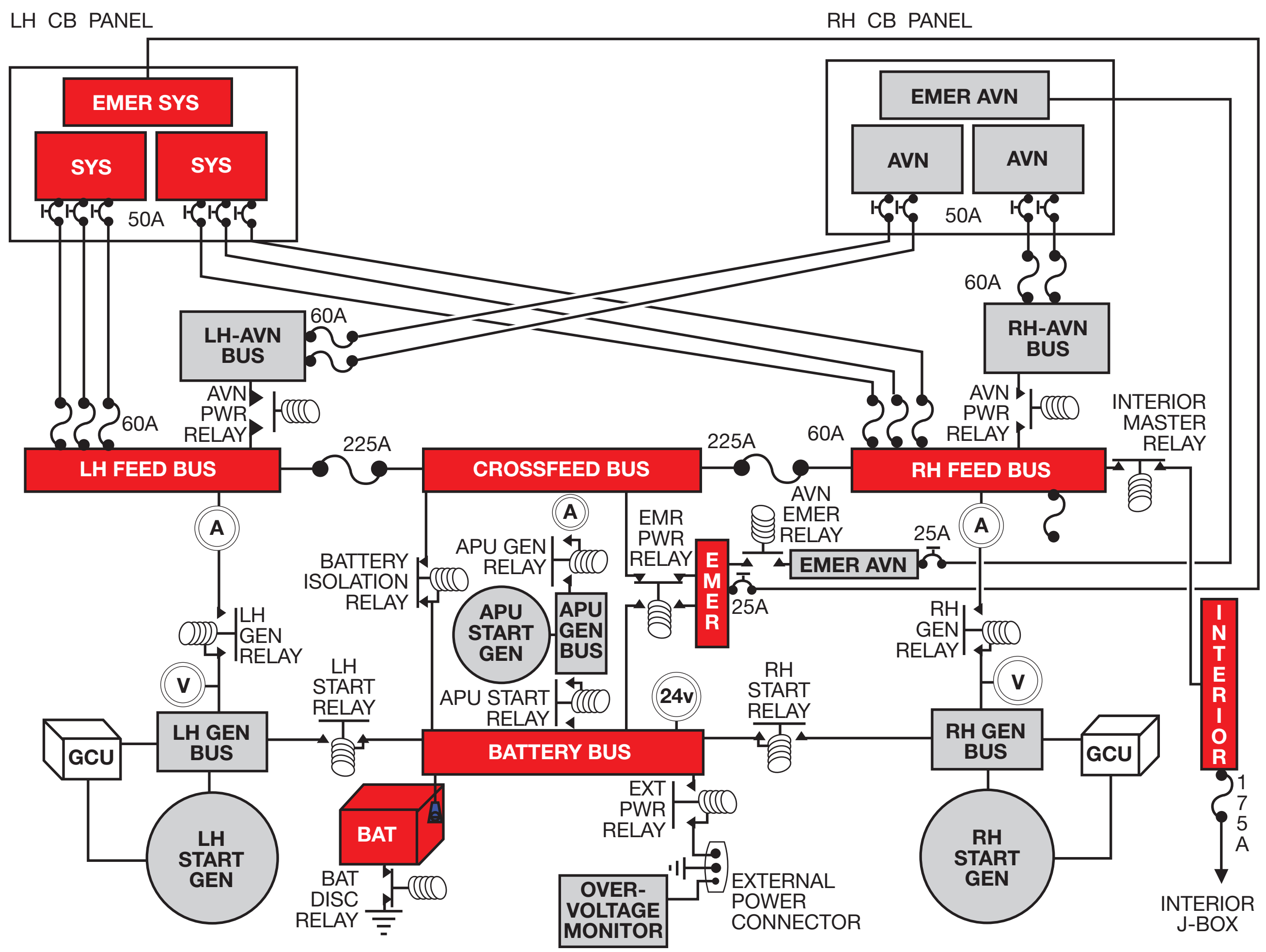
BATTERY CONNECTED

- Battery voltage is supplied to the Hot Battery Bus through the battery connector

BATTERY SWITCH IN THE "EMER" POSITION

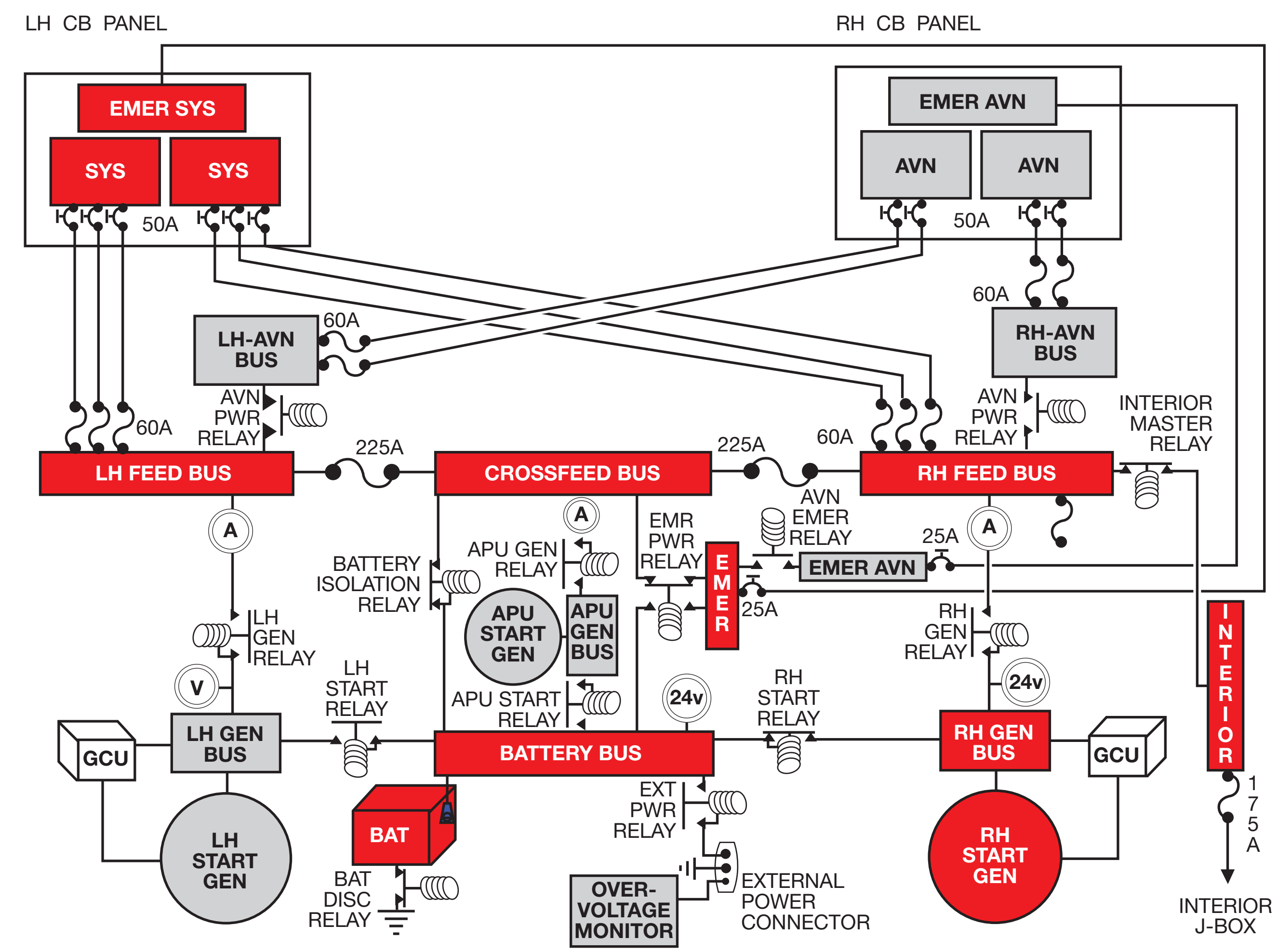
- EMER POWER RELAY is energized
- Voltage feeds through the energized set of contacts of the EMER PWR RELAY to the emer system buses
- Voltage then feeds through the relaxed contacts of the AVN EMER RELAY to the emer avionics buses

Figure 24-5. Battery Power



BATTERY SWITCH IN THE "ON" POSITION

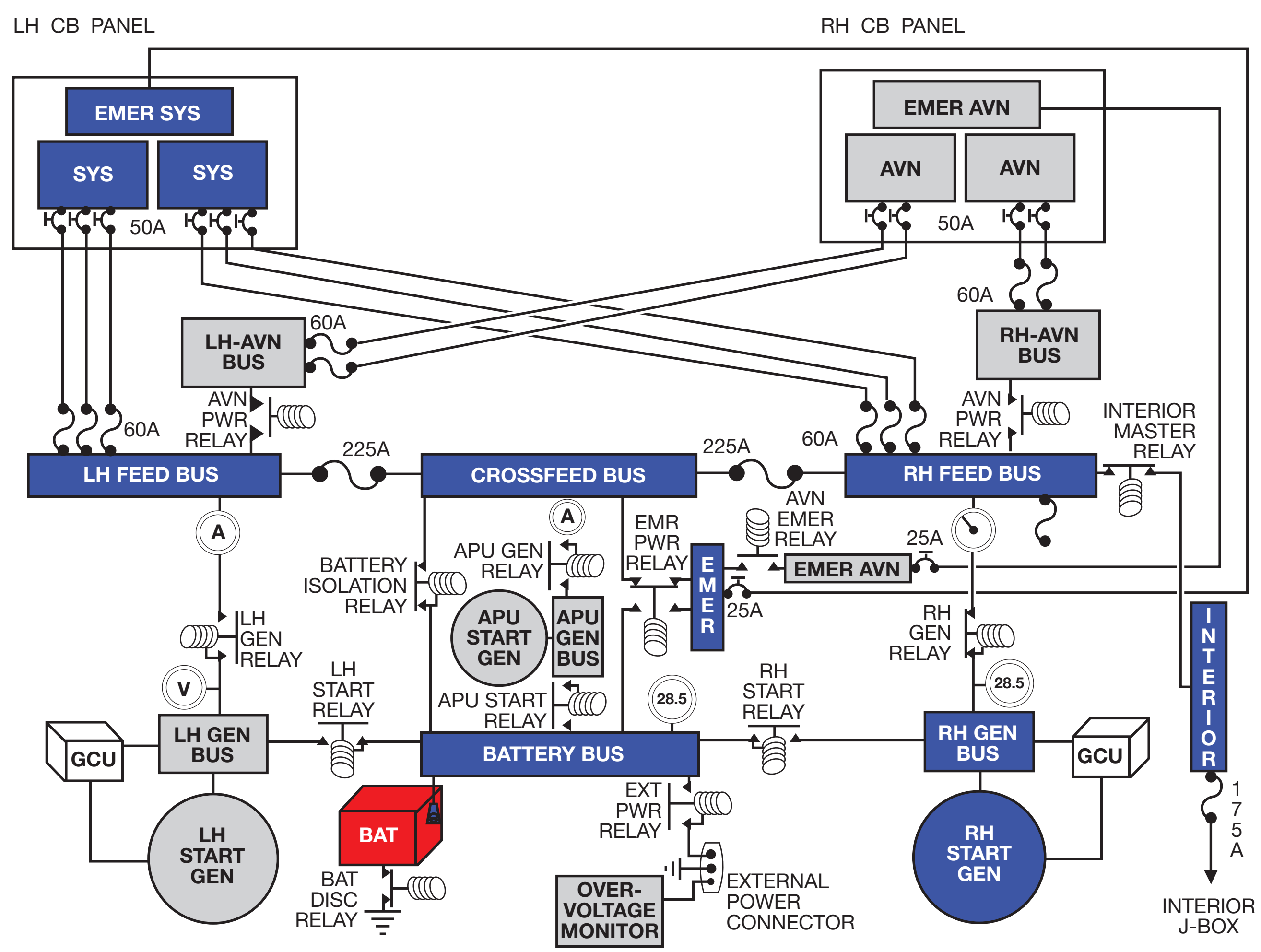
- BATTERY ISOLATION RELAY is energized closed
- Voltage feeds through the relaxed set of contacts of the EMER PWR RELAY to the emer system buses
- Interior master relay is also energized closed due to the interior master switch being in the guarded position



RH ENGINE START

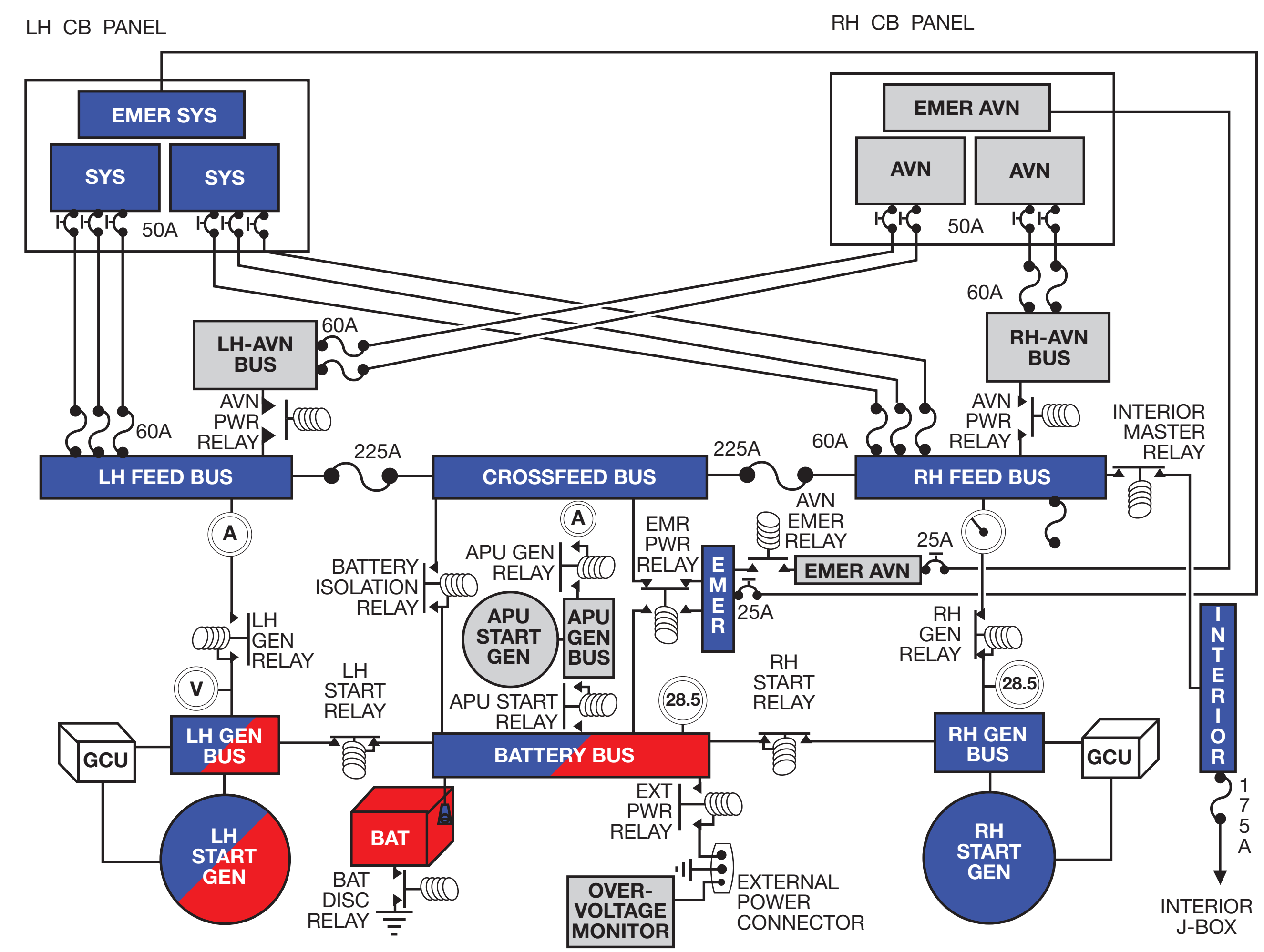
- RH start button was pressed and released
- RH START RELAY is energized closed causing the start button light to illuminate and battery power to supply the RH starter/generator

Figure 24-6. Battery Start



RH GENERATOR ON LINE

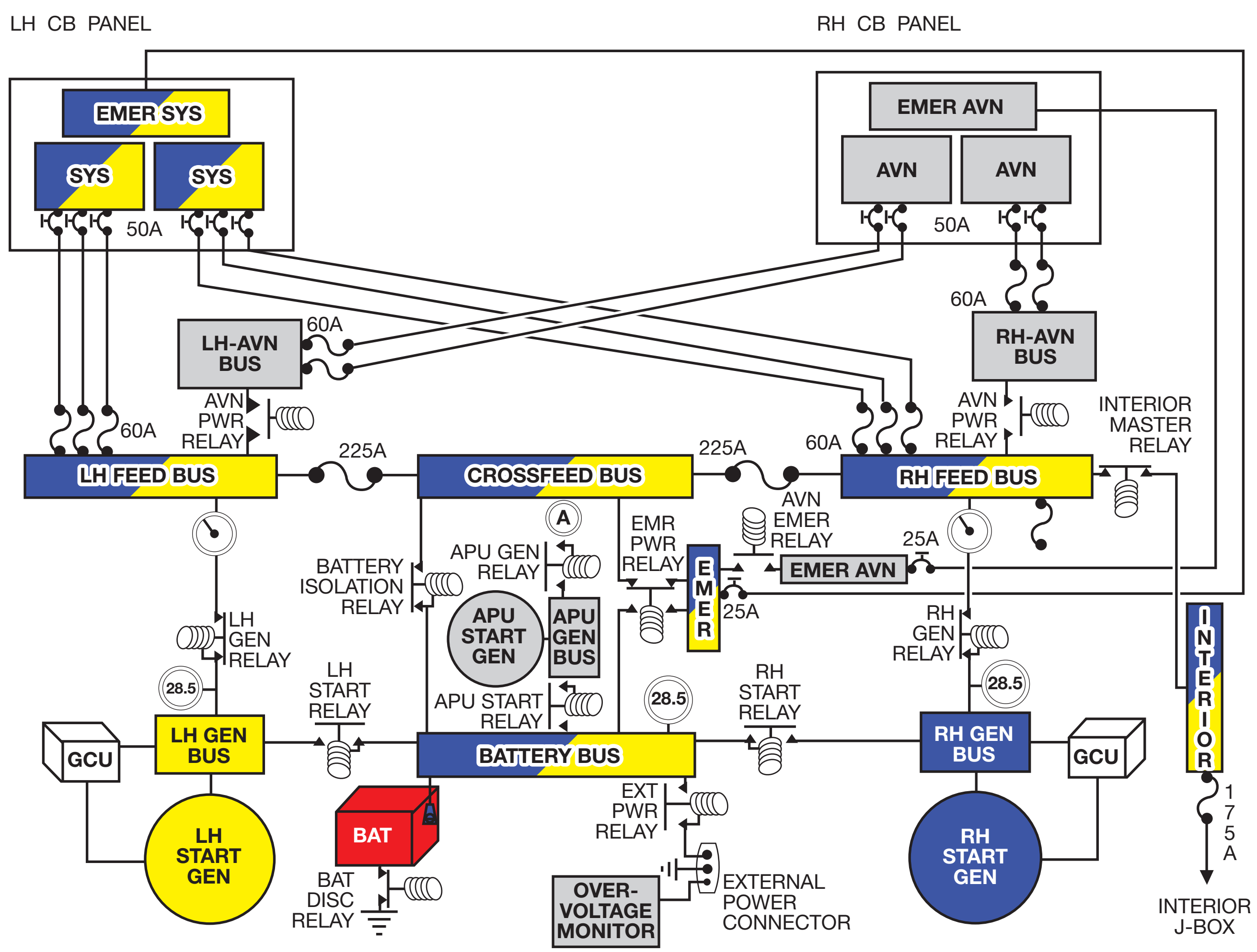
- The RH start relay is no longer energized closed since the engine accelerated past 42-46% N_2
- RH GEN switch is in the “ON” position which is the correct position for battery starts
- The RH GEN RELAY is energized closed supplying all of the buses with 28.5 volts from the RH Gen.



LH ENGINE START (GENERATOR ASSIST)

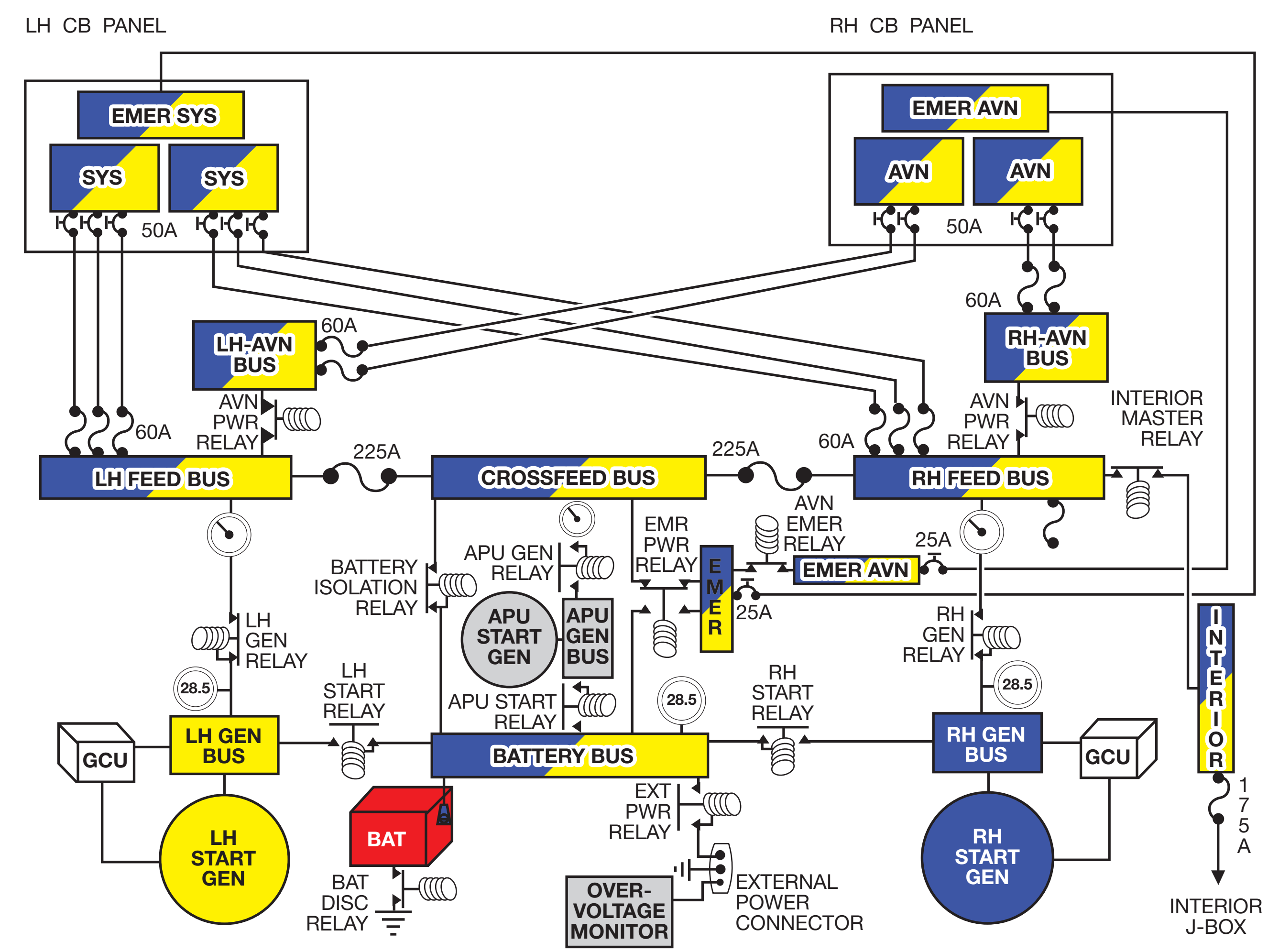
- LH start button was pressed and released
- BATTERY ISOLATION RELAY is de-energized open
- Both LH and RH START RELAYS are energized closed illuminating both start buttons and providing a path for RH gen and Battery power to supply the LH start/generator

Figure 24-7. Generator Assisted Start



LH GENERATOR ON LINE

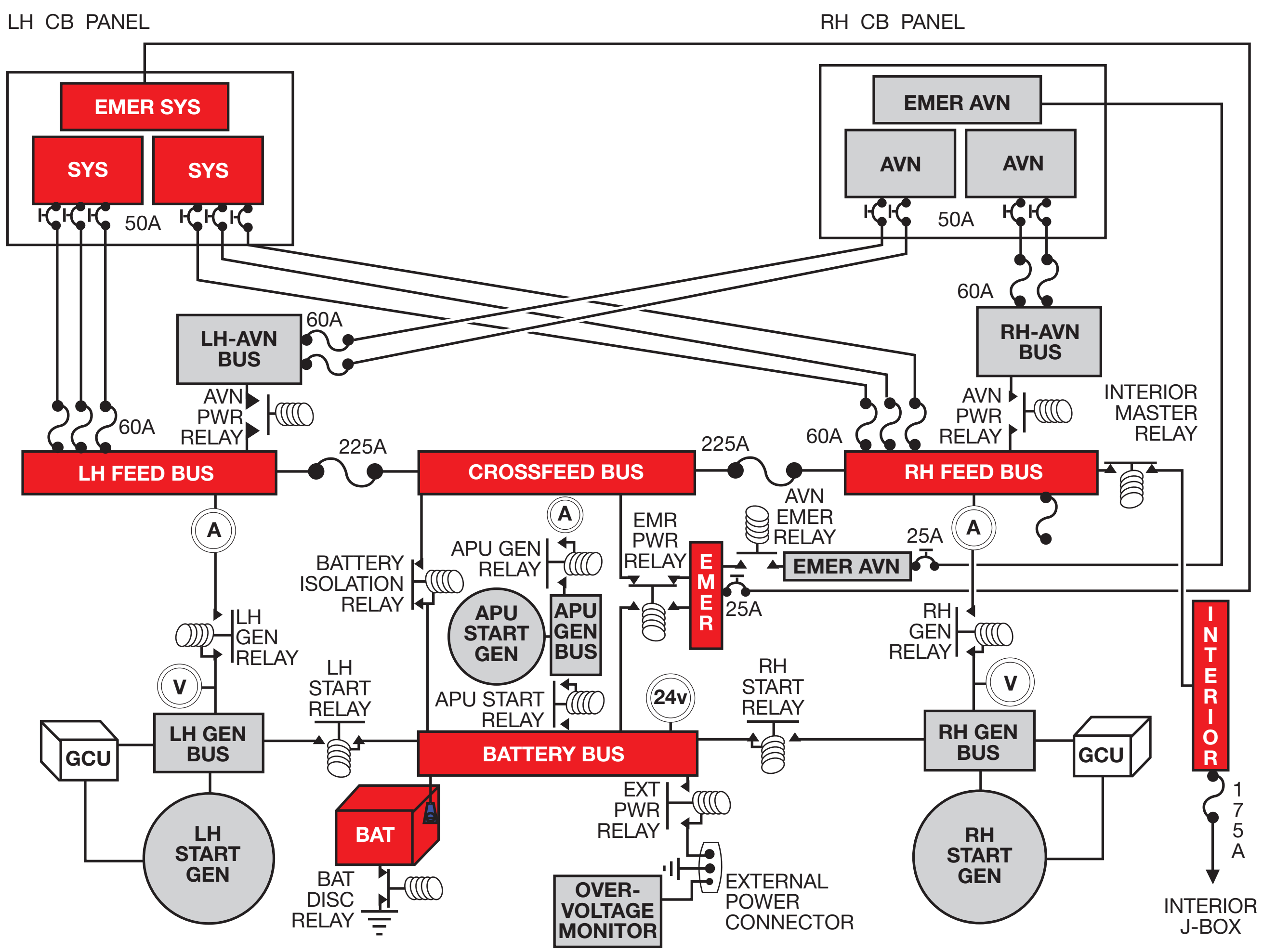
- The start relays are no longer energized closed since the LH engine accelerated past 42-46% N_2
- Both GEN switches are in the “ON” position which is the correct position for the battery starts
- Both GEN RELAYS are energized closed supplying all of the buses with 28.5 volts from both Gens.



AVIONICS POWER ON

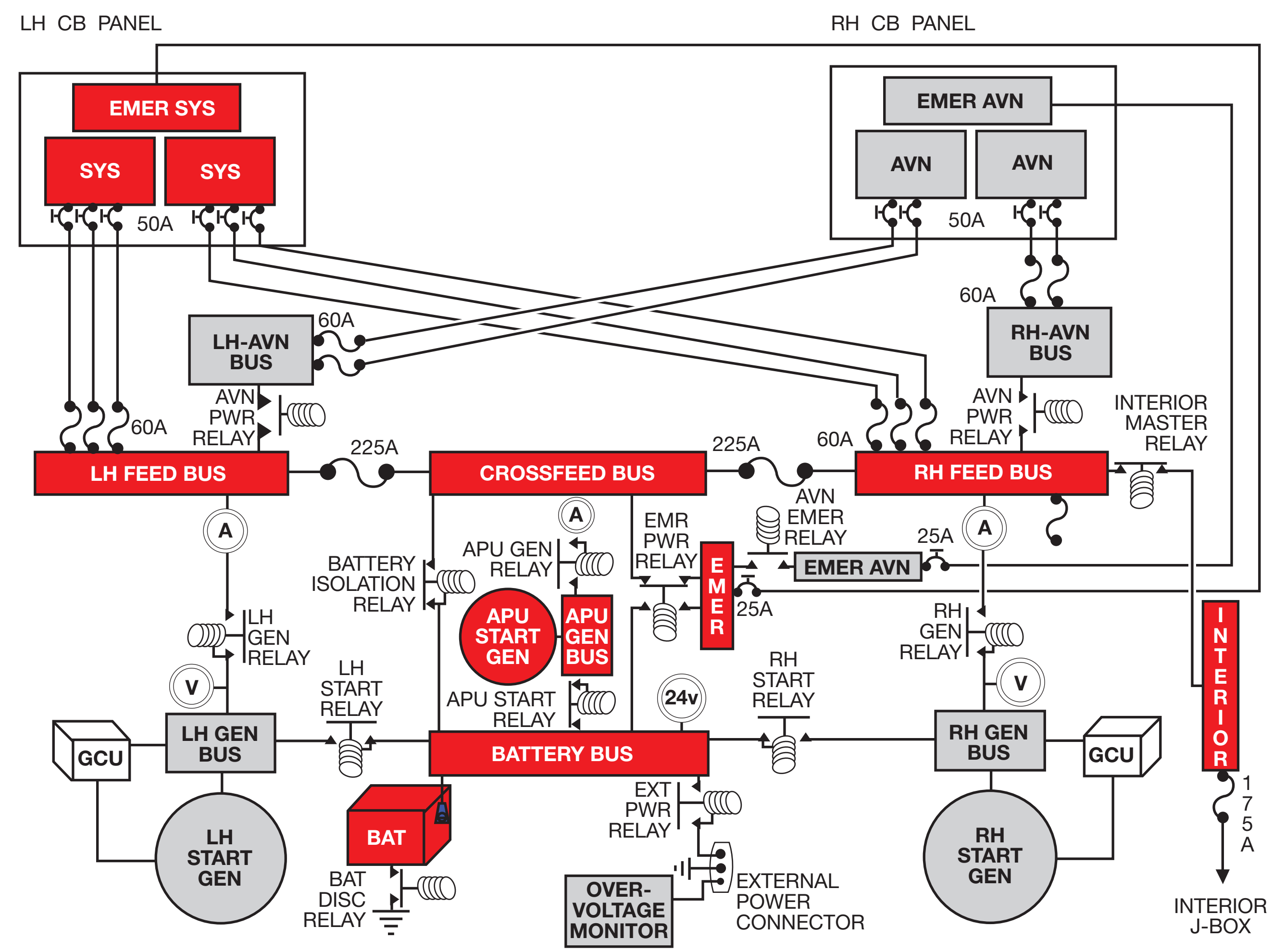
- AVIONICS POWER switch has been placed in the “ON” position
- The AVN EMER RELAY and both AVN PWR RELAYS are de-energized closed to supply voltage to all of the avionics buses

Figure 24-8. Generators Online



BATTERY SWITCH ON

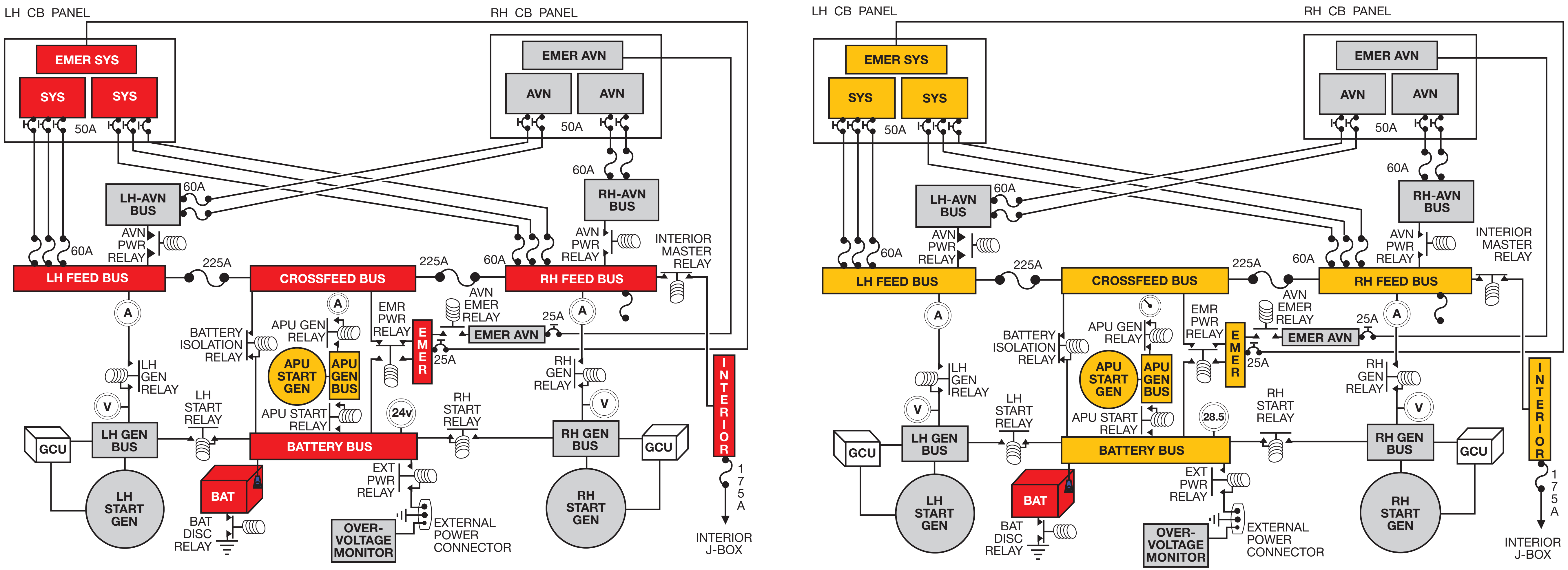
- Battery voltage is supplied to the Hot Battery Bus through the battery connector



APU START

- APU START/STOP switch momentarily placed in start
- The APU START RELAY is energized closed illuminating the “APU RELAY ENGAGED” annunciator and supplying power to the APU starter/generator

Figure 24-9. APU Start



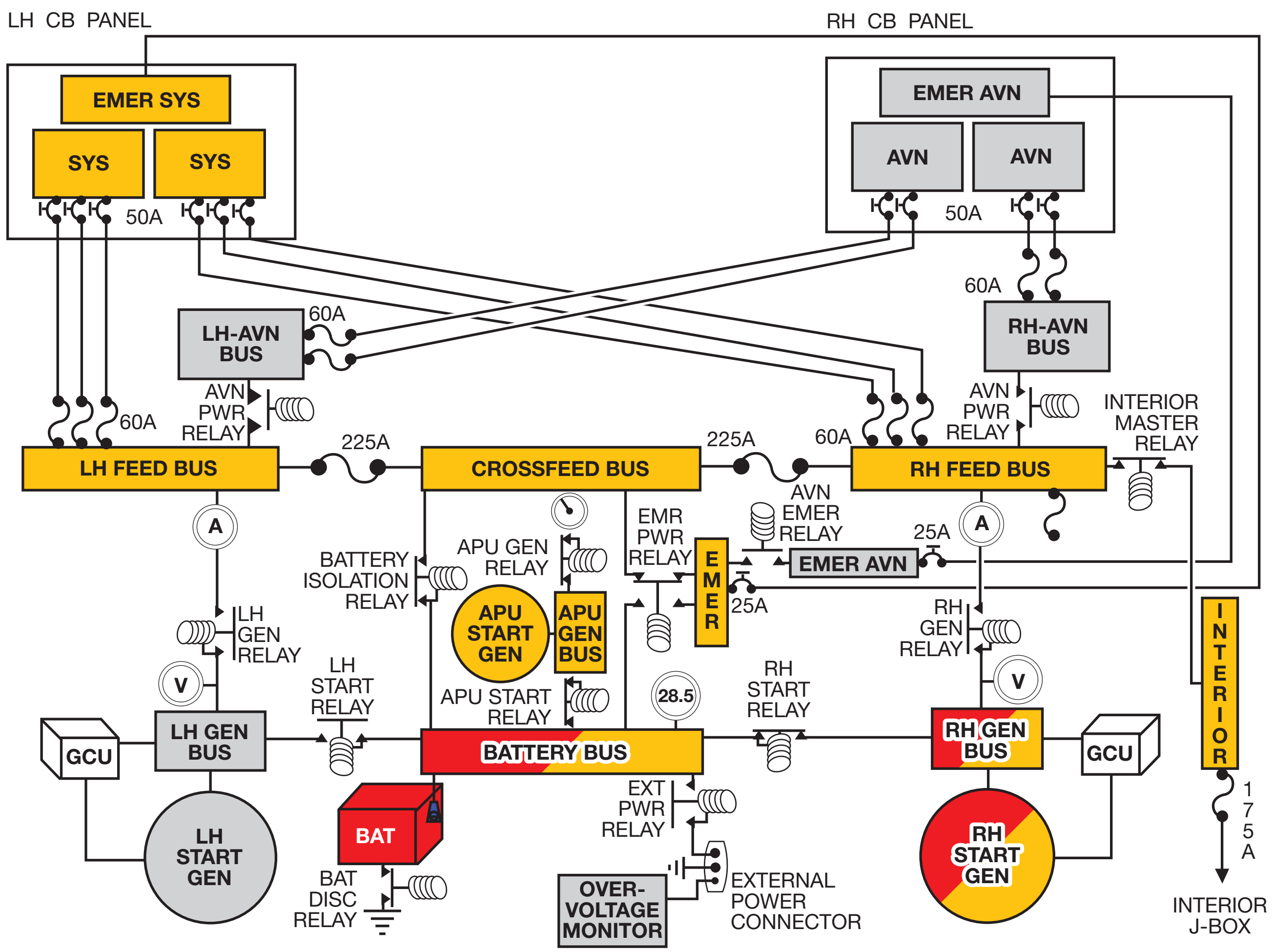
APU RUNNING

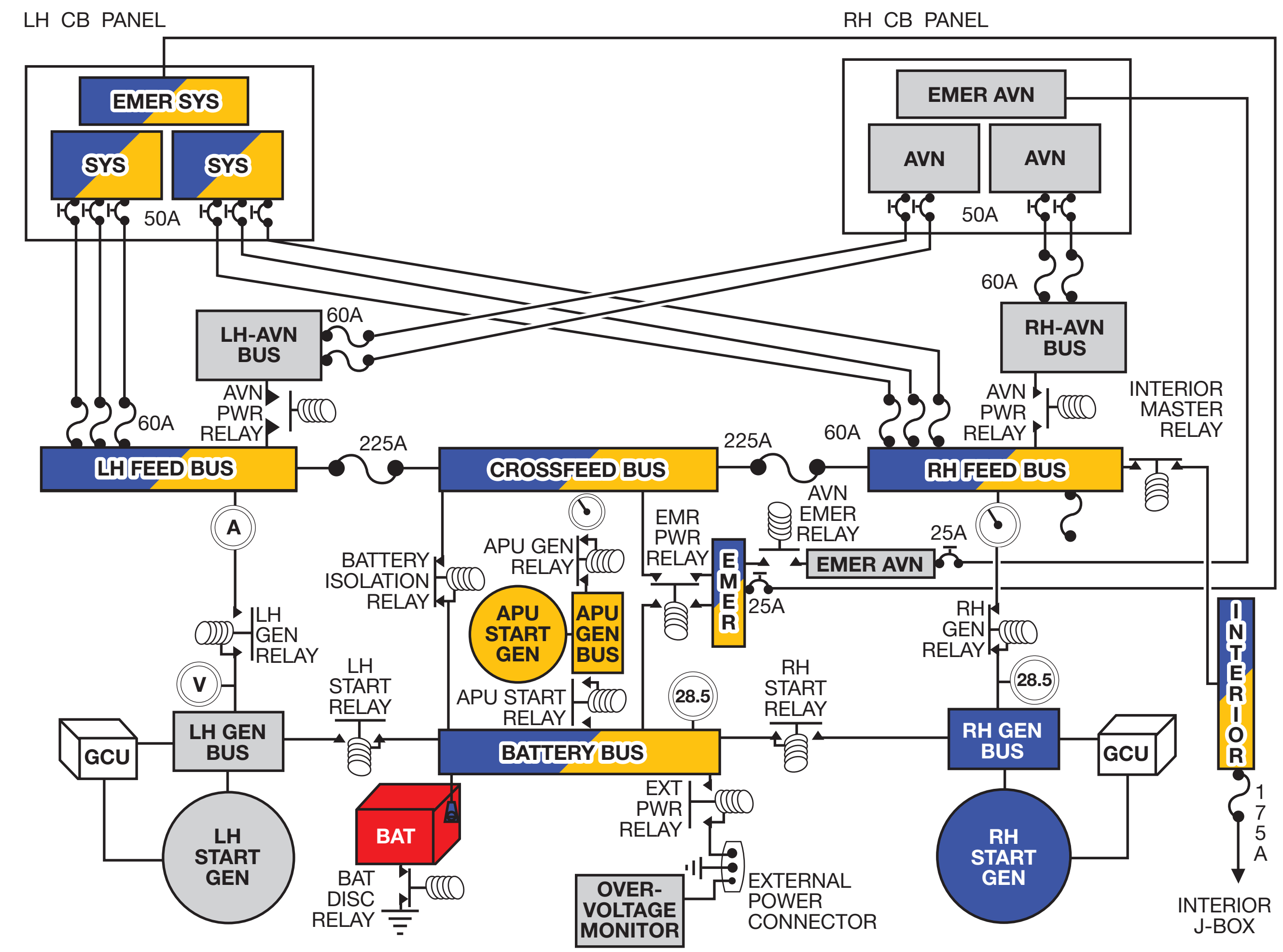
- At approx. 50% speed the start sequence terminates and the APU START RELAY re-opens extinguishing the “APU RELAY ENGAGED” annunciator
- The apu continues to accelerate to 100% speed (idle) at which point the “READY TO LOAD” annunciator will illuminate

APU GENERATOR ON LINE

- APU GENERATOR switch was placed in the “ON” position
- The APU GENERATOR RELAY closed providing APU gen power to the corssfeed bus and on to all of the system buses

Figure 24-10. APU Generator Online





RH ENGINE START

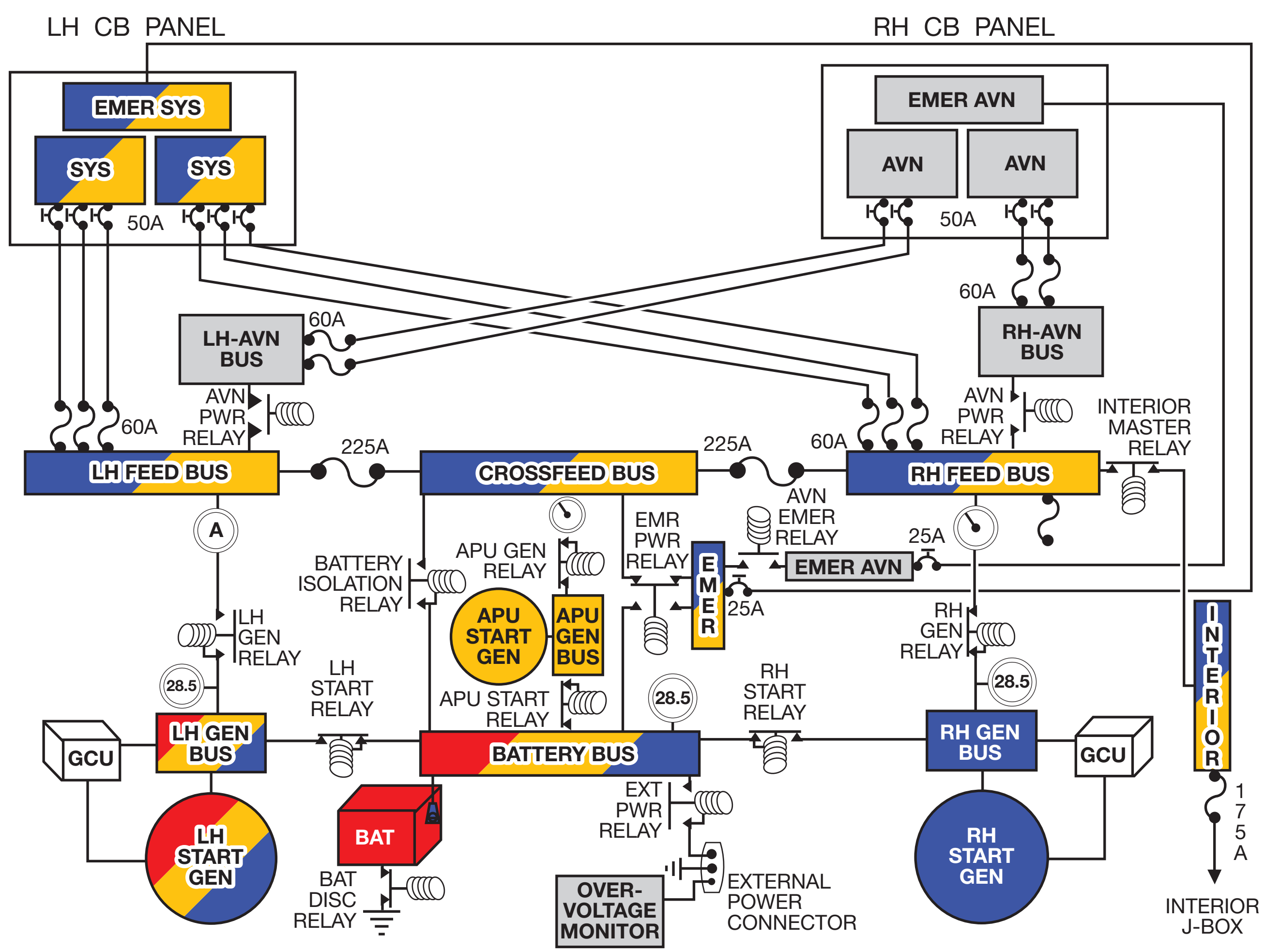
NOTE: Pre-Service Publication 24-14

- The RH START button was pressed and released
- The RH START RELAY closed illuminating the start button light
- The APU START RELAY closed illuminating the apu relay engaged annunciator

APU AND RH GENERATORS ON LINE

- The start relays are no longer energized closed since the RH engine accelerated past 42-46% N_2
- Both GEN switches are in the "ON" position which is the correct position for engine starts using the APU
- The RH GEN RELAY is energized closed

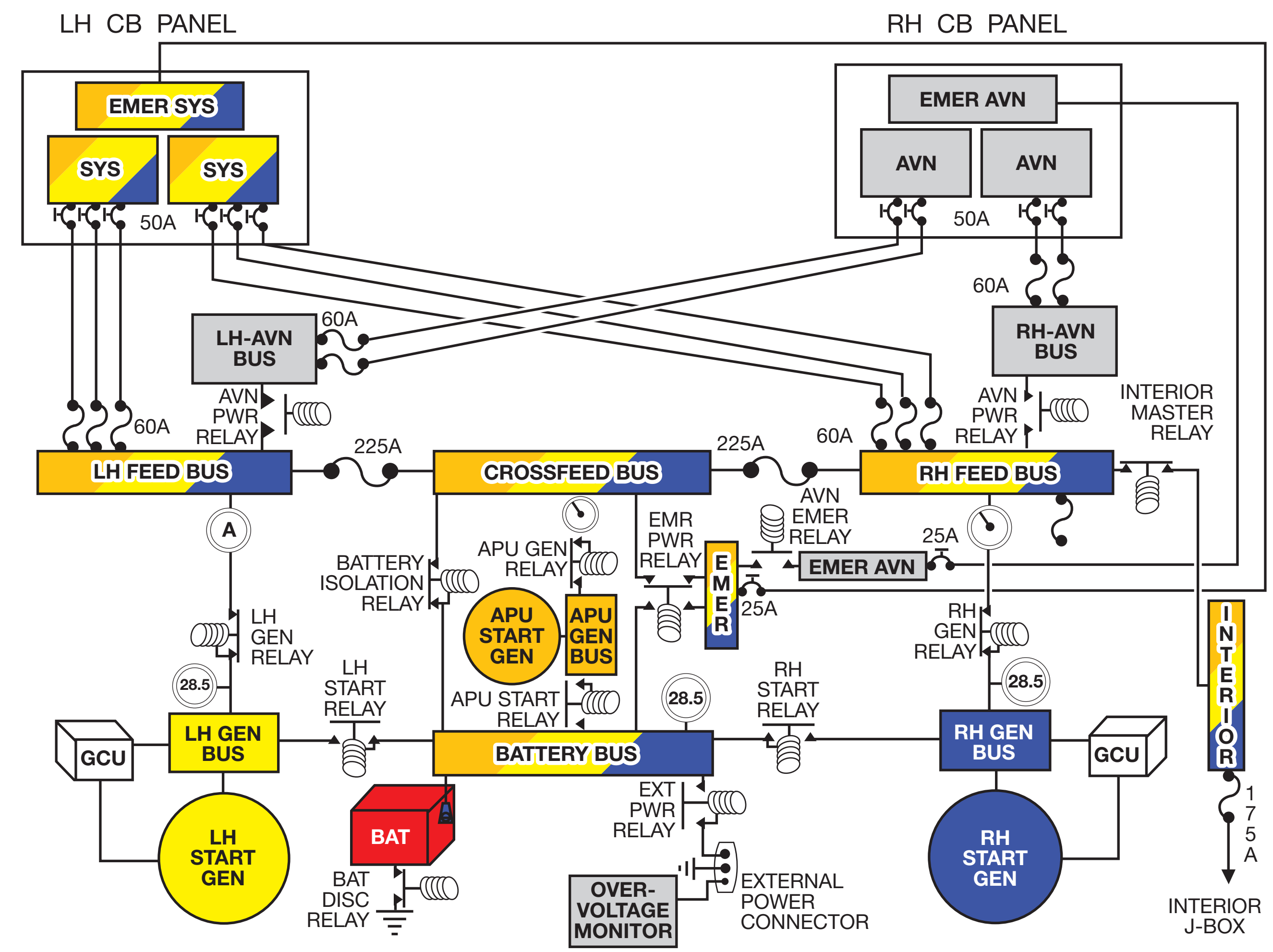
Figure 24-11. APU Assisted Start (Sheet 1 of 2)



LH ENGINE START (GENERATOR ASSIST)

NOTE: Pre-Service Publication 24-14

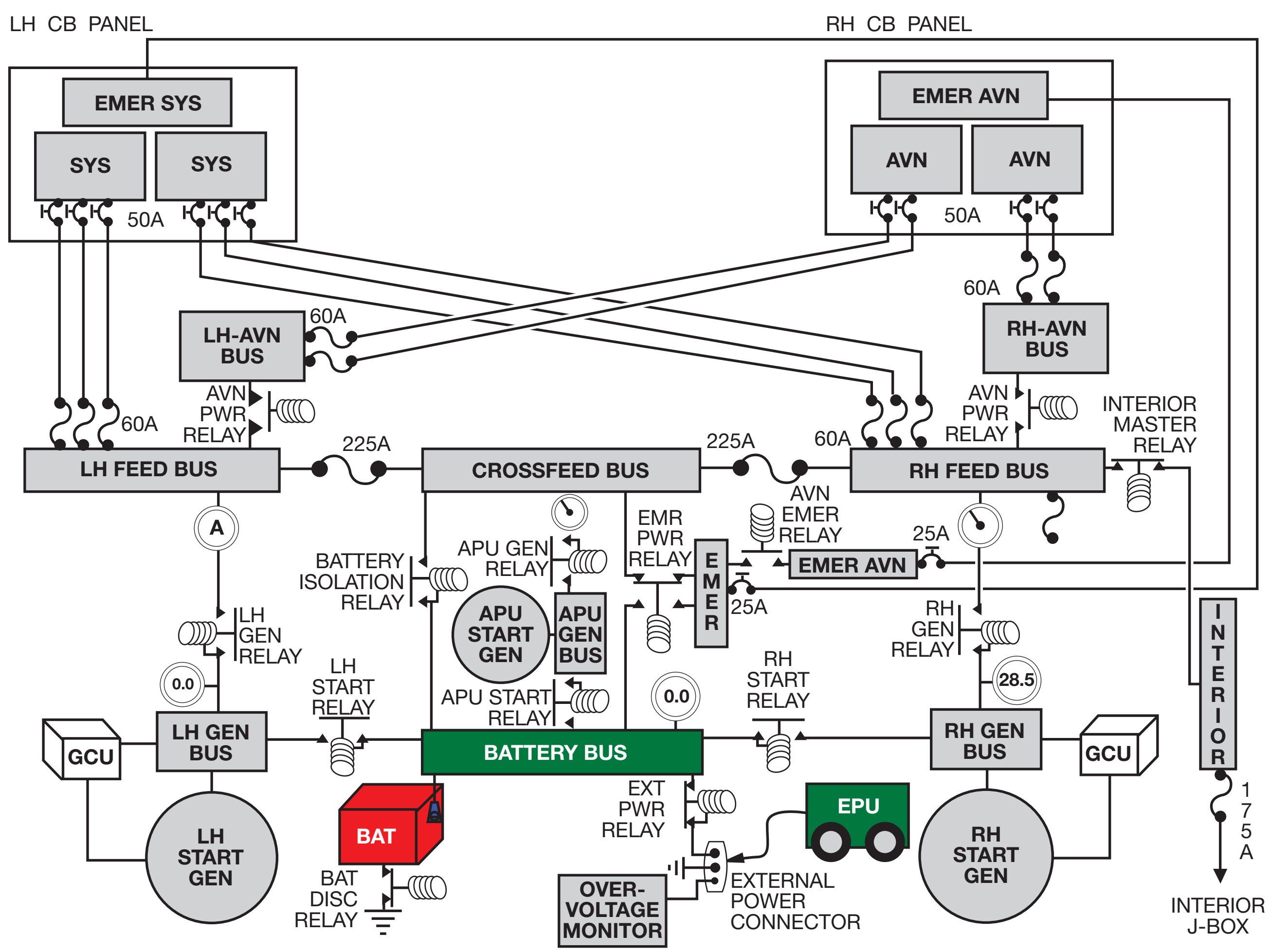
- LH START button was pressed and released
- The APU and both LH and RH start relays are energized closed illuminating both start buttons and the APU relay engaged annunciator
- The Battery Isolation relay is de-energized open protecting the 225A current limiters



LH GENERATOR ON LINE

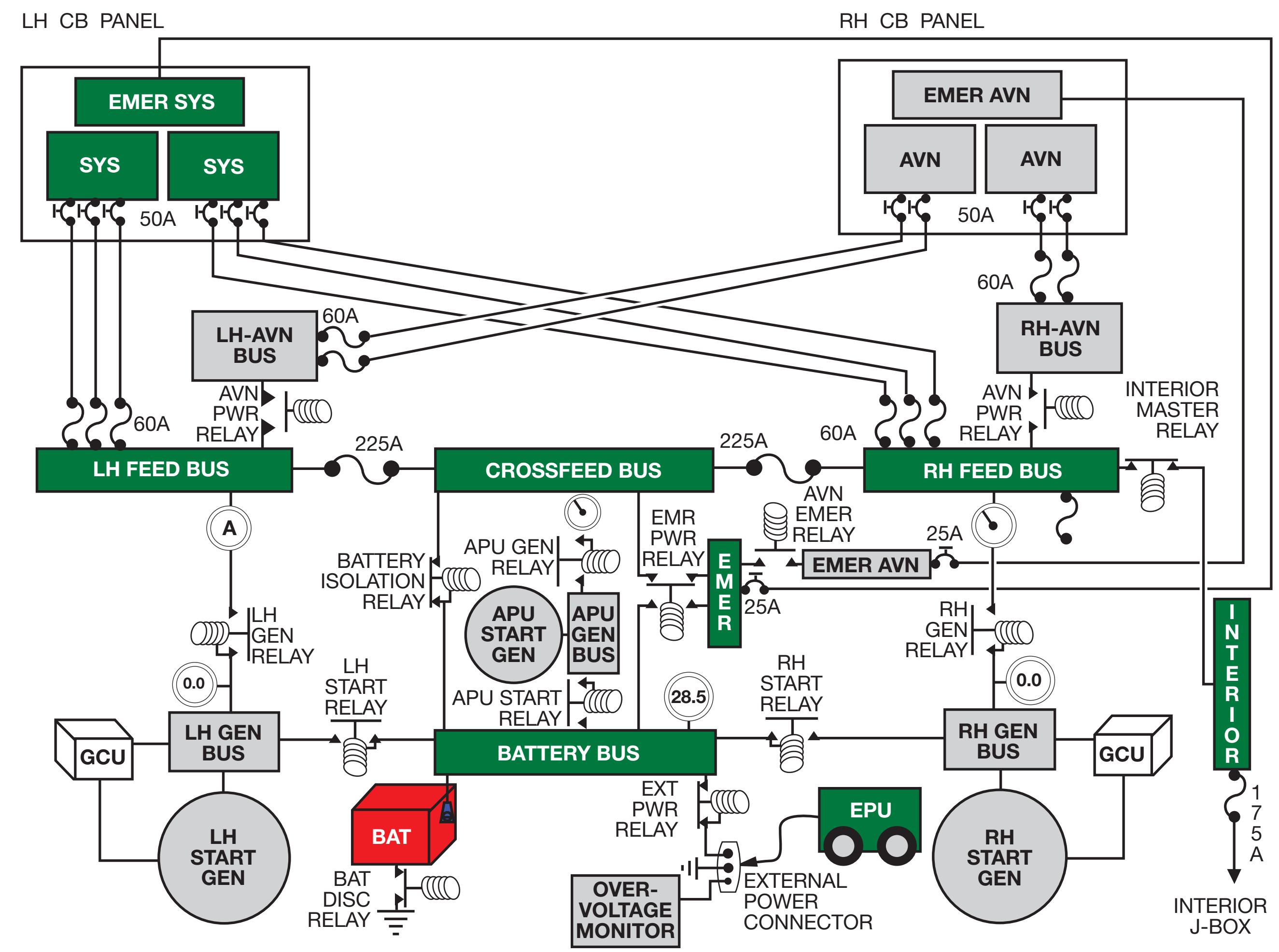
- The start relays are no longer energized closed nor is the battery isolation relay de-energized open since the LH engine accelerated past 42-46% N_2
- The LH gen relay closed supplying the system buses with the power from both main engine generators and the APU generator

Figure 24-11. APU Assisted Start (Sheet 2 of 2)



EPU CONNECTED

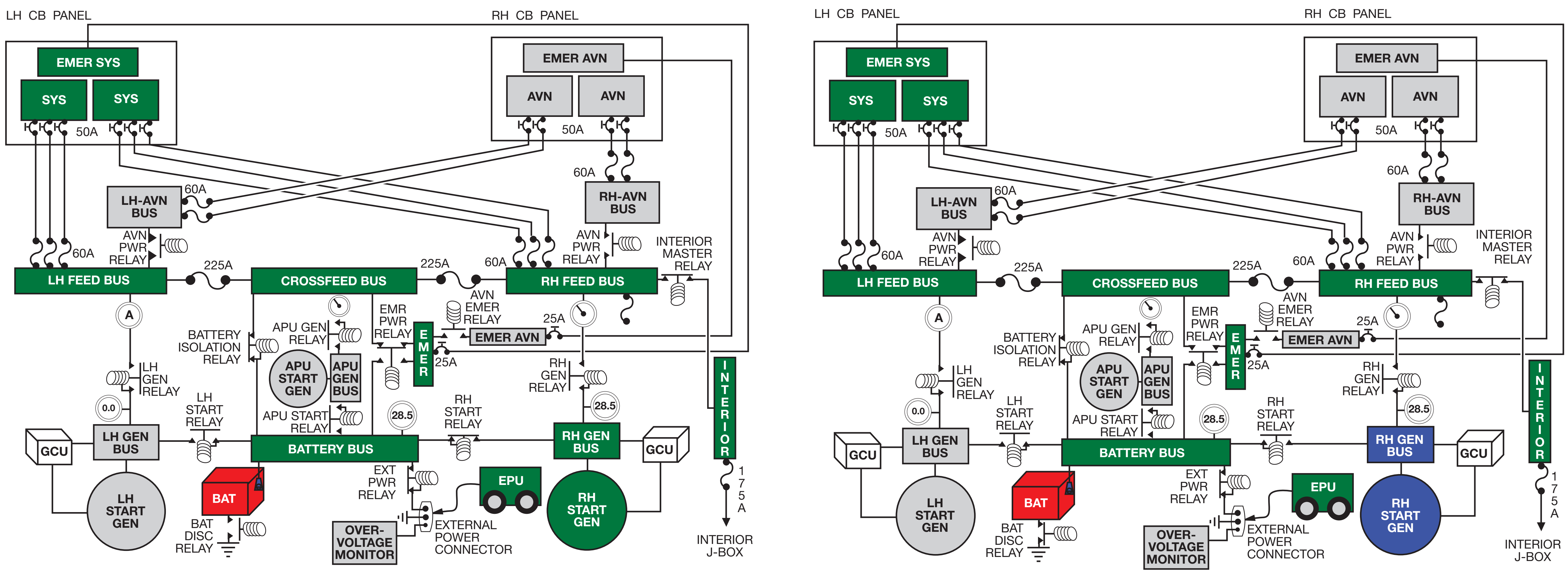
- The external power unit is producing 28 vdc and is connected to the aircraft
- Since the EPU voltage is less than 32.5 vdc and with no aircraft generators on line, the EXTERNAL POWER RELAY is energized closed providing EPU power to the Hot Battery Bus and also to the aircraft battery



BATTERY SWITCH ON

- Placing the battery switch in the "ON" position, the BATTERY ISOLATION RELAY is energized closed
- All of the system buses are now receiving EPU power

Figure 24-12. External Power Unit Starts (Sheet 1 of 4)



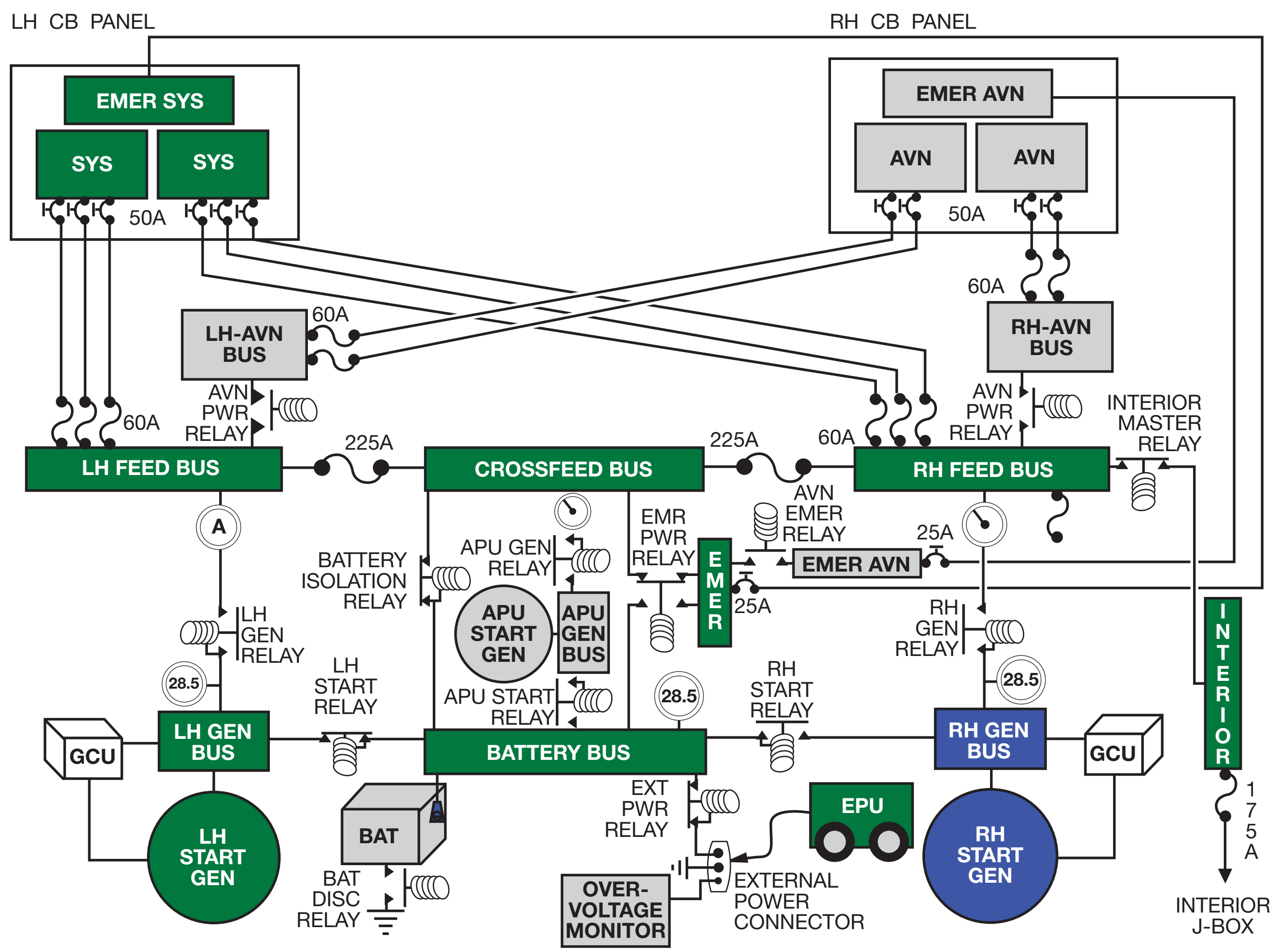
RH ENGINE START

- RH start button was pressed and released
- RH START RELAY is energized closed causing the start button light to illuminate and EPU power to supply the RH starter/generator
- The BATT DISC RELAY is energized open to prevent the battery from assisting with the start

RH ENGINE RUNNING

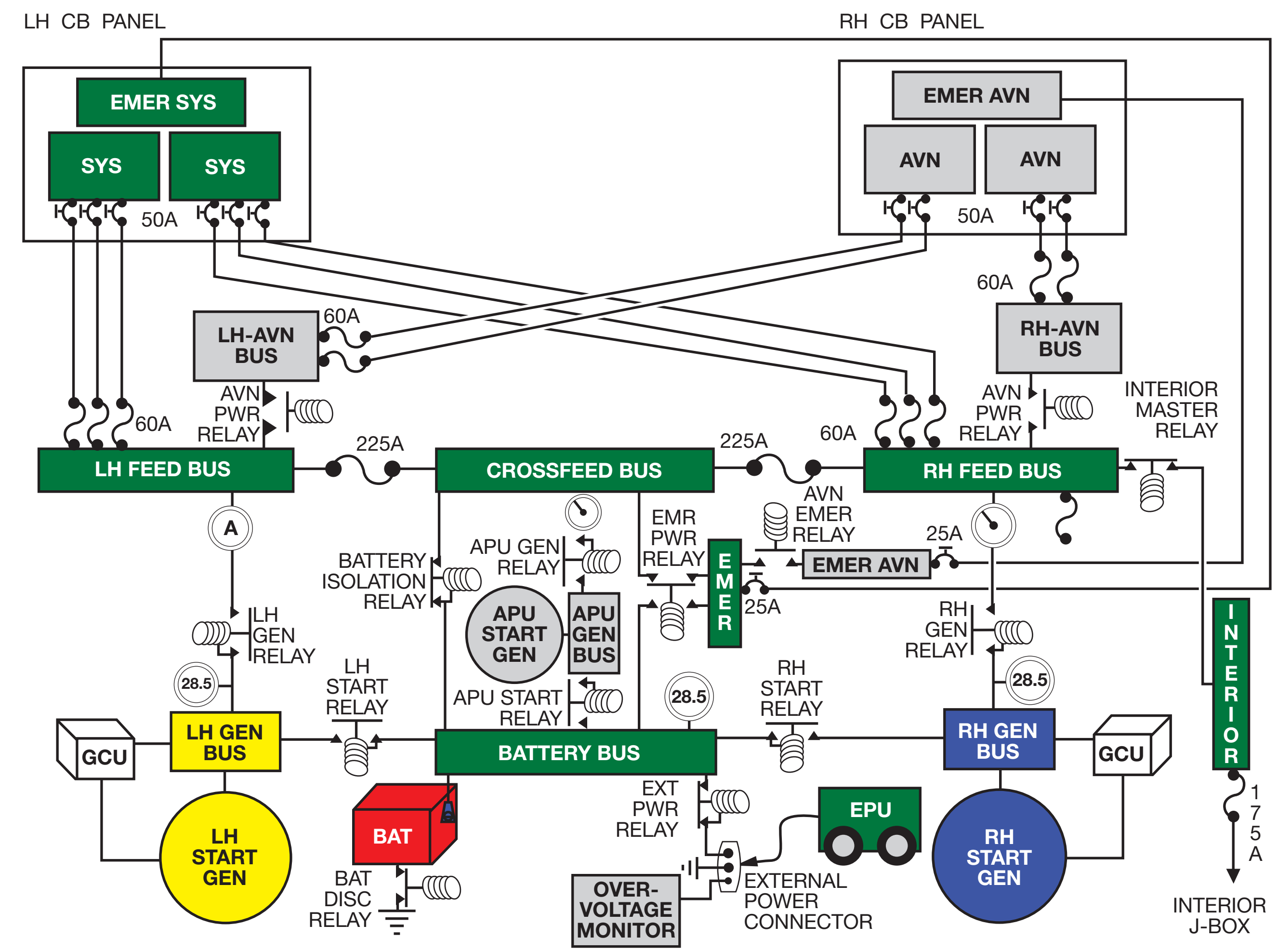
- The RH START RELAY is no longer energized closed nor is the BATT DISC RELAY energized open since the engine accelerated past 42-46% N_2
- The RH generator will produce 28.5 vdc, but its power will not be supplied to the buses since "OFF" is the correct generator switch position for EPU starts

Figure 24-12. External Power Unit Starts (Sheet 2 of 4)



LH ENGINE START

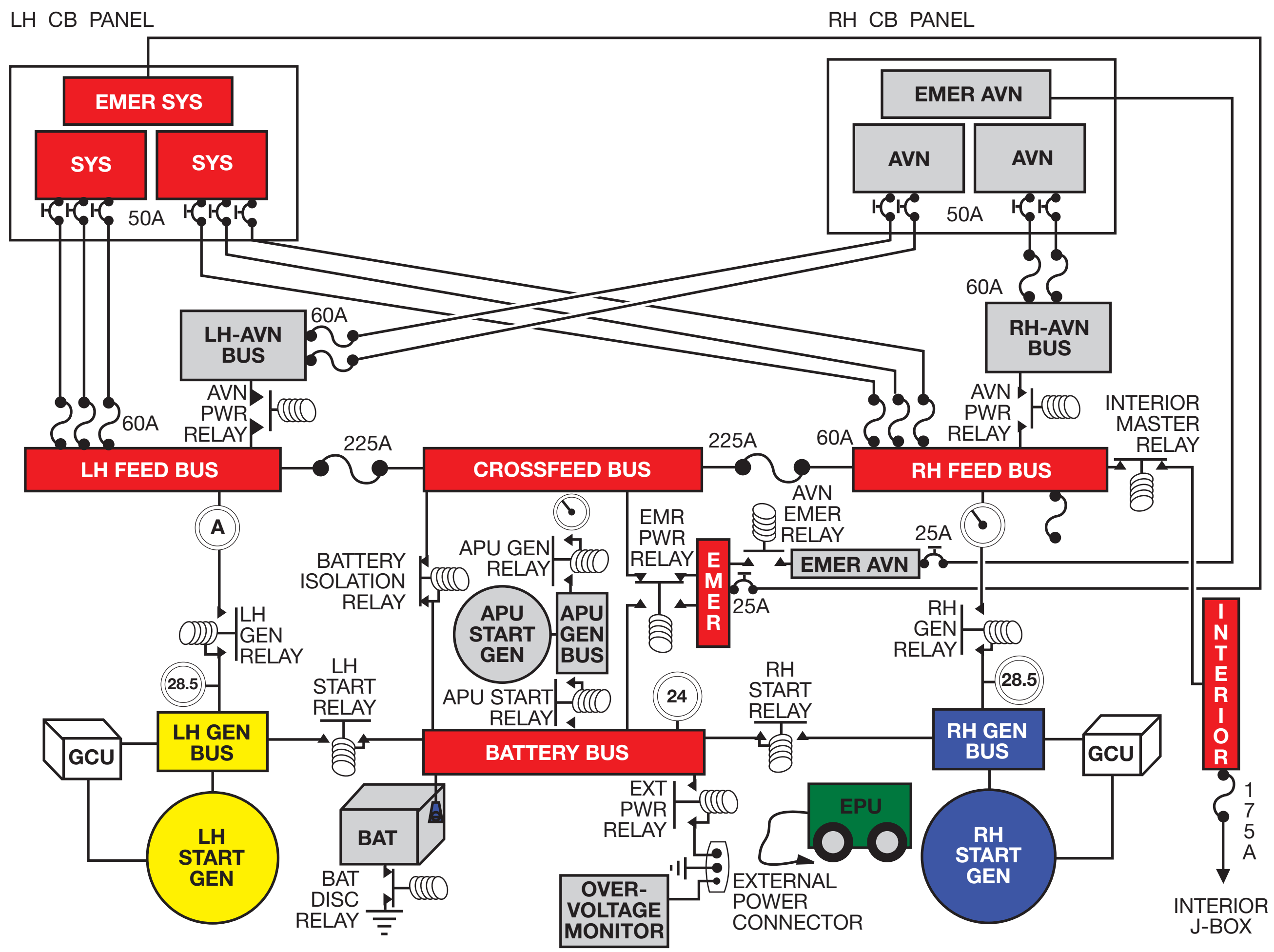
- LH start button was pressed and released
- LH START RELAY is energized closed causing the start button light to illuminate and EPU power to supply the LH starter/generator
- The BATT DISC RELAY is energized open to prevent the battery from assisting with the start



LH ENGINE RUNNING

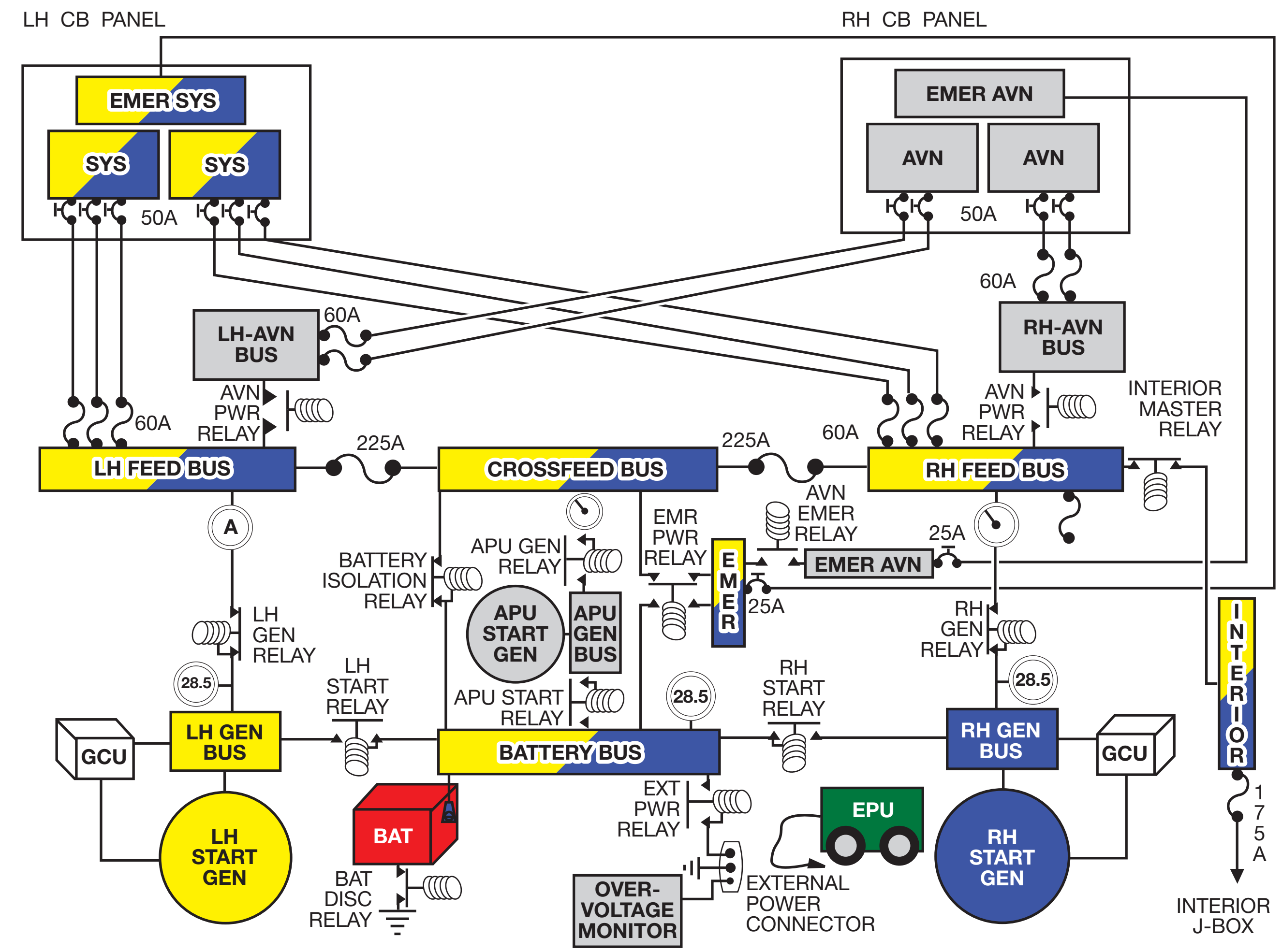
- The LH START RELAY is no longer energized closed nor is the BATT DISC RELAY energized open since the engine accelerated past 42-46% N_2
- The LH generator will produce 28.5 vdc, but its power will not be supplied to the buses since “OFF” is the correct generator switch position for EPU starts

Figure 24-12. External Power Unit Starts (Sheet 3 of 4)



EPU DISCONNECTED

- Per the checklist, the EPU should be disconnected prior to placing the generators on line
- When the EPU power plug is removed, the EXT PWR RELAY will be de-energized open
- The cockpit voltmeter will drop back down to Battery voltage of 24-25 vdc



LH AND RH GENERATORS ON LINE

- After the EPU is disconnected, the GEN switches are placed in the "ON" position
- The LH and RH GEN RELAYS are energized closed supplying all of the system buses with 28.5 vdc from the two engine generators which are then sharing the current load

Figure 24-12. External Power Unit Starts (Sheet 4 of 4)

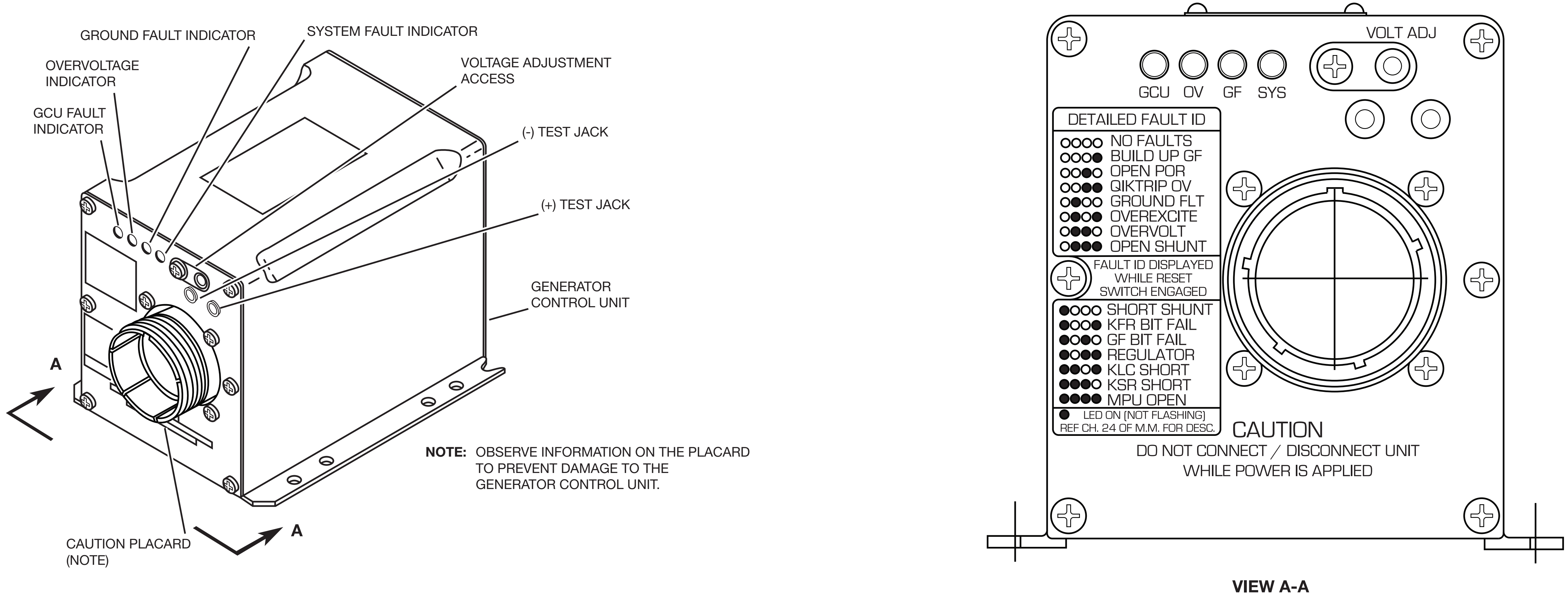


Figure 24-13. Generator Control Unit (GCU)

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BATTERY CONNECTED

CONDITIONS

1. Battery plugged in.

OBJECTIVES

1. Connect battery to hot battery bus.
2. Provide power to battery relay, left and right start relays, and left and right K2 PCB relays.

SEQUENCE OF EVENTS

1. Power applied to hot battery bus.
2. Power on pin 19 of the left and right start PCB's applies power to the K2 relays.

NOTES

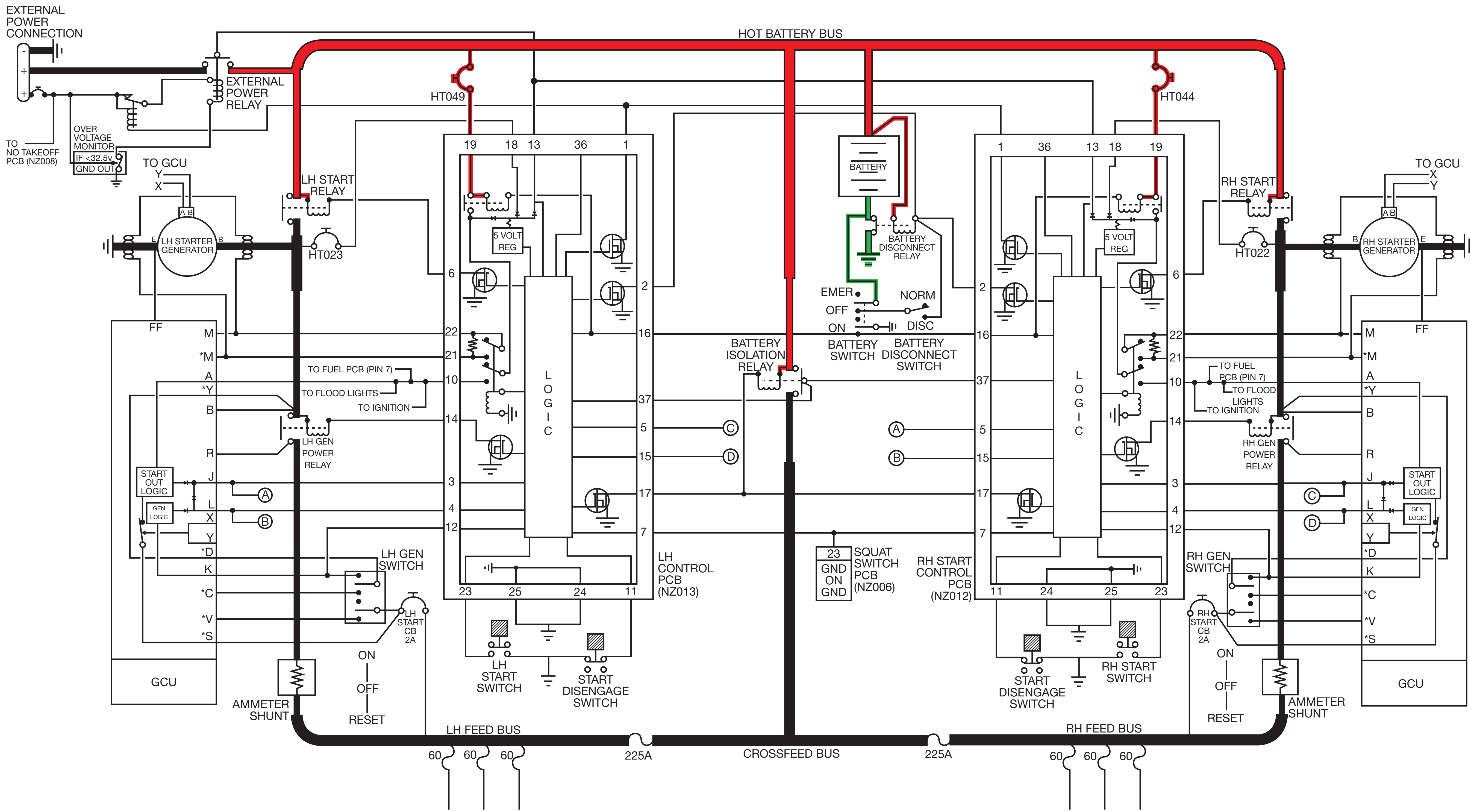

Figure 24-14. Simplified Electrical System - Battery Connected

BATTERY SWITCH ON

CONDITIONS

1. Battery switch ON

OBJECTIVES

1. Energize the battery relay closed to connect the hot battery bus to the crossfeed bus.
2. Energize the left and right K2 PCB relays closed.

SEQUENCE OF EVENTS

1. Both K2 PCB relays energized closed because of ground provided on pin 16 of both start PCB's through the battery switch.
2. Power applied to PAL and one contact of K1 on both start PCB's.
3. Discrete ground input to PAL provided on pin 16 of both start PCB's through the battery switch.
4. PAL causes ground on Pin 17 of right and left start PCB's energizing the battery relay closed.

NOTES

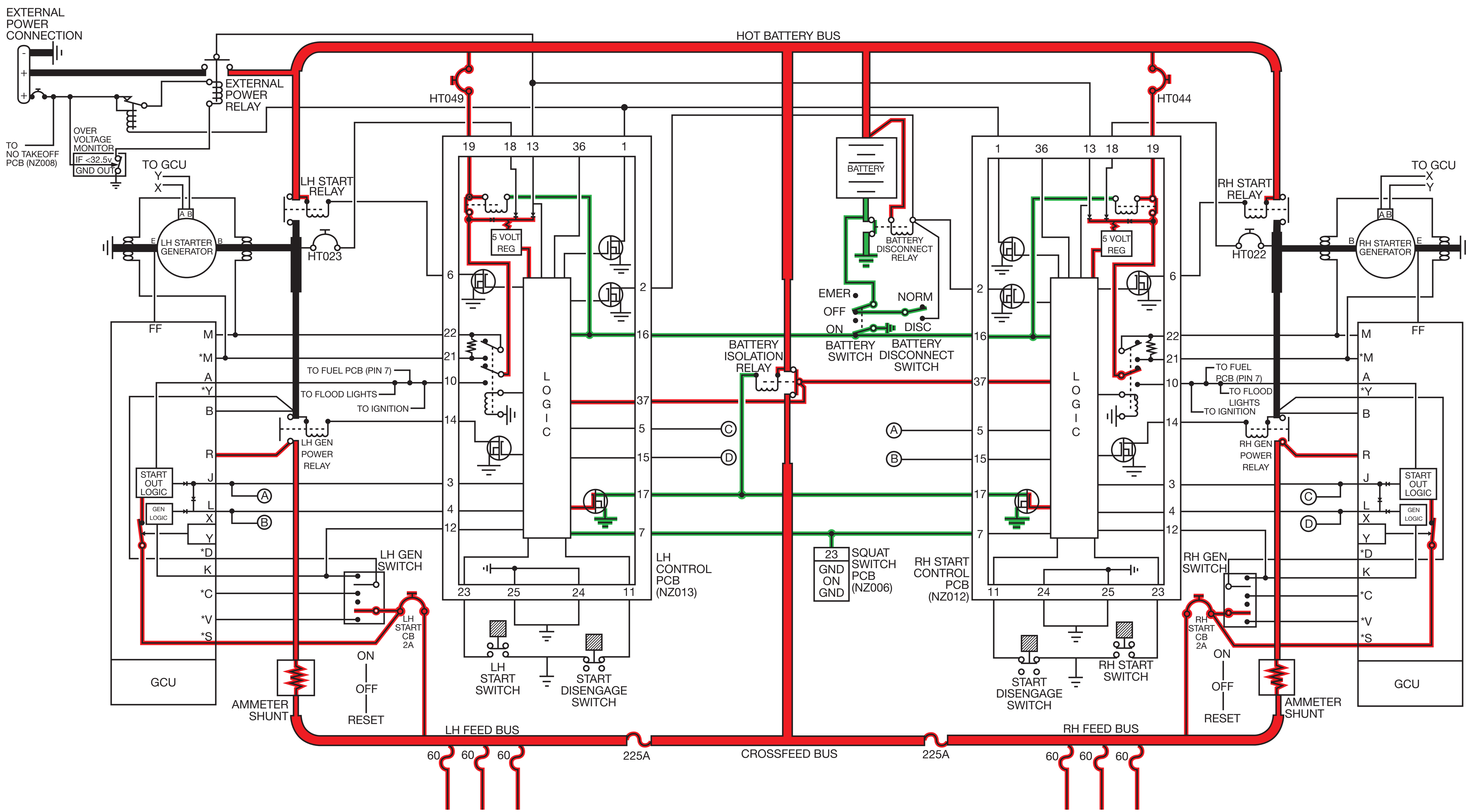

Figure 24-15. Simplified Electrical System - Battery Switch On

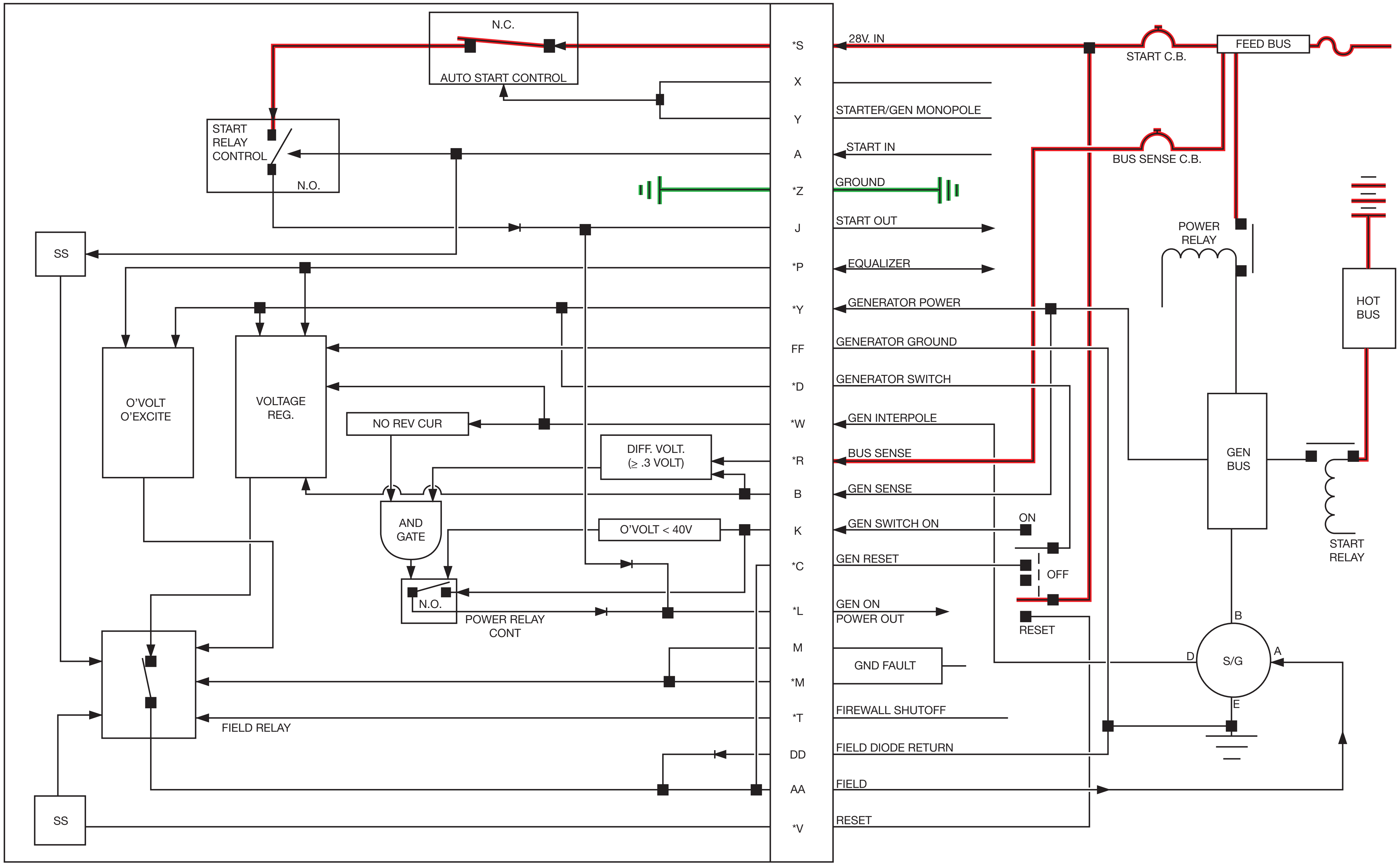

Figure 24-16. Generator Control Unit (GCU) - With Battery Switch On

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BATTERY START RH ENGINE

CONDITIONS

1. Battery start of the right engine.

OBJECTIVES

1. Close the right start relay.
2. Engage latching circuit for start relay and K1.

SEQUENCE OF EVENTS

1. Pressing the right start button provides momentary ground input on pin 23 of right start PCB.
2. Power output from PAL causes K1 PAL relay to initially energize, closing 2 sets of contacts.
   A. One set of contacts short Pins 21 & 22 of start PCB together disabling ground fault sensing circuit of GCU.
   B. Other set of contacts apply power to pin 10 of the start PCB. The following circuits are affected.
      (1) Overhead floodlights ON full bright
      (2) Power to Ignition Power Relay (no effect until throttle is positioned in idle).
      (3) Power to pin 7 of fuel PCB causing fuel boost pump to come ON.
      (4) Power to pin A of GCU.
3. Power on pin A of GCU supplies power on pins J and L to pins 3 & 4 of the right start PCB.
4. Power to PAL from pin 3 of start PCB activates the following circuits.
   1. Ground on pin 6 of right start PCB energizes the start relay closed.
   2. Start relay circuit and PAL relay (K1) circuit are latched ON.
5. Start switch is released removing ground on pin 23 of start PCB.

NOTES

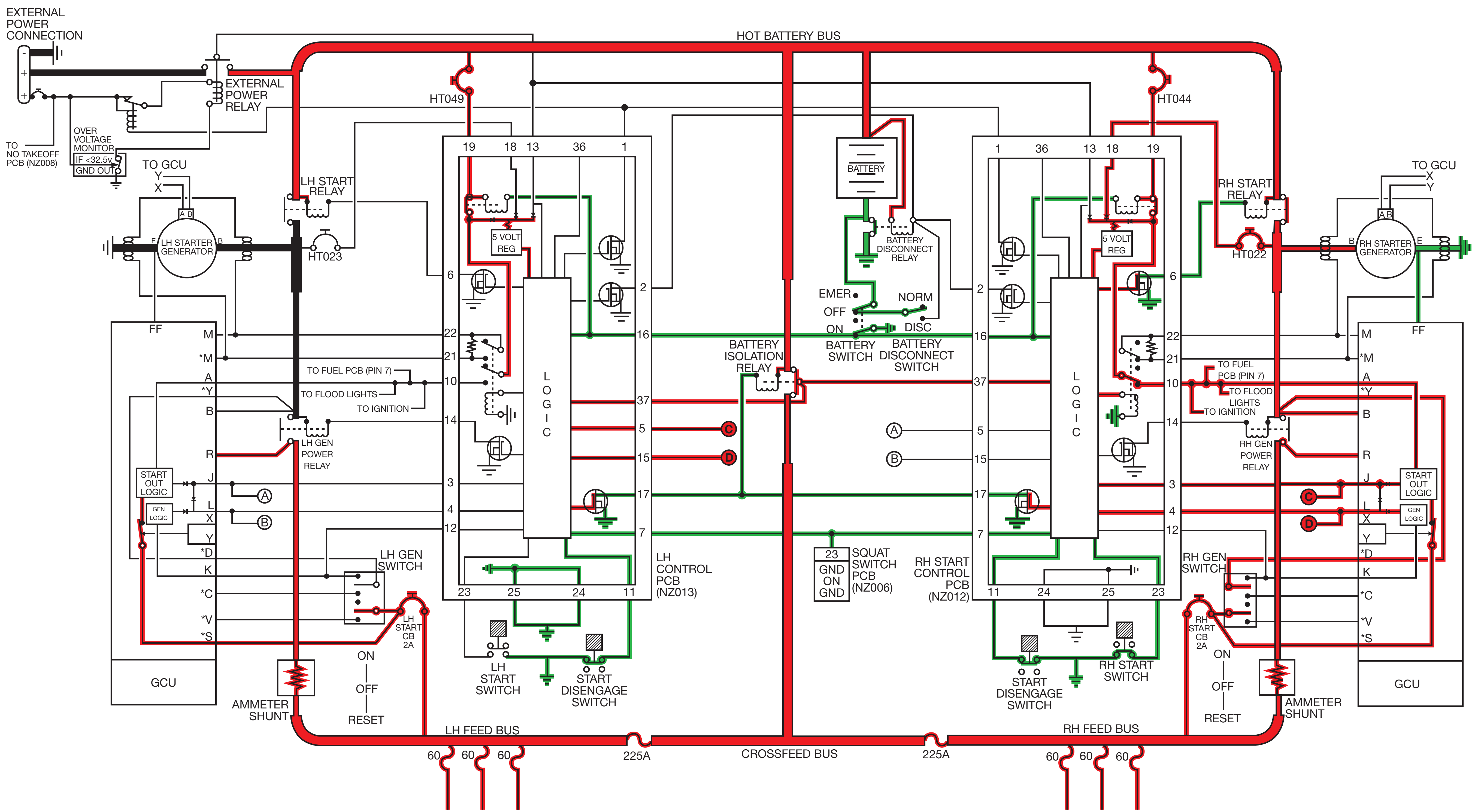

Figure 24-17. Simplified Electrical System - Battery Start RH Engine

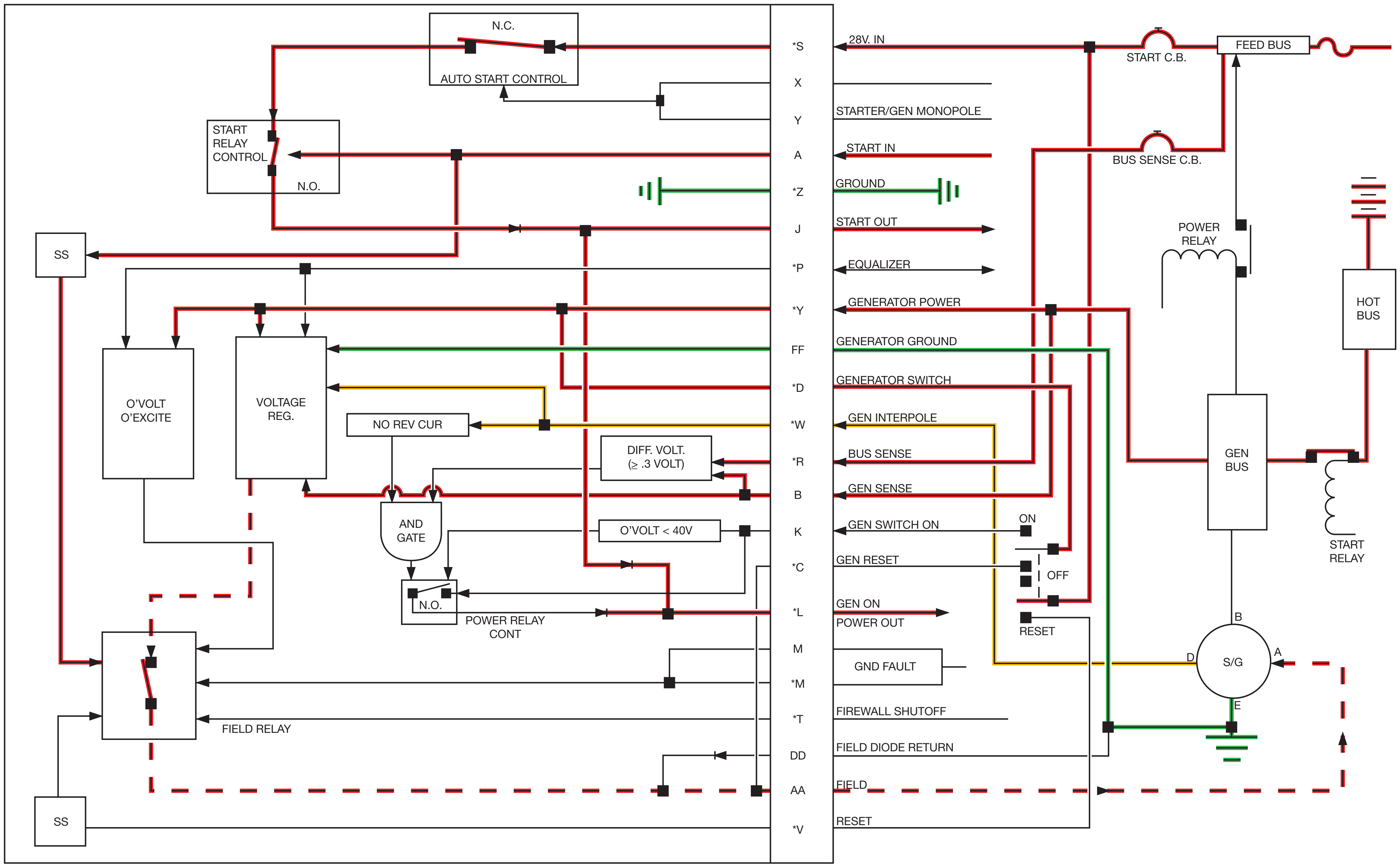

Figure 24-18. Generator Control Unit (GCU) - During Engine Start

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RH GENERATOR ON LINE

CONDITIONS

1. Engine speed of approximately 42-46% N2.

OBJECTIVES

1. Release latching circuit terminating start of right engine.
2. Close power relay to connect right generator to feed bus.

SEQUENCE OF EVENTS

1. Start is terminated as:

   A. Speed sensing monopole provides inputs in on pins X & Y of the GCU.

   B. GCU logic interrupts power out on pin J of the GCU.

   C. This releases the latching circuit of pin 3 of the PCB affecting the following circuits.

      (1) PAL relay (K1) opens.

      (2) Start relay circuit opens as ground is removed on pin 6.

2. Power relay is closed as:

   A. If generator switch is ON, power is applied to pin 12 of the start PCB.

   B. GCU logic allows power to continue out on pin L of GCU to pin 4 of start PCB.

   C. Power in on pins 4 & 12 to the PAL energizes the circuit that provides a ground for pin 14.

3. An input on pin 18 of the right start PCB provides generator power to the PAL.

NOTES

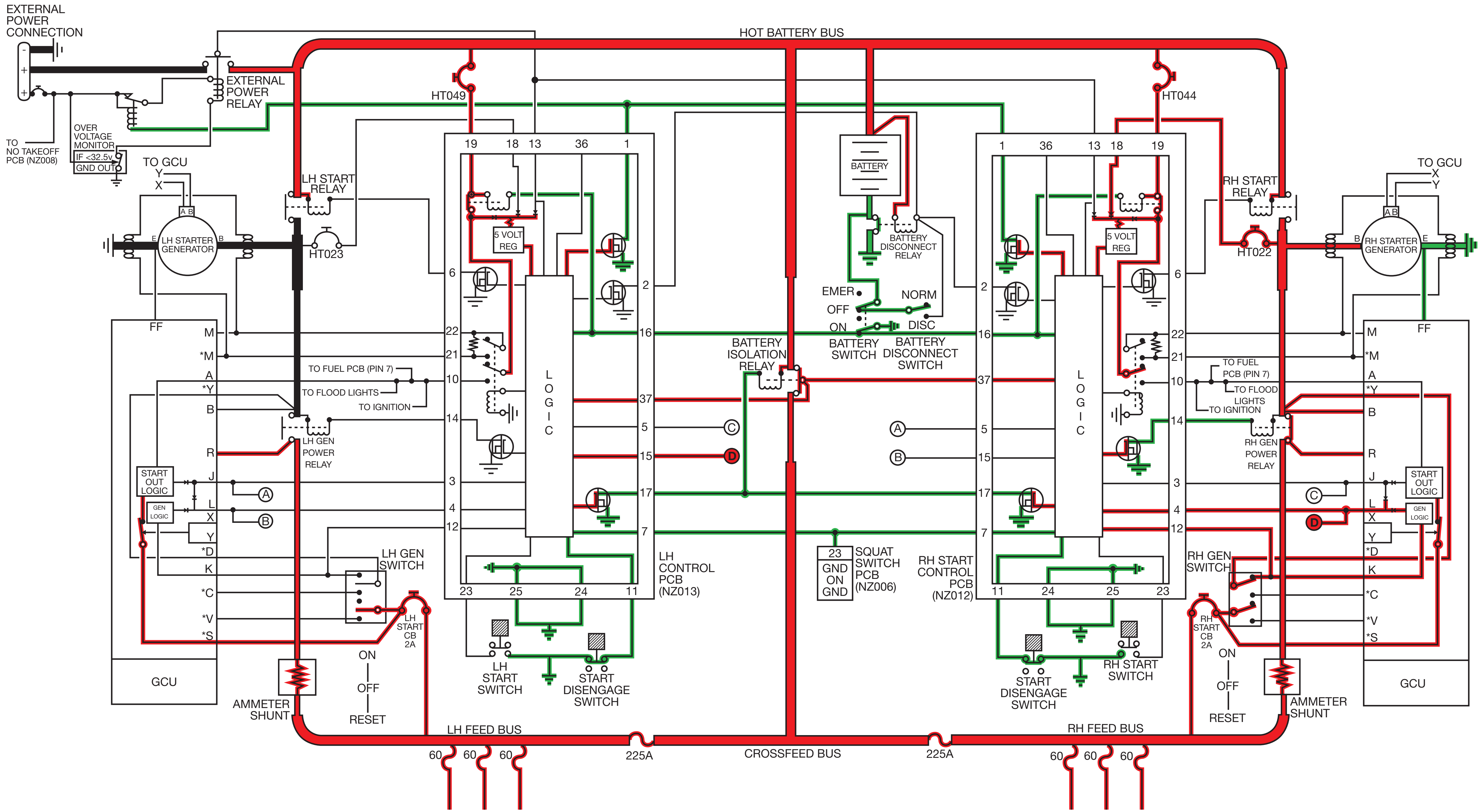

Figure 24-19. Simplified Electrical System - RH Generator Online

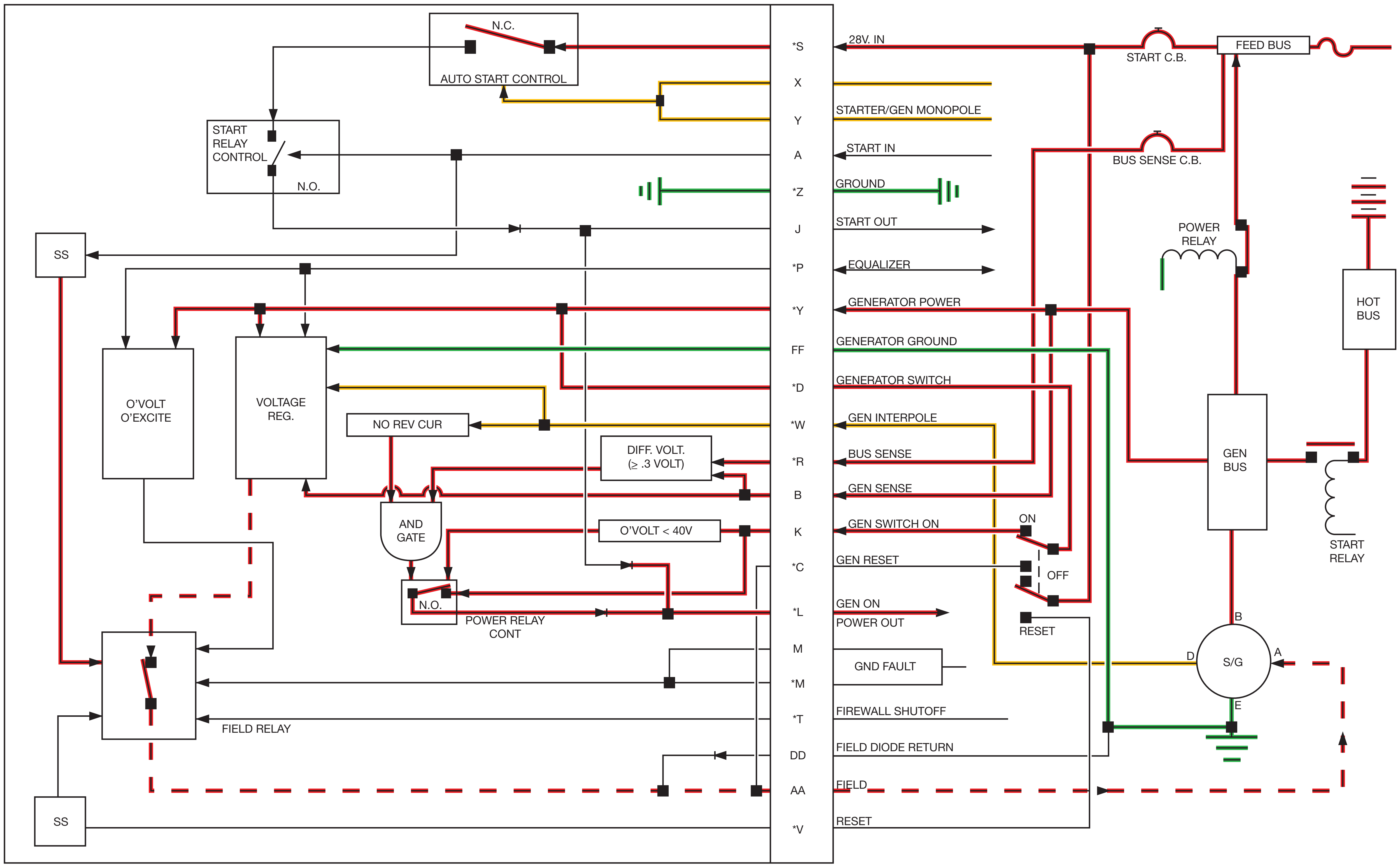

Figure 24-20. Generator Control Unit (GCU) - With Generator Online

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GENERATOR ASSIST START ON LH ENGINE

CONDITIONS

1. Generator assisted start of left engine.

OBJECTIVES

1. Open the battery relay.
2. Close both right and left start relays.

SEQUENCE OF EVENTS

1. Squat Switch PCB (NZ006) provides a ground on pin 7 of both start PCB's.
2. With generator on line, power out of pin L of right GCU is applied to pin 15 of left start PCB.
3. Momentary ground on pin 23 of left start PCB affects the following circuits.
   - A. Opens the ground circuit on pin 17 of the left start PCB.
   - B. Initiates the start of the left engine.
4. Power output from PAL causes K1 PAL relay to initially energize, closing 2 sets of contacts.
   - A. One set of contacts short Pins 21 & 22 of start PCB together disabling ground fault sensing circuit of GCU.
   - B. Other set of contacts applies power to pin 10 of the start PCB. The following circuits are affected.
     - (1) Overhead floodlights ON full bright
     - (2) Power to pin 5 of ignition PCB (no effect until throttle is positioned in idle).
     - (3) Power to pin 7 of fuel PCB causing fuel boost pump to come ON.
     - (4) Power to pin A of GCU.
5. Power on pin A of GCU supplies power on pins J and L to pins 3 & 4 of the left start PCB and pins 5 and 15 of right start PCB.
6. Power to PAL from pin 3 of left start PCB activates the following circuits.
   1. Ground on pin 6 of left start PCB energizes the start relay closed after an open appears on pin 37.
   2. Start relay circuit and PAL relay (K1) circuit are latched ON.
7. Power in on pins 5 & 15 combined with generator power on pins 4 & 12 of the right start PCB affects the following circuits.
   - A. Ground on pin 17 of right start PCB opens. Battery relay opens.
   - B. Causes a ground on pin 6 of the right start PCB when an open appears on pin 37. The right start relay closes.

NOTES

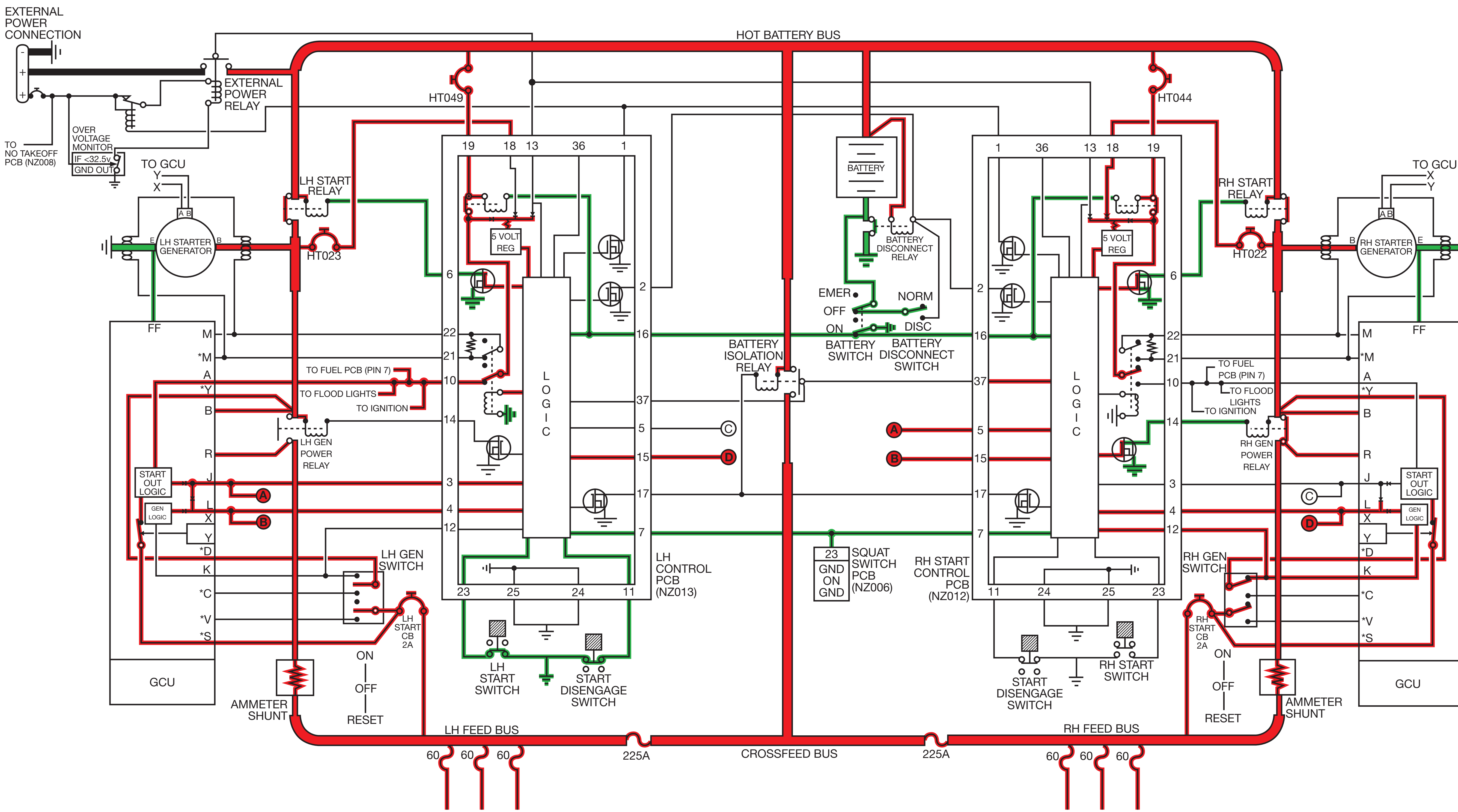


Figure 24-21. Simplified Electrical System - Generator Assist Start On LH Engine

24

EXTERNAL POWER CONNECTED

CONDITIONS

1. External power connected.

OBJECTIVES

1. Connect external power to the power distribution system.

SEQUENCE OF EVENTS

1. Connect external power plug to receptacle.

   A. External power Unit voltage passes through the External Power Control Relay's relaxed contact and goes to the External Power Relay.

   B. External Power Unit voltage is supplied to the overvoltage monitor.

   C. If the EPU voltage is < 32.5 vdc, then the overvoltage monitor will supply a ground for the External Power Relay.

2. External power relay closes applying power to hot battery bus.

   A. Power in on pin 13 of both start PCB's

3. Position battery switch in BATT.

   A. Both K2 PCB relays energized closed because of ground provided on pin 16 of both start PCB's through the battery switch.

   B. Discrete ground input to PAL provided on pin 16 of both start PCB's through the battery switch.

   C. PAL causes ground on Pin 17 of right and left start PCB's energizing the battery relay closed.

   D. External power charges battery and is connected to crossfeed bus.

NOTE

If there is an input to pin 4 (generator on) or pin 15 (other side generate) of the either start PCB with an open on pins 3 or 5, there will be a ground supplied on pin 1 and the external power relay will open.

NOTES

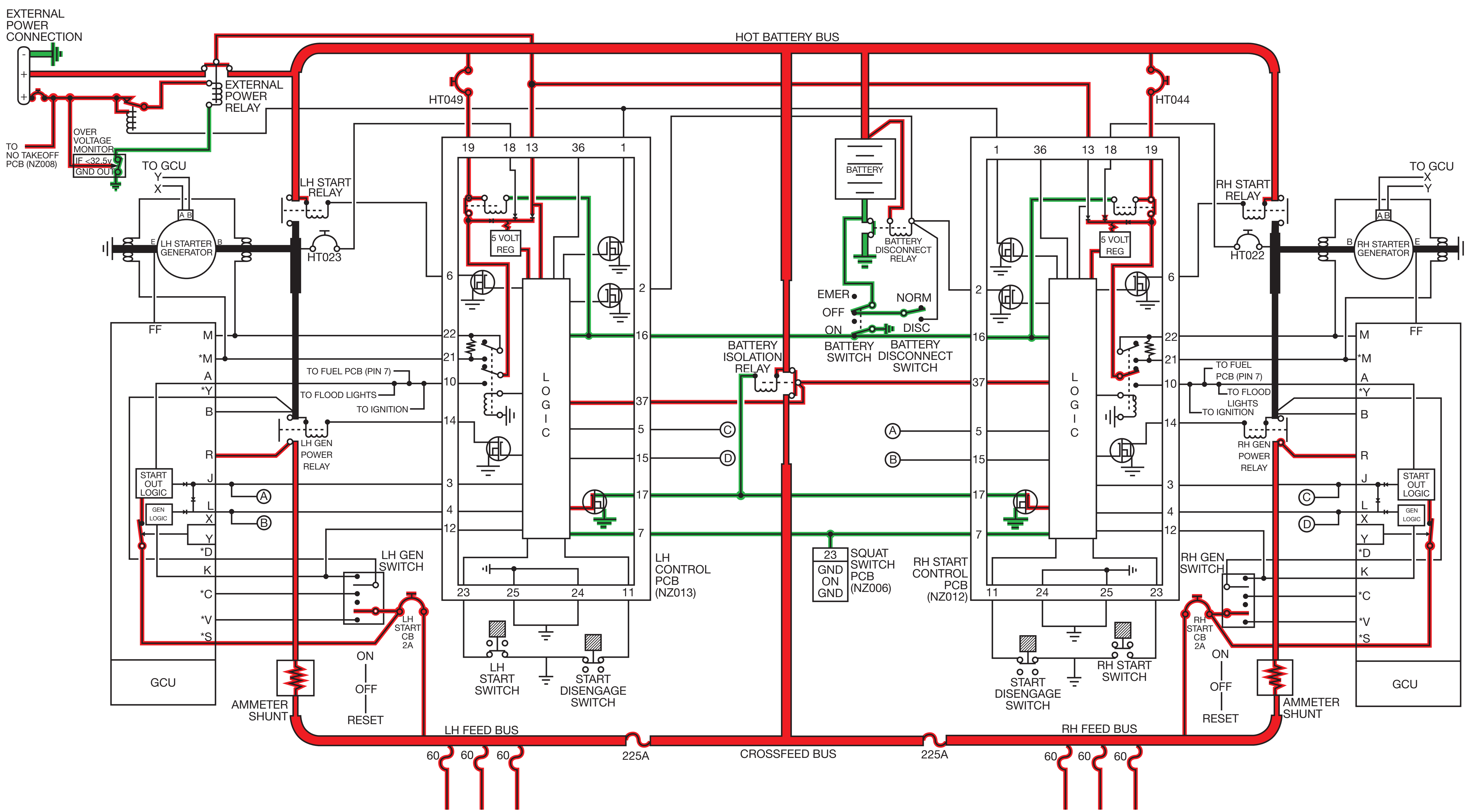


Figure 24-22. Simplified Electrical System - External Power Connected

EXTERNAL POWER START RH ENGINE

CONDITIONS

1. External power start (right engine)

OBJECTIVES

1. Energize open the battery disconnect relay
2. Initiate the right start circuit.
3. Latch the start circuit.

SEQUENCE OF EVENTS

1. Ground on pin 7 of both start PCB's.
2. Ground on pin 23 of right start PCB.
3. Battery disconnect relay opens when either of the following occurs:

   A. Power in on pins 13 (external power on), 3 & 4 (start power from right GCU) provides a ground on pin 2 of the right start PCB.

   B. Power in on pins 13 (external power on), 5 & 15 (start power from right GCU) provides a ground on pin 2 of the left start PCB.

4. Right start relay closes when:

   A. Power output from PAL causes K1 PAL relay to initially energize, closing 2 sets of contacts.

      (1) One set of contacts short Pins 21 & 22 of start PCB together disabling ground fault sensing circuit of GCU.

      (2) Other set of contacts apply power to pin 10 of the start PCB. The following circuits are affected.

         a. Overhead floodlights ON full bright

         b. Power to pin 5 of ignition PCB (no effect until throttle is positioned in idle).

         c. Power to pin 7 of fuel PCB causing fuel boost pump to come ON.

         d. Power to pin A of GCU.

   B. Power on pin A of GCU supplies power on pins J and L to pins 3 & 4 of the right start PCB.

   C. Power to PAL from pin 3 of start PCB activates the following circuits.

      (1) Ground on pin 6 of right start PCB energizes the start relay closed.

      (2) Start relay circuit and PAL relay (K1) circuit are latched ON.

NOTES

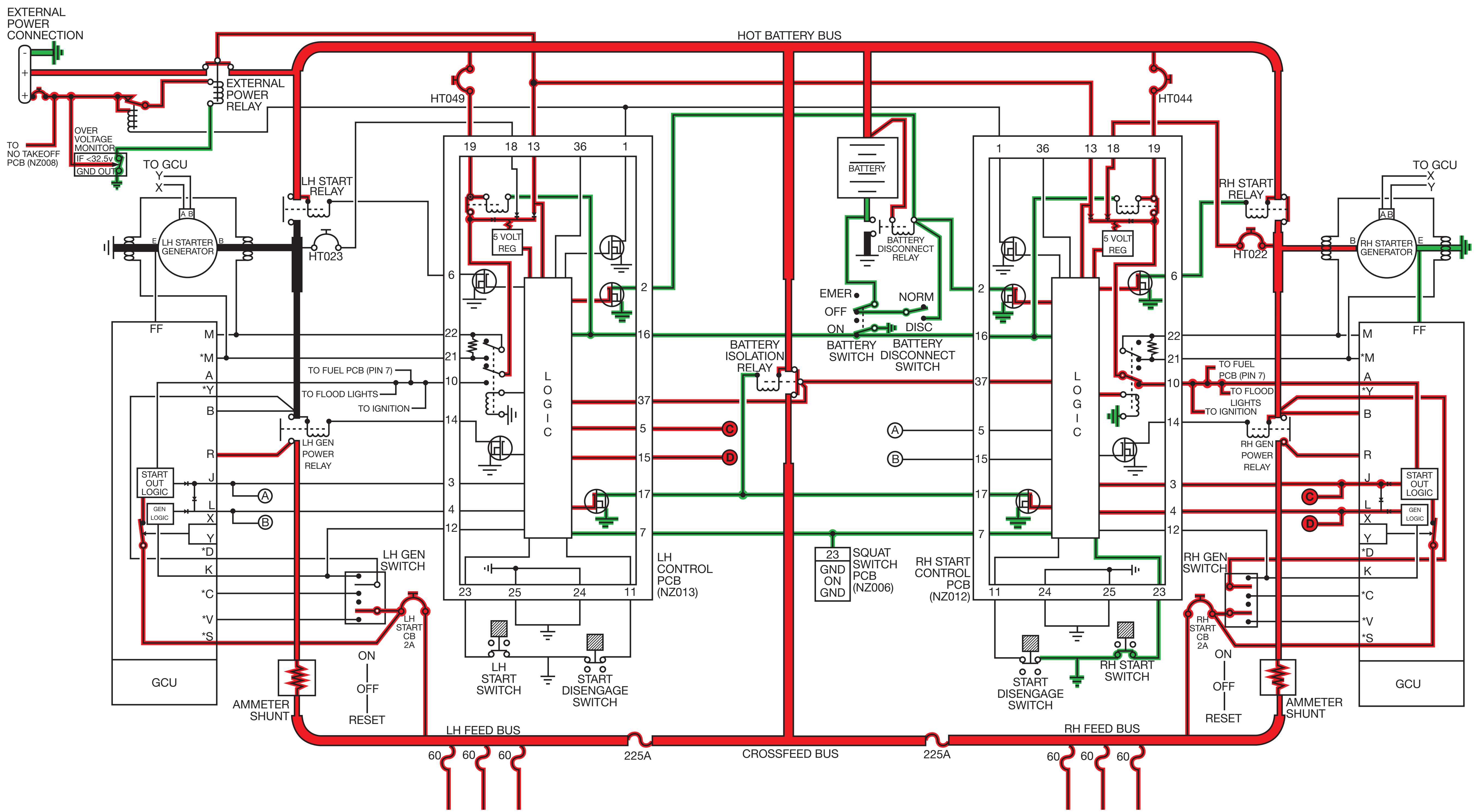
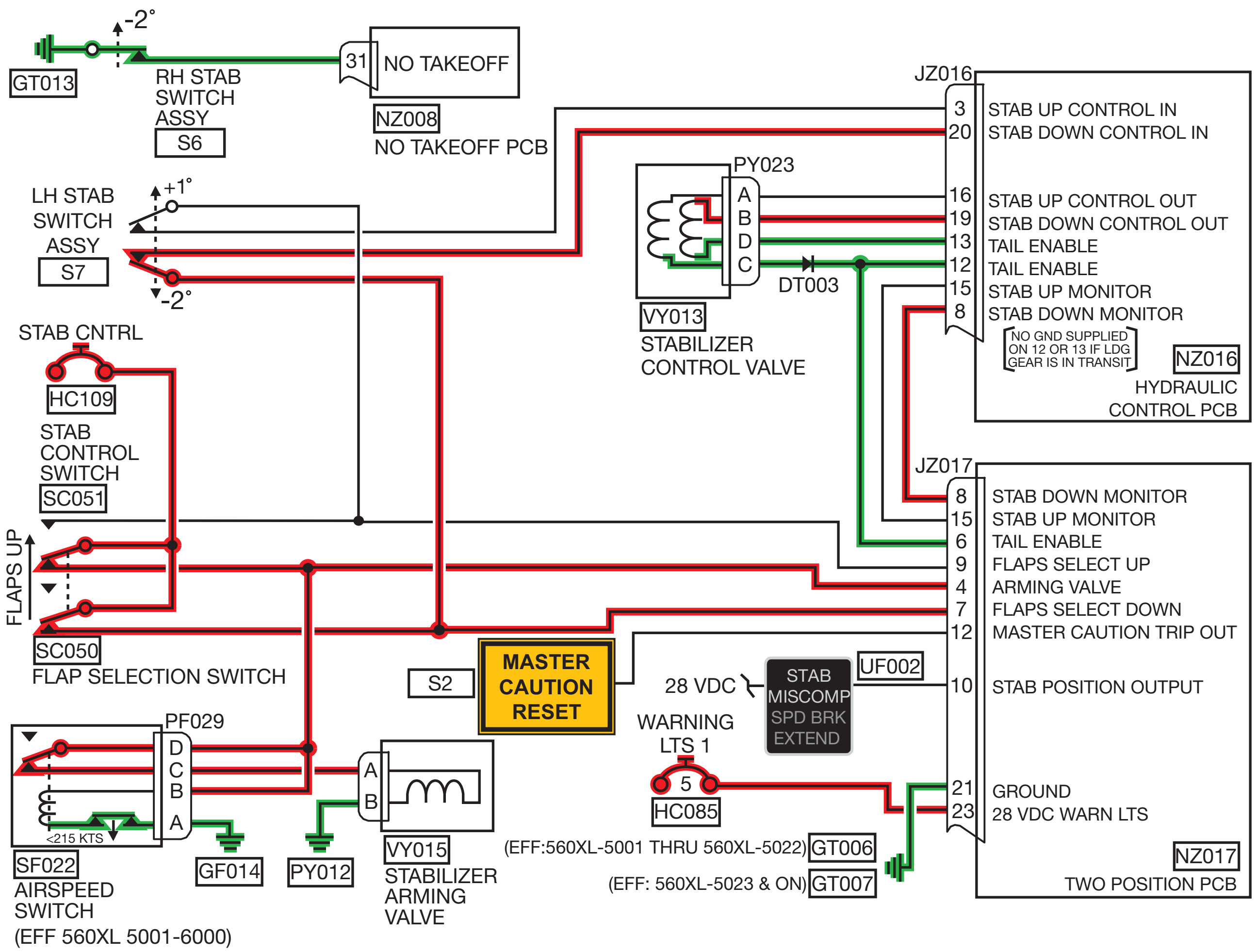
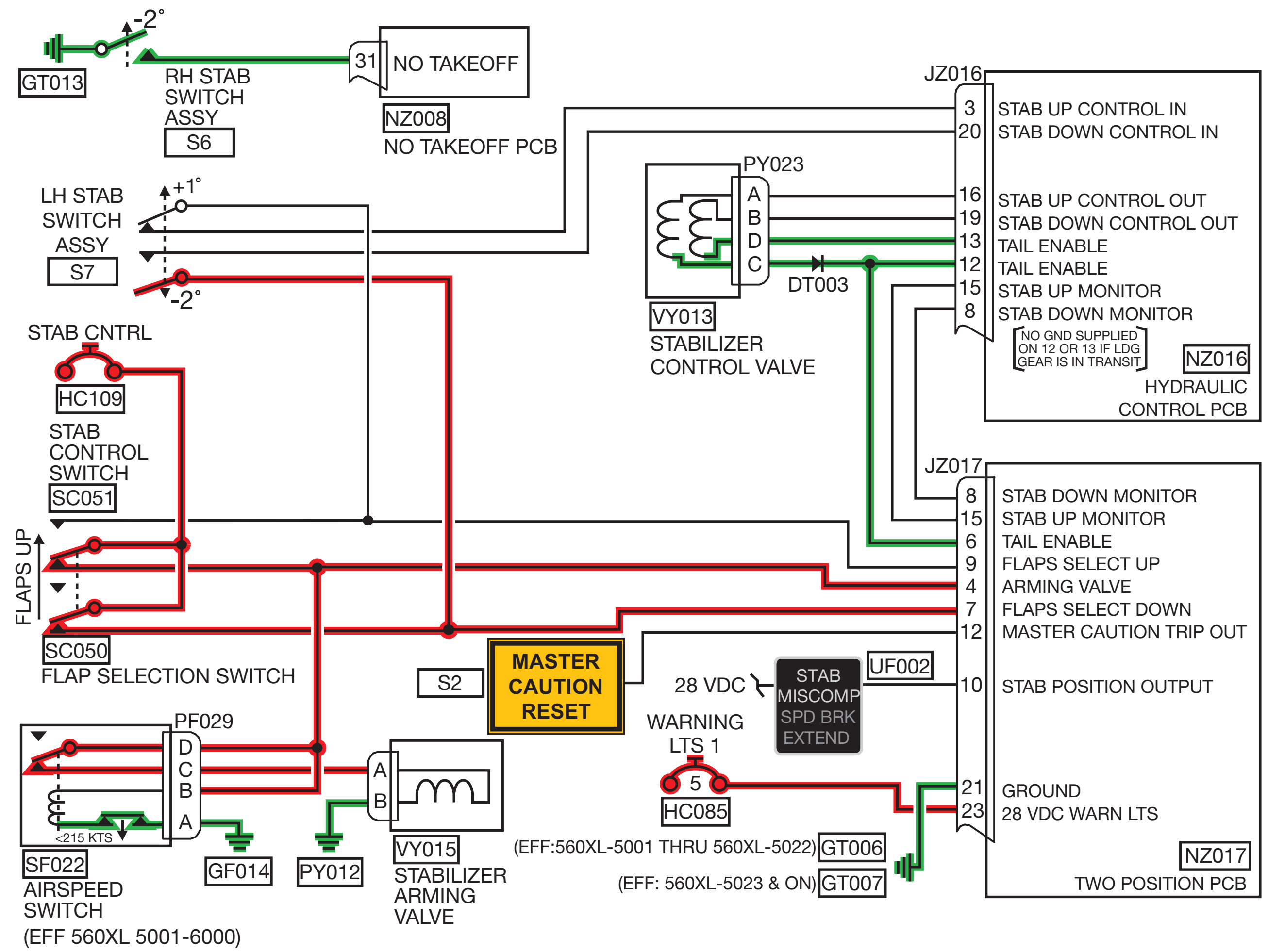


Figure 24-23. Simplified Electrical System - External Power Start

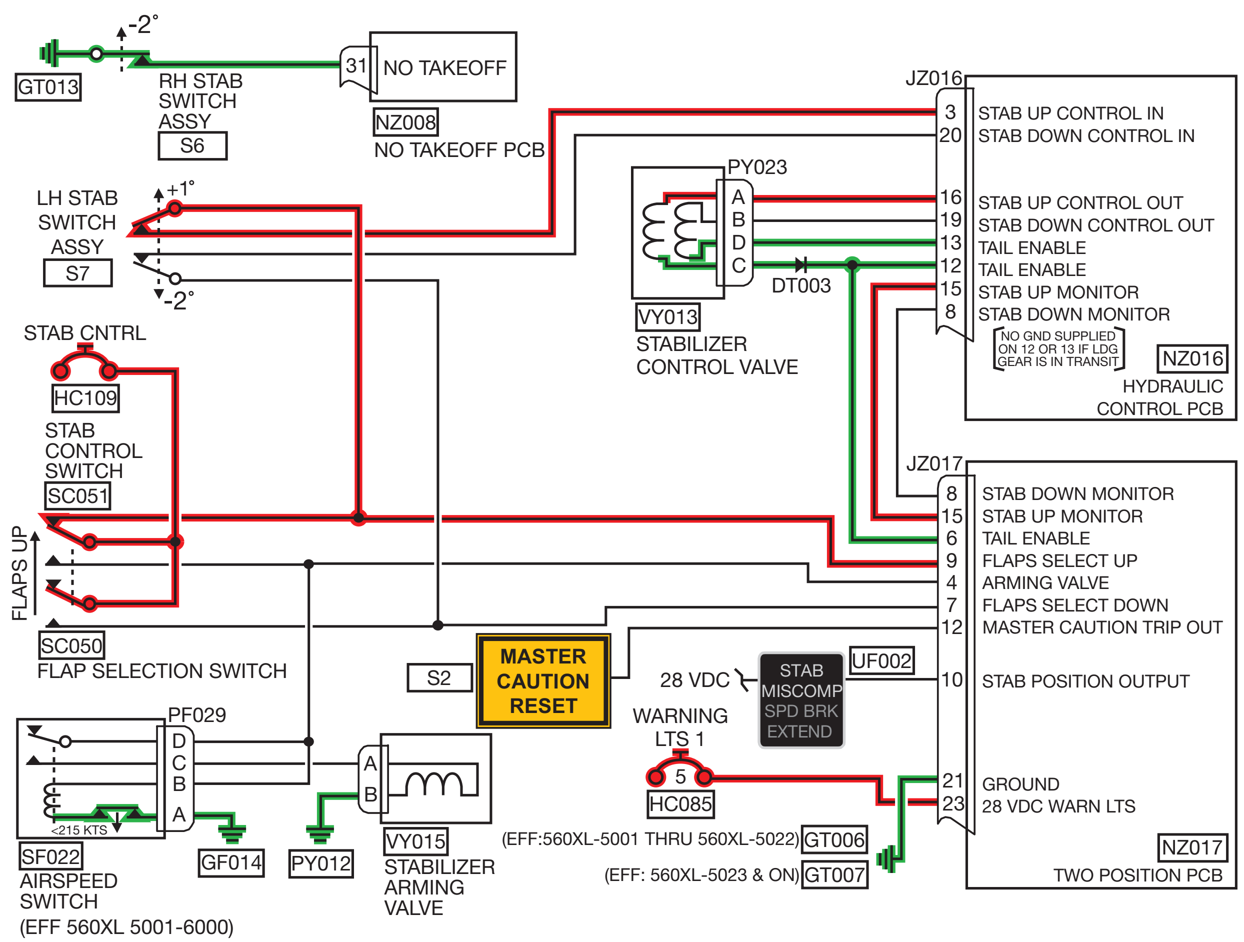


(STABLIZER TRANSITIONING TO TAKEOFF AND APPROACH)

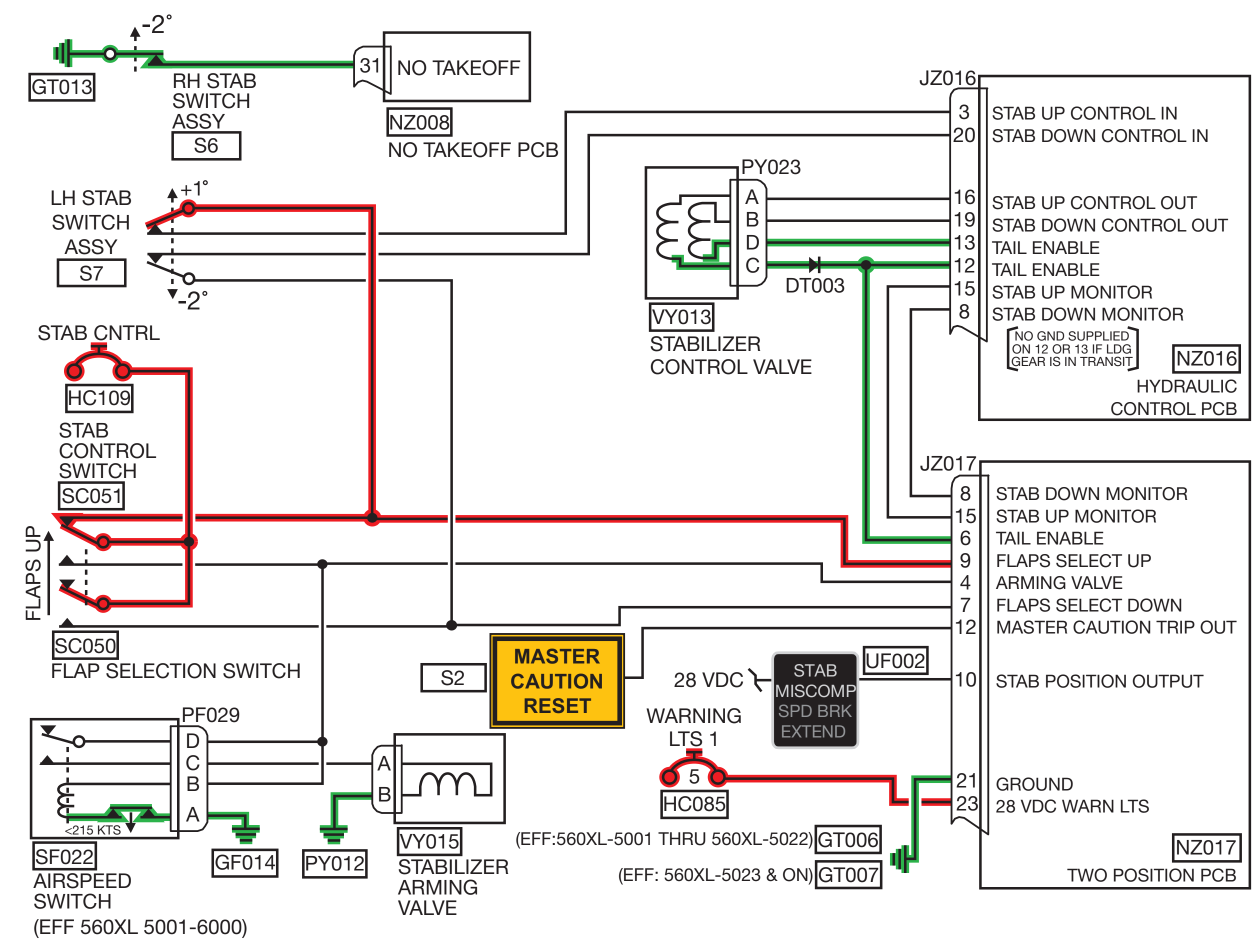


(STABLIZER DOWN AT TAKEOFF AND APPROACH)

Figure 27-1. Horizontal Stabilizer Control (Sheet 1 of 2)



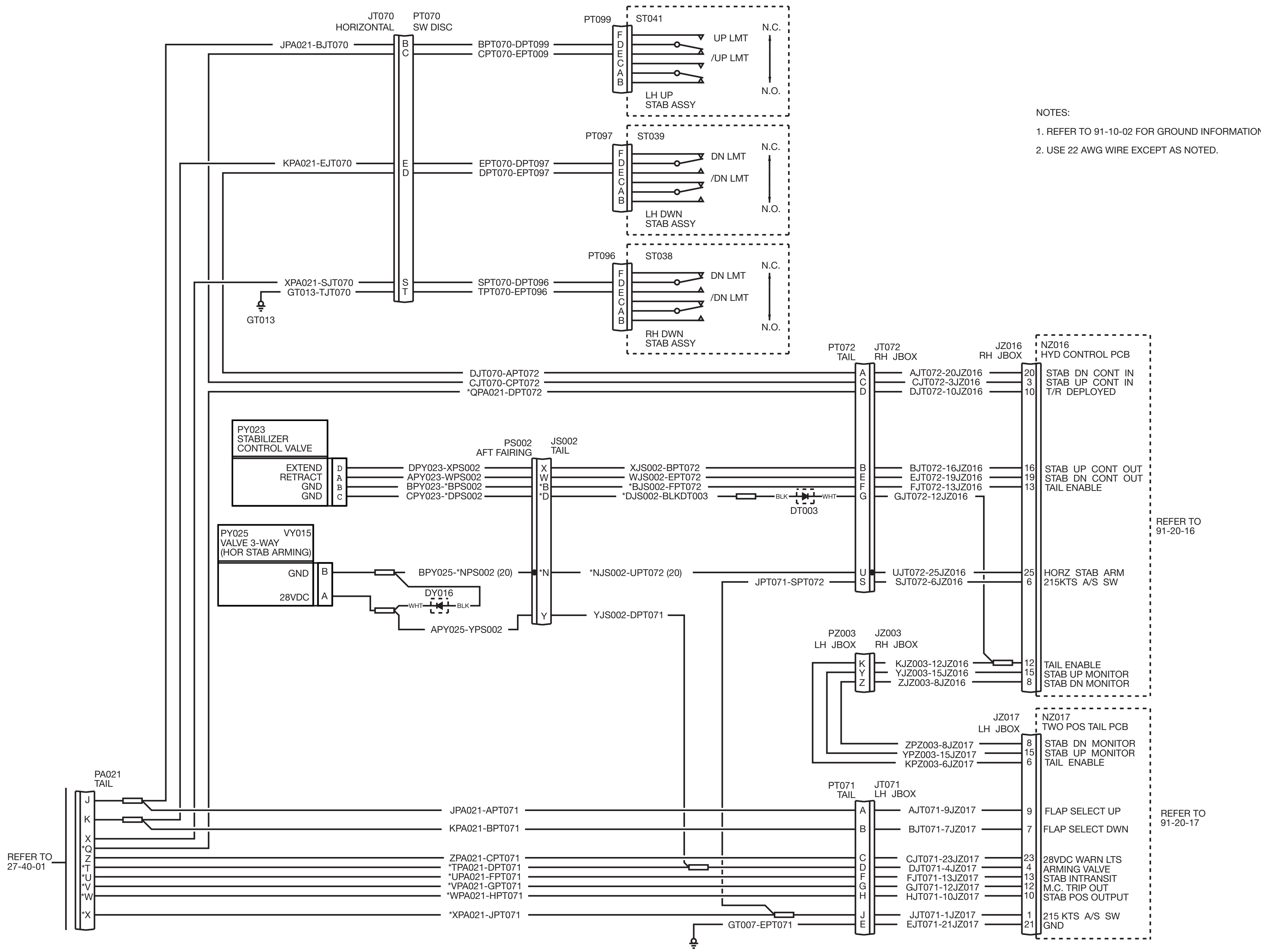
(STABLIZER TRANSITIONING TO CRUISE)



(STABLIZER UP AT CRUISE)

Figure 27-1. Horizontal Stabilizer Control (Sheet 2 of 2)

INTENTIONALLY LEFT BLANK

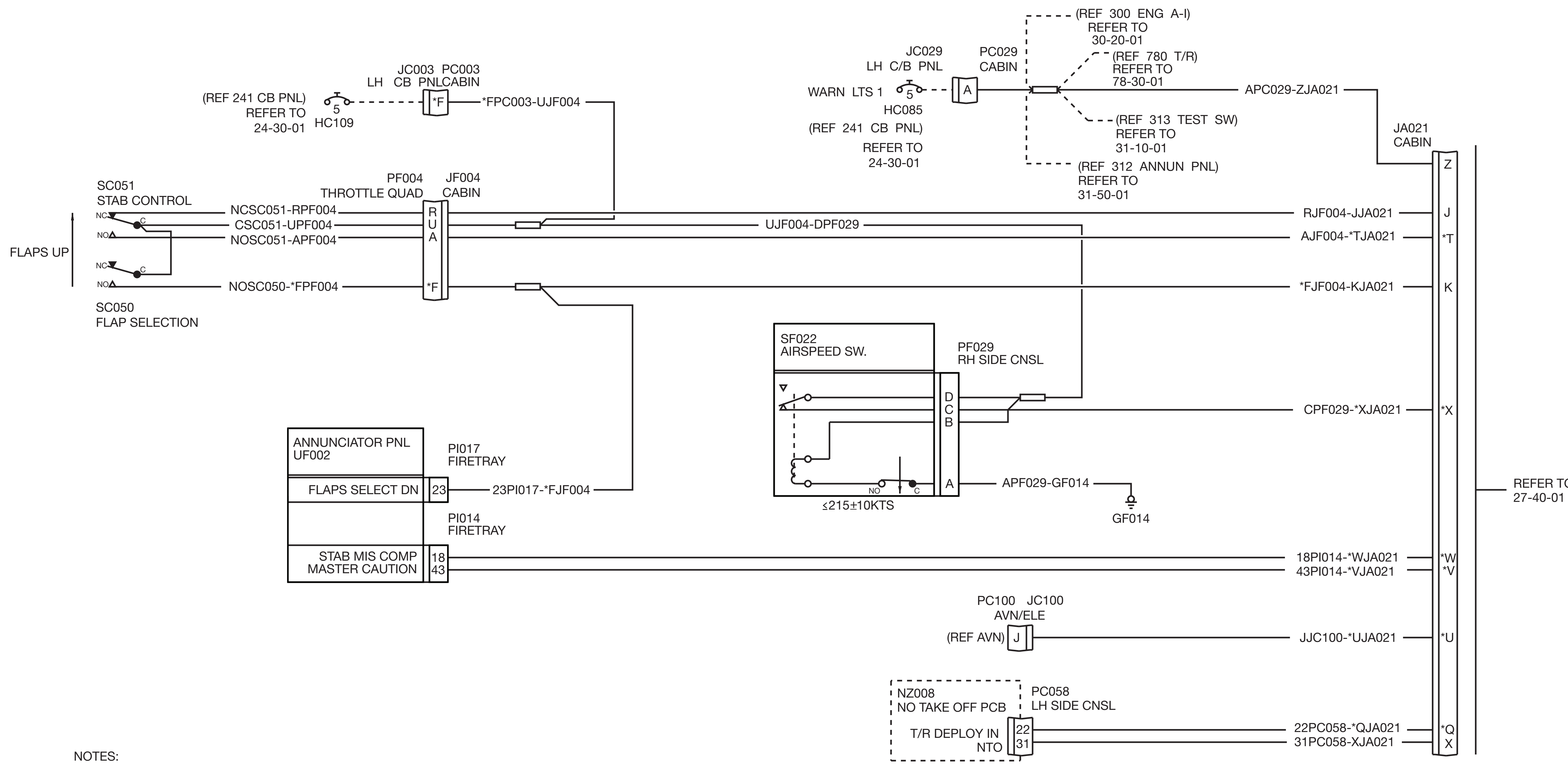


NOTES:

1. REFER TO 91-10-02 FOR GROUND INFORMATION

2. USE 22 AWG WIRE EXCEPT AS NOTED.

Figure 27-2. Two Position Tail - Units 5501-5545 (Sheet 1 of 2)



NOTES:

1. REFER TO 91-10-02 FOR GROUND INFORMATION.

2. USE 22 AWG WIRE EXCEPT AS NOTED.

Figure 27-2. Two Position Tail - Units 5501-5545 (Sheet 2 of 2)

NOTES:

1. REFER TO 91-10-02 FOR GROUND INFORMATION.
2. USE 22 AWG WIRE EXCEPT AS NOTED.

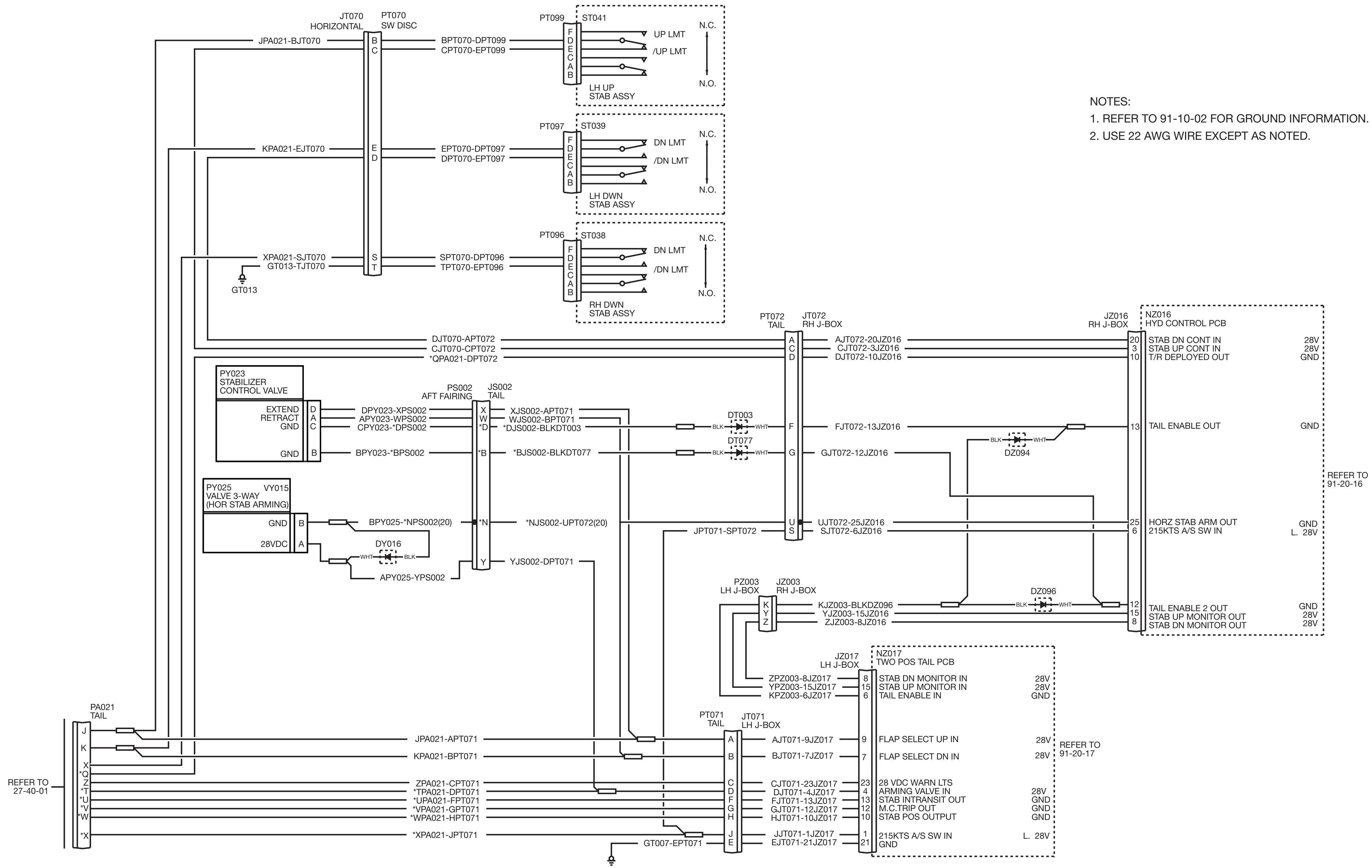
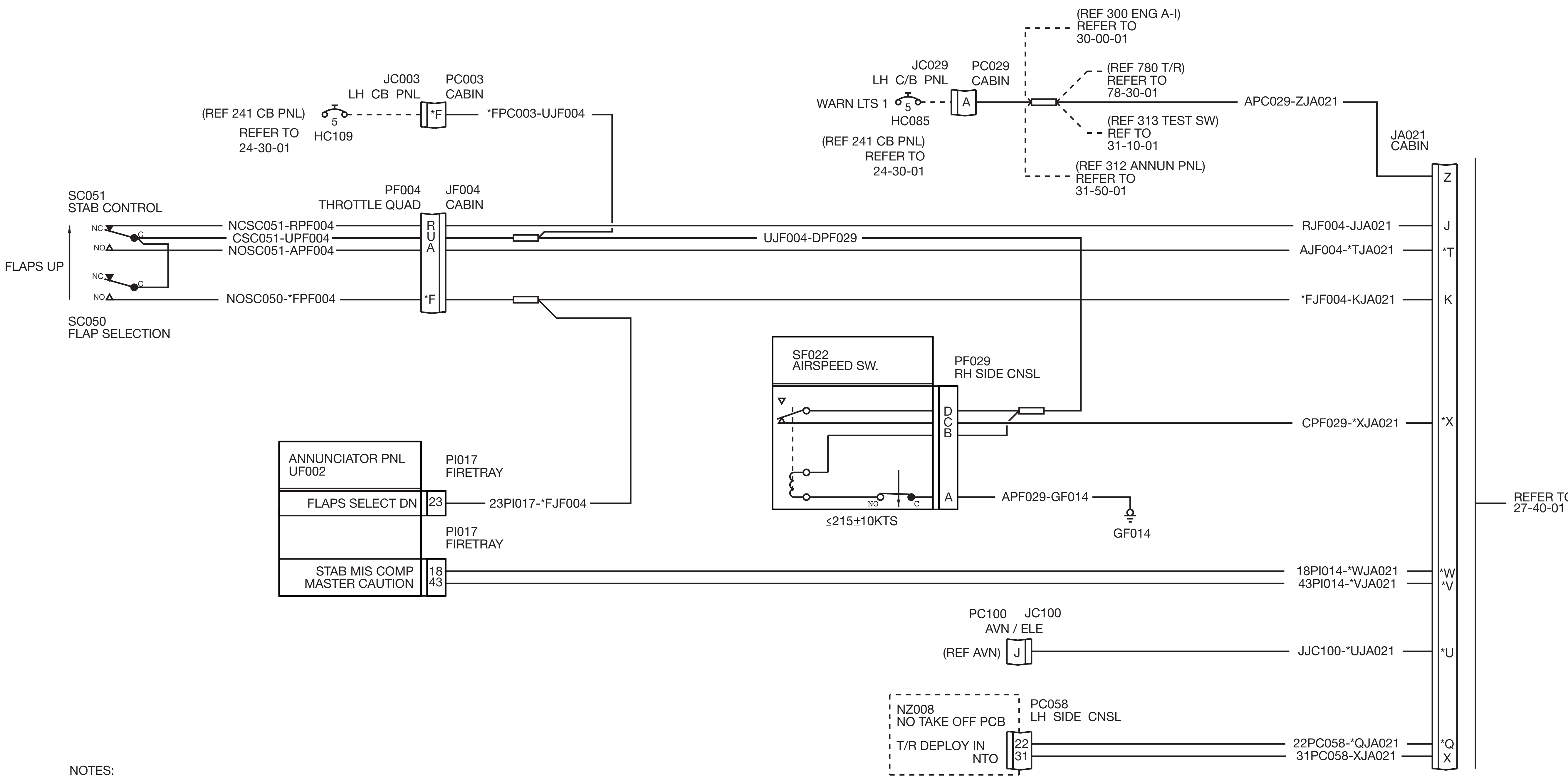


Figure 27-3. Two Position Tail - Units 5546-6000 (Sheet 1 of 2)



NOTES:

1. REFER TO 91-10-02 FOR GROUND INFORMATION.
2. USE 22 AWG WIRE EXCEPT AS NOTED.

Figure 27-3. Two Position Tail - Units 5546-6000 (Sheet 2 of 2)

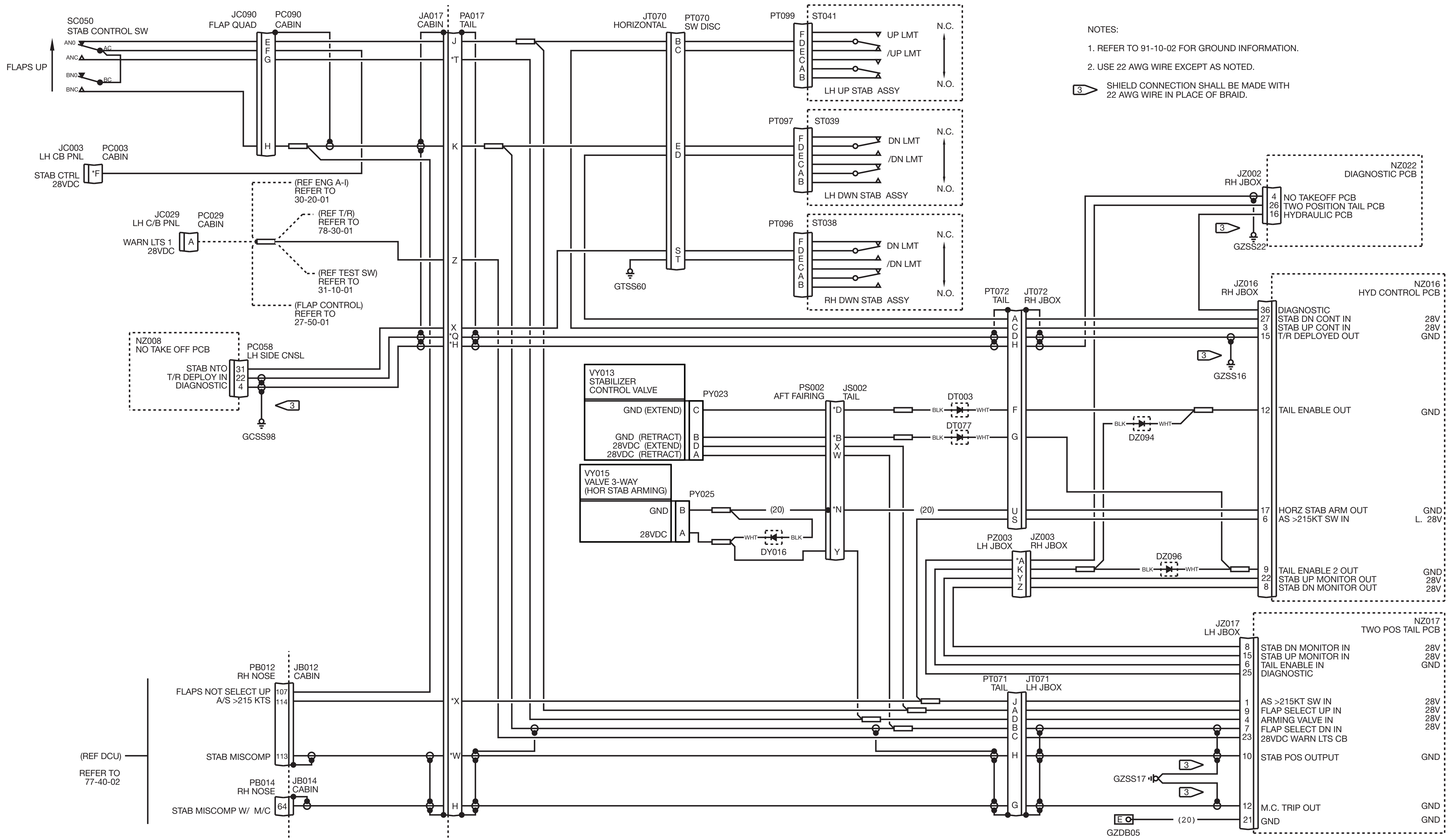
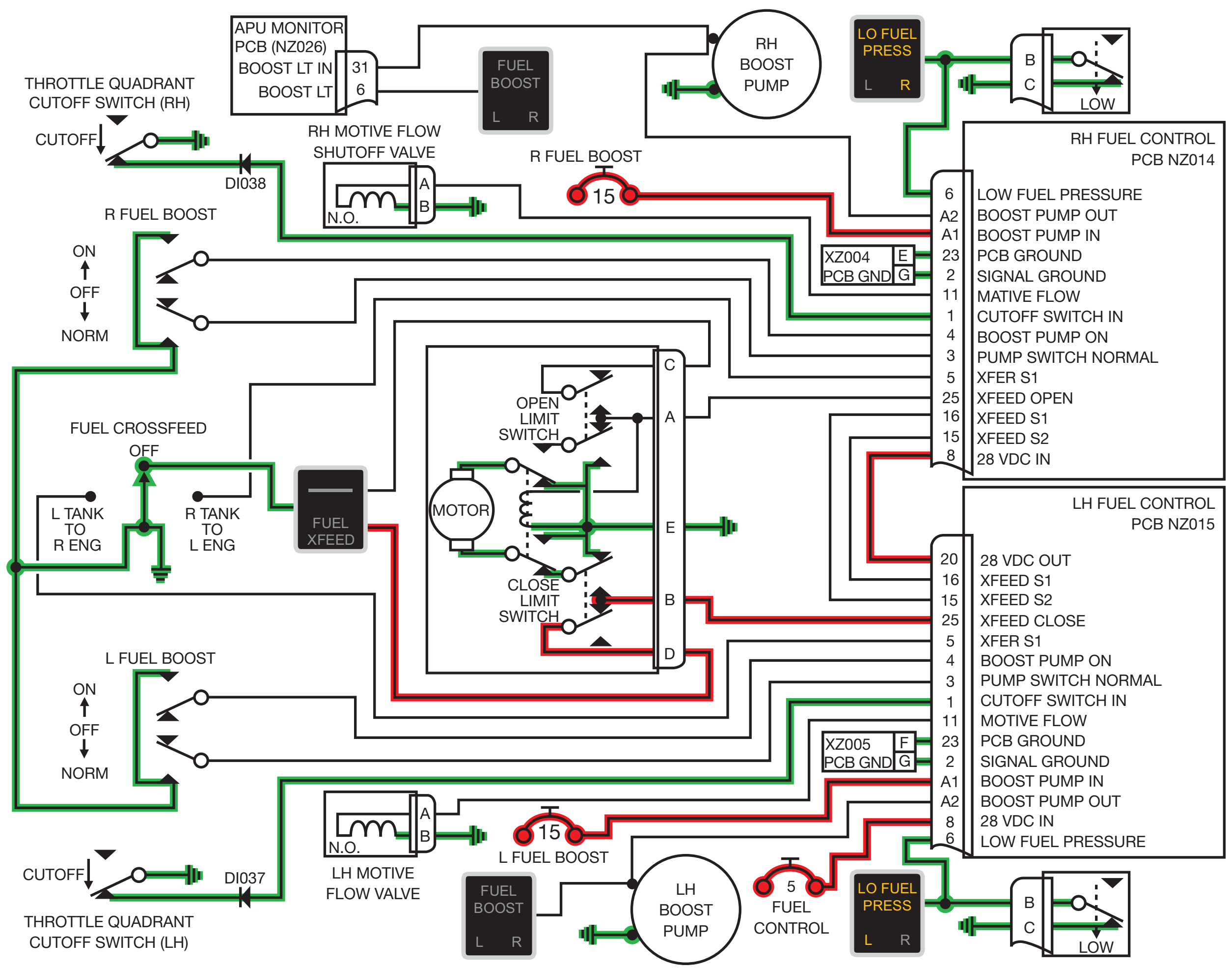
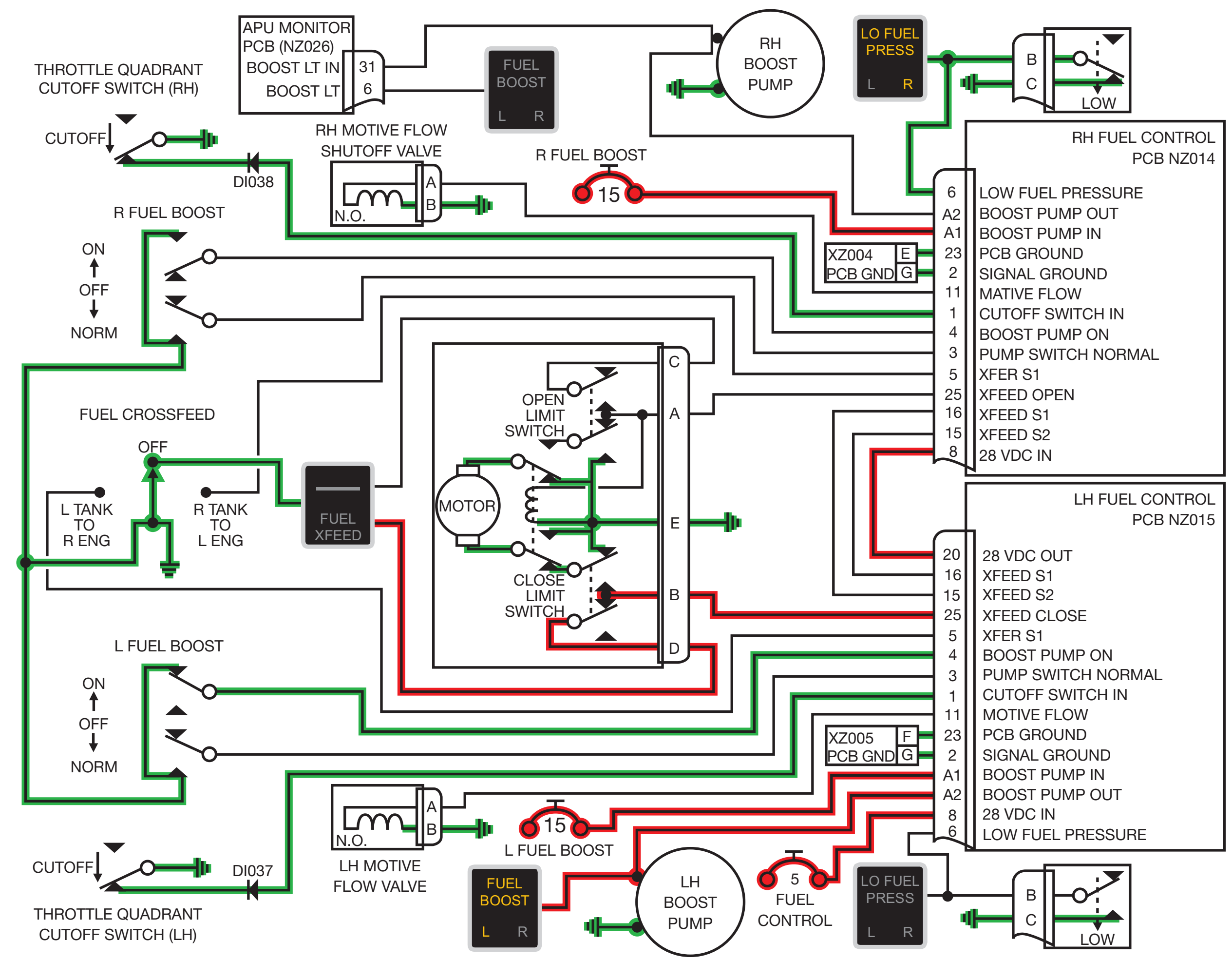


Figure 27-4. Two Position Tail - Units 6001 and Subsequent

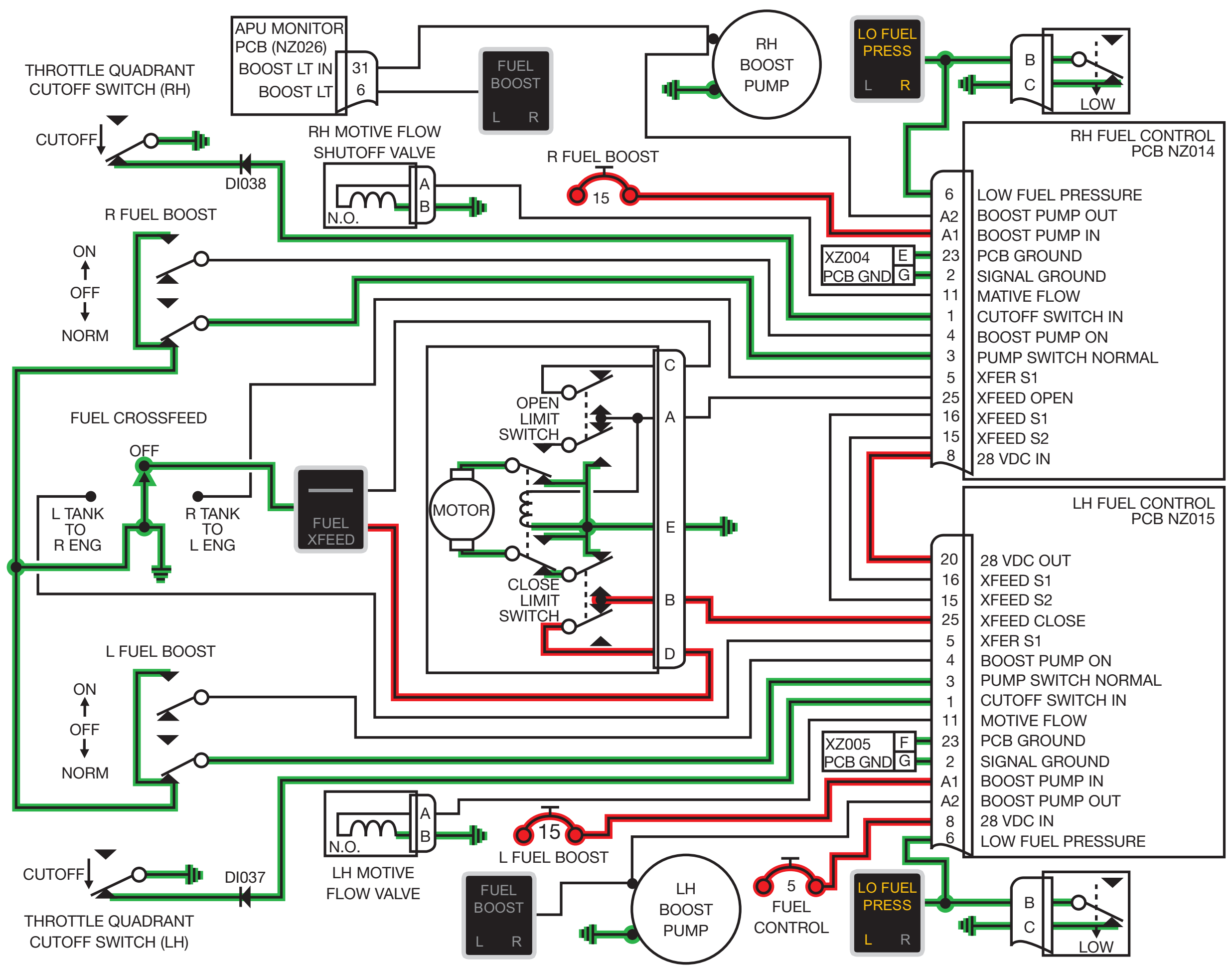


BATTERY SWITCH ON

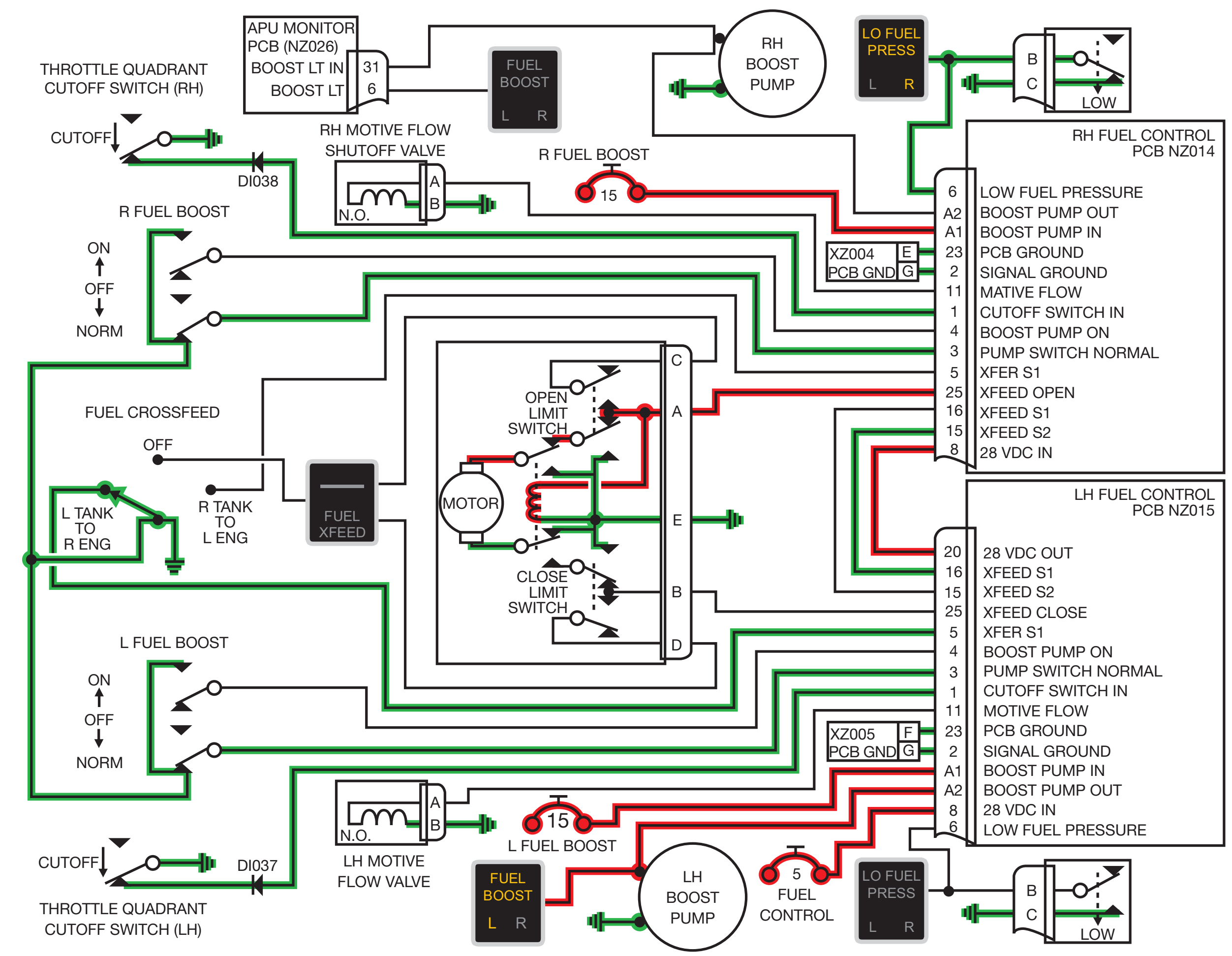


L FUEL BOOST SELECTED ON

Figure 28-1. Fuel Control System (Sheet 1 of 3)

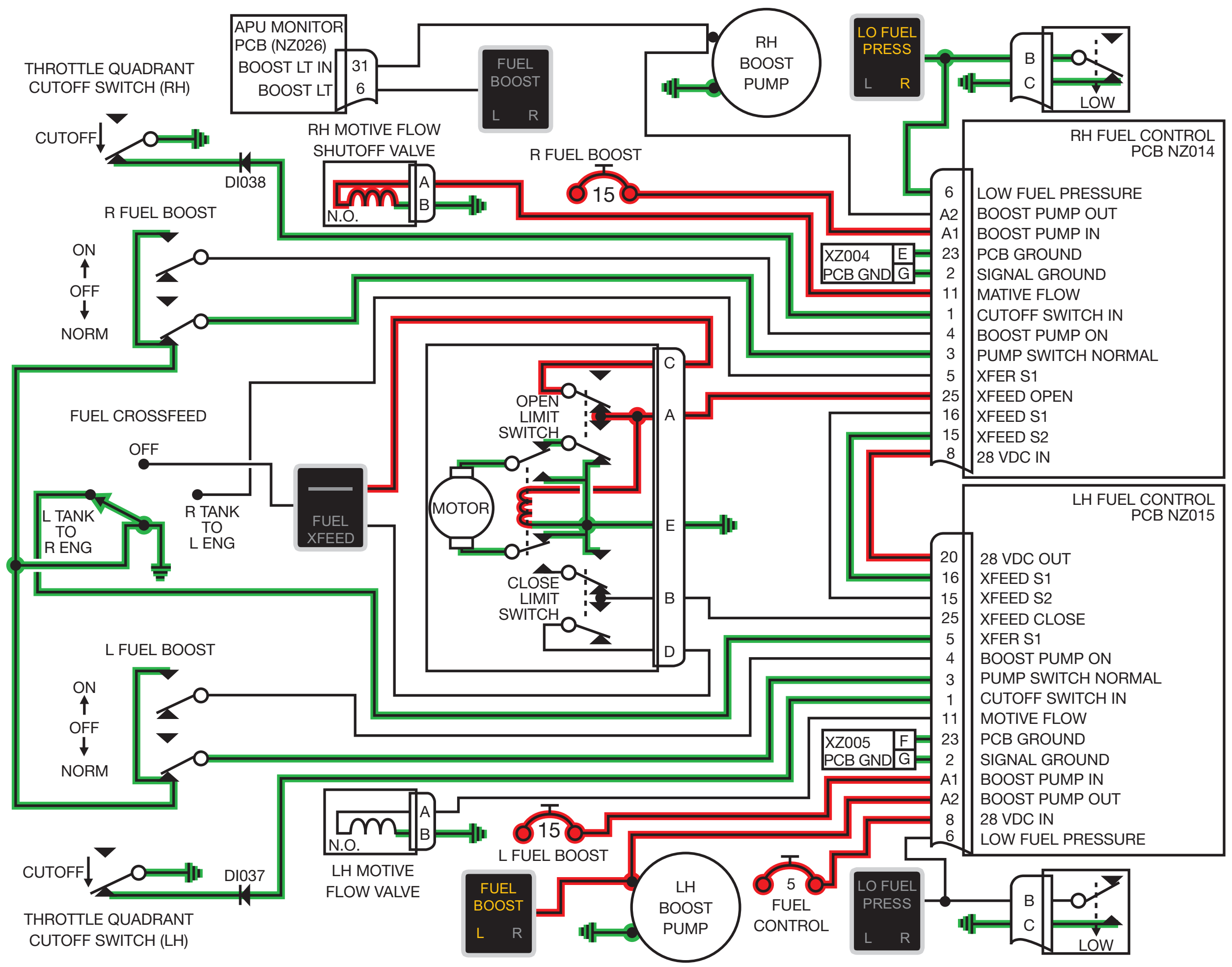


L AND R FUEL BOOST SWITCHES IN NORM

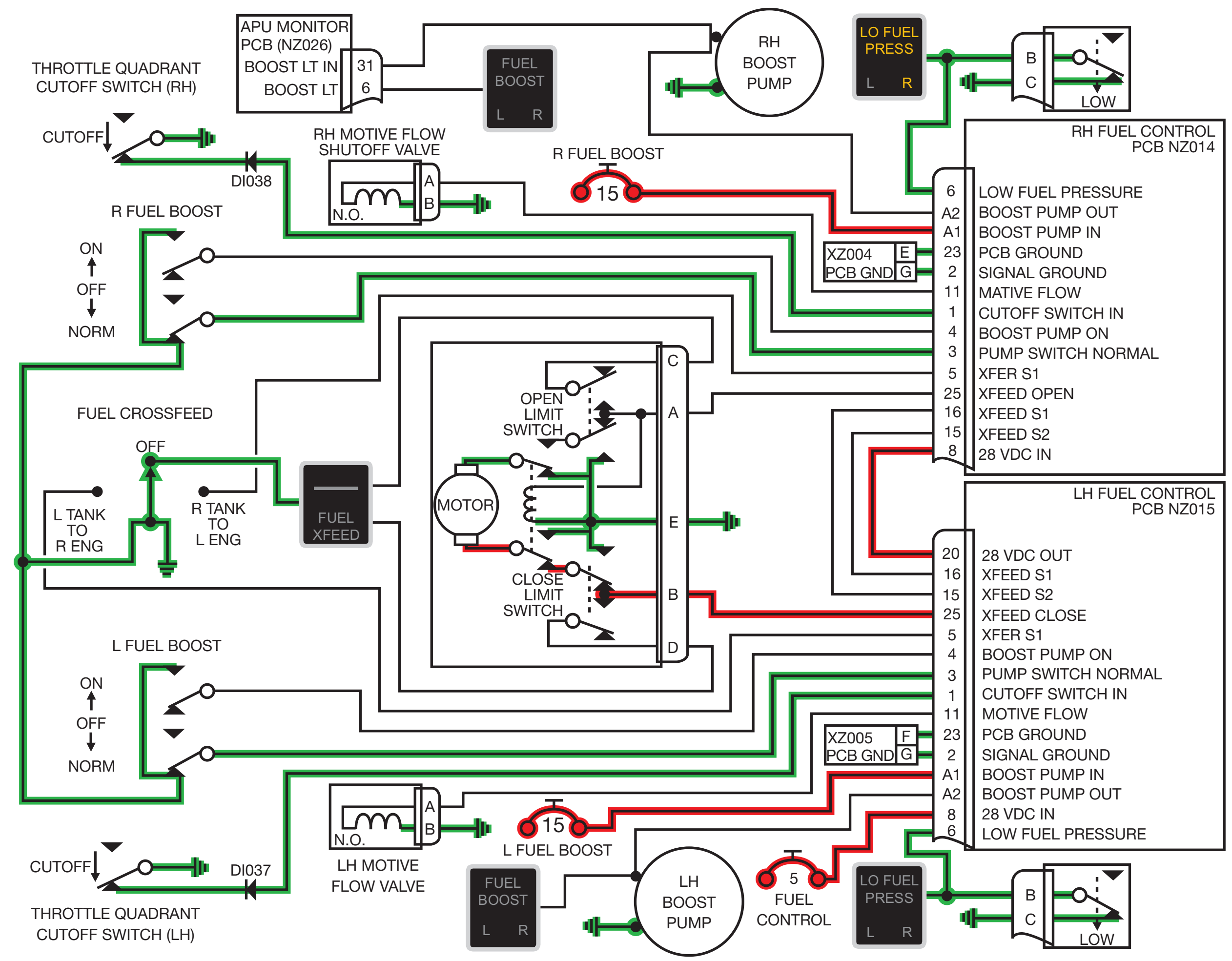


XFEED SELECTED (L TANK TO R ENG)

Figure 28-1. Fuel Control System (Sheet 2 of 3)



XFEED SELECTED (L TANK TO R ENG) AFTER 3 SEC



XFEED SELECTED OFF

Figure 28-1. Fuel Control System (Sheet 3 of 3)

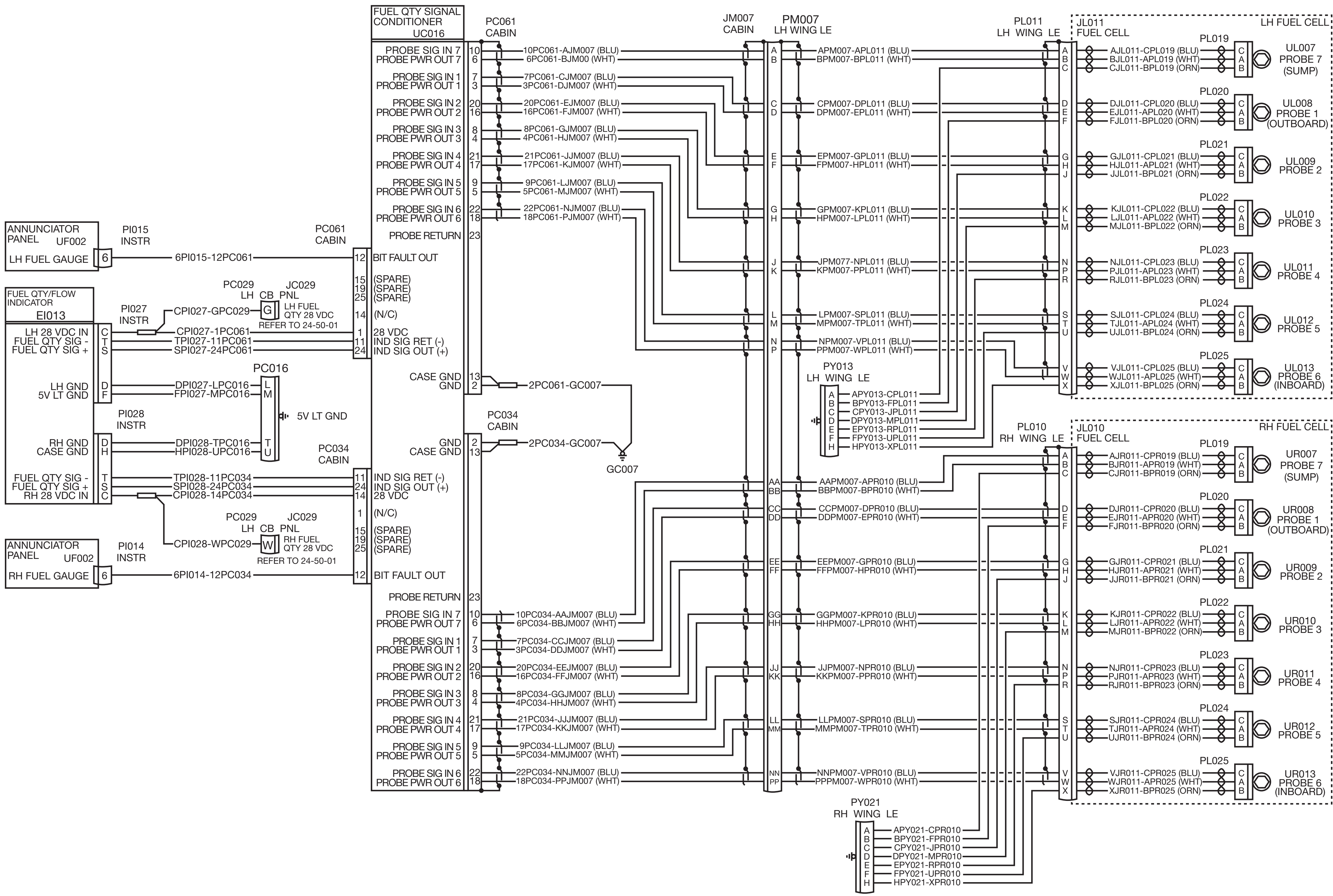


Figure 28-2. Fuel Quantity System

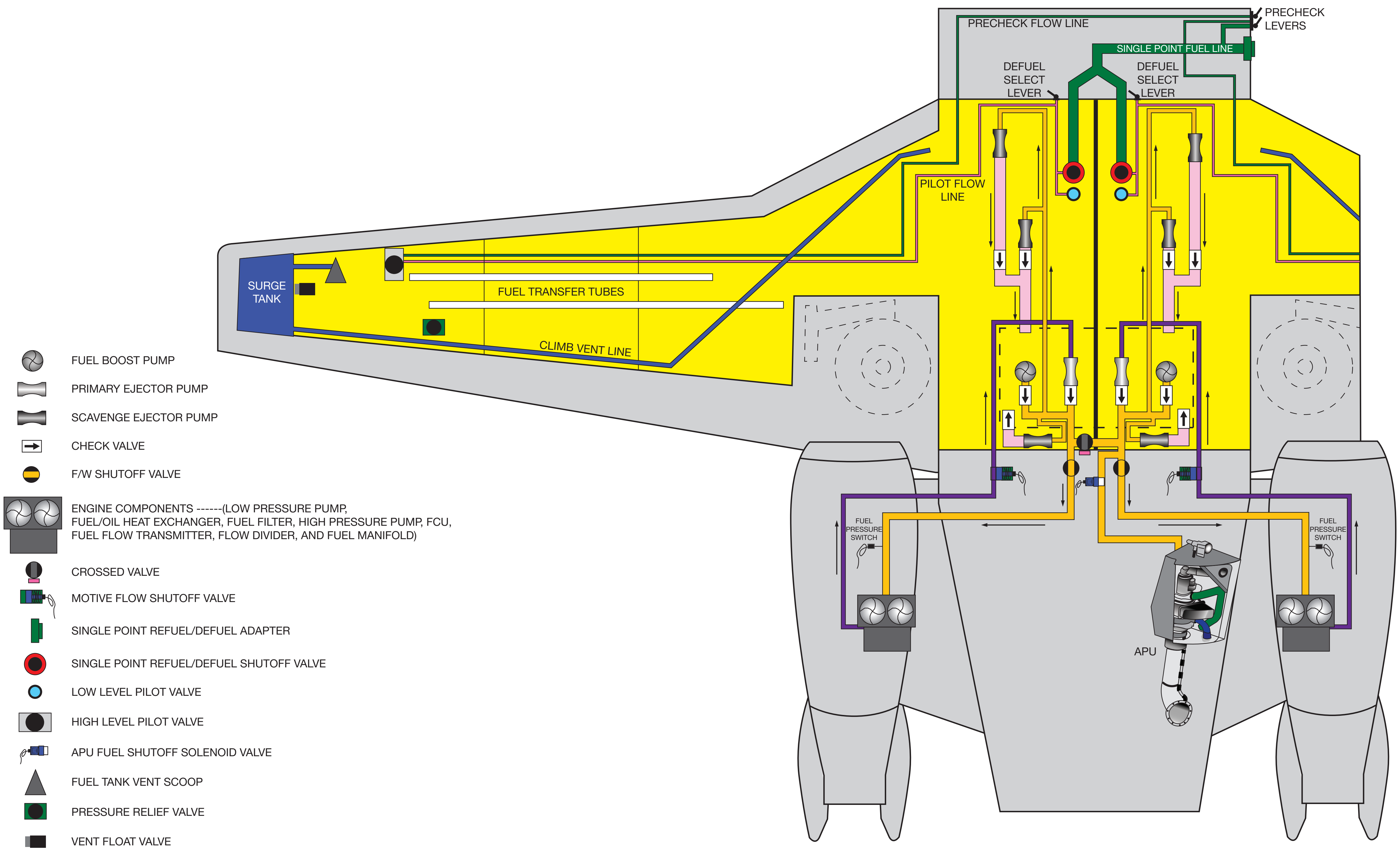


Figure 28-3. Fuel System Flow Schematic

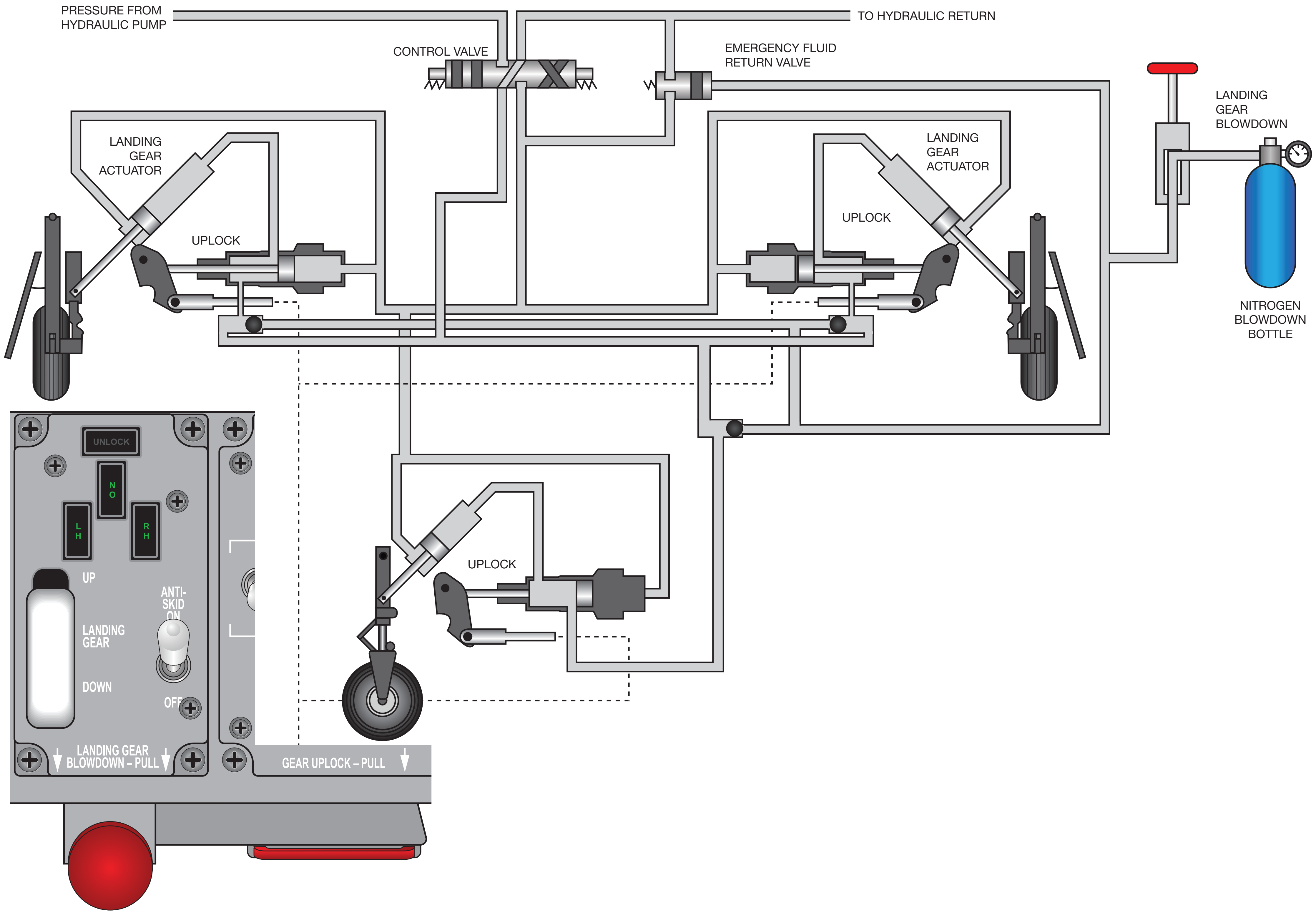


Figure 32-1. Landing Gear Hydraulic System

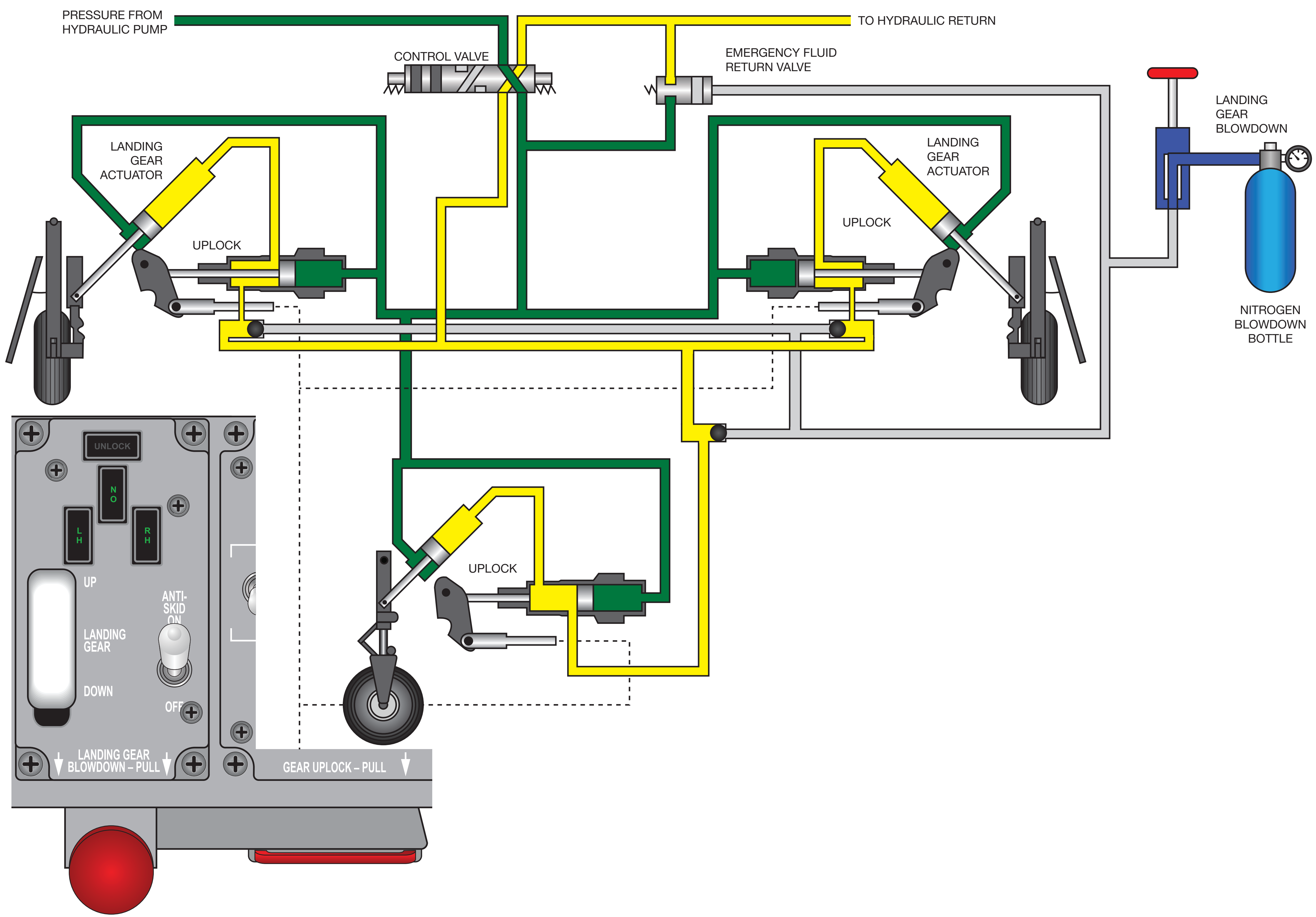


Figure 32-2. Landing Gear Retracting

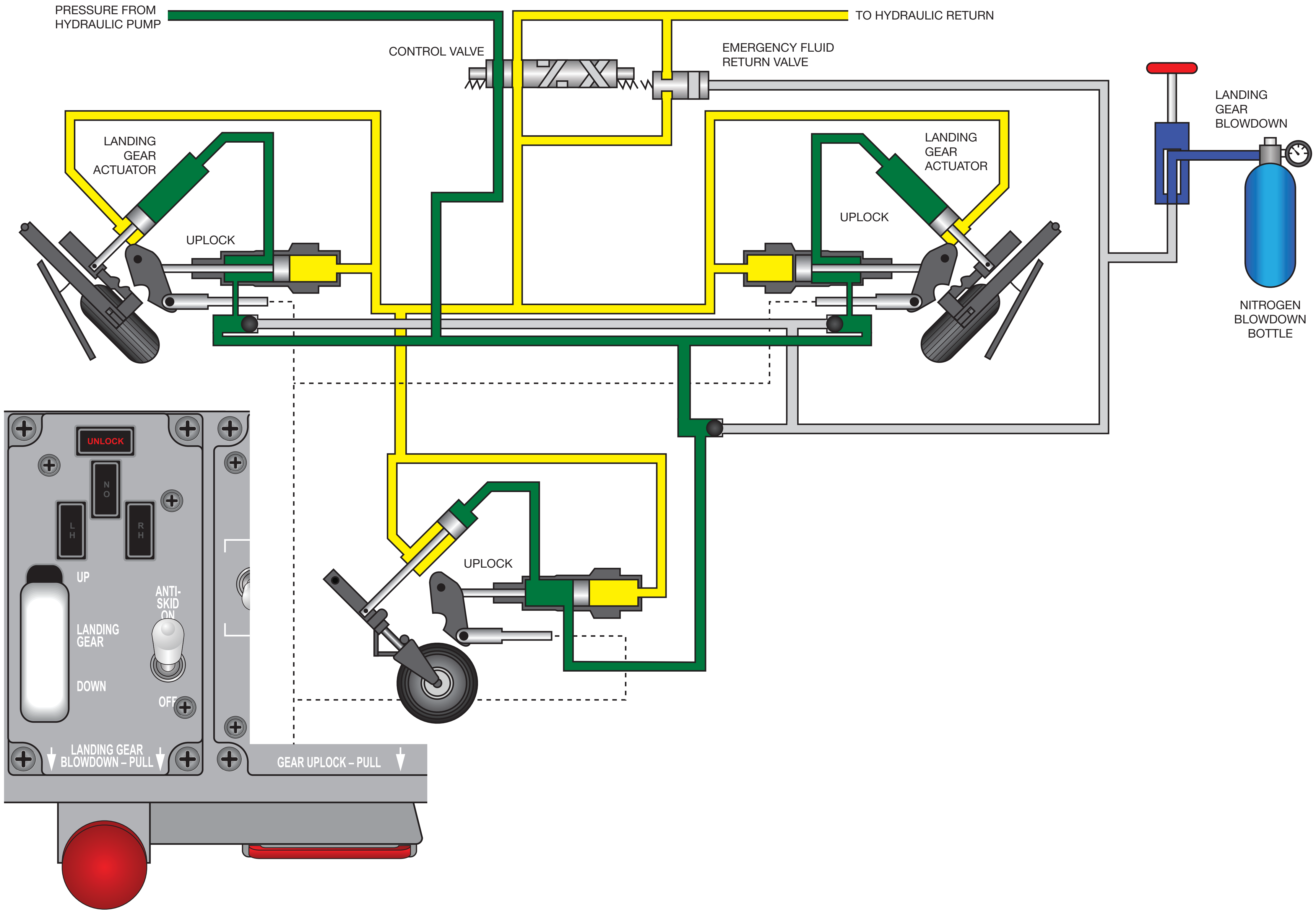


Figure 32-3. Landing Gear Extending

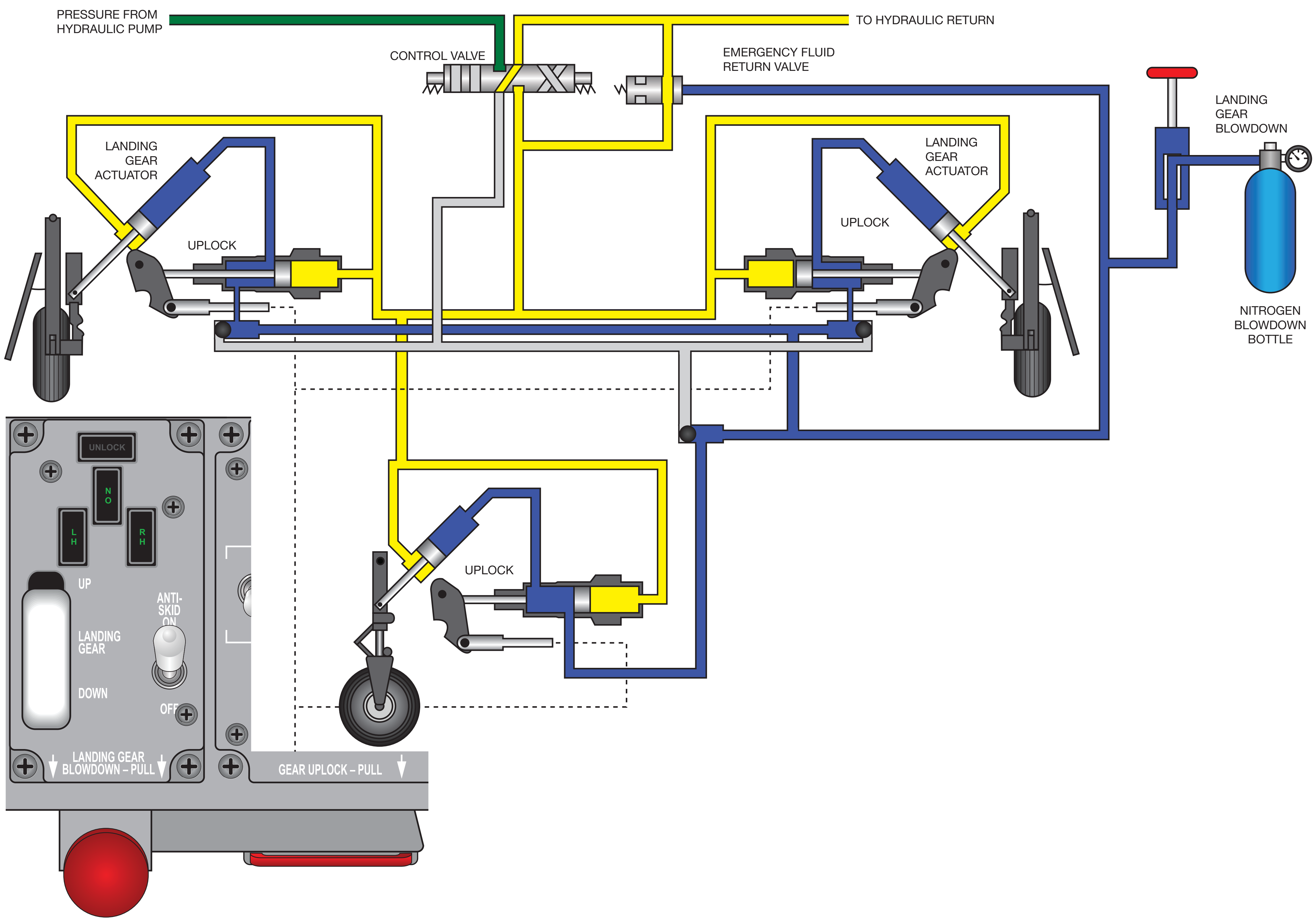


Figure 32-4. Landing Gear Emergency Extension

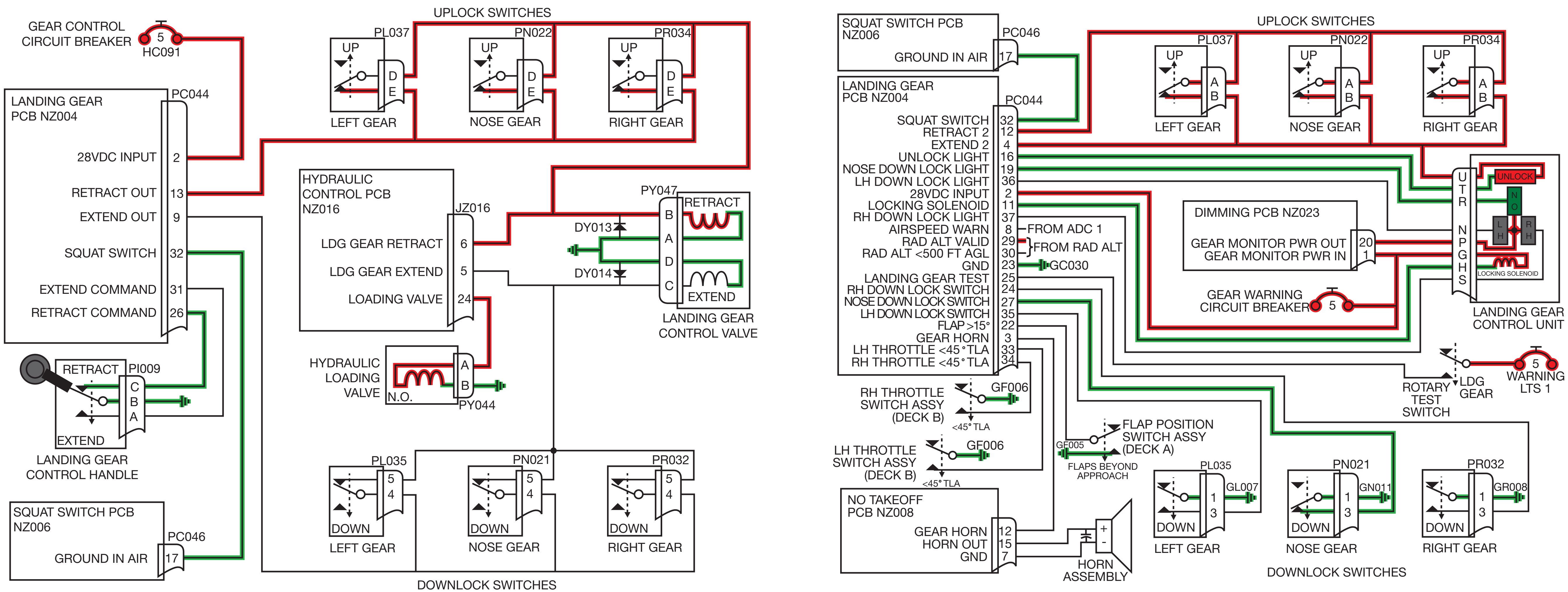


Figure 32-5. Landing Gear Control and Landing Gear Warning (Retracting)

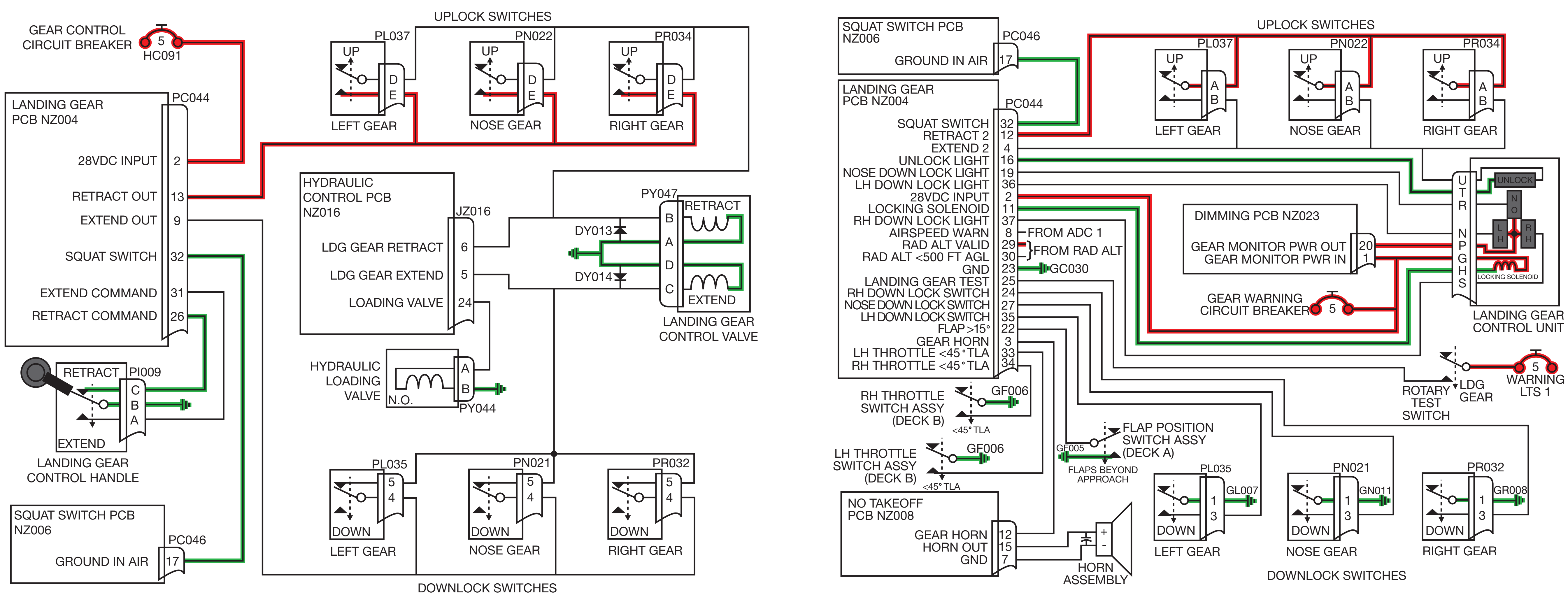


Figure 32-6. Landing Gear Control and Landing Gear Warning (Retracted)

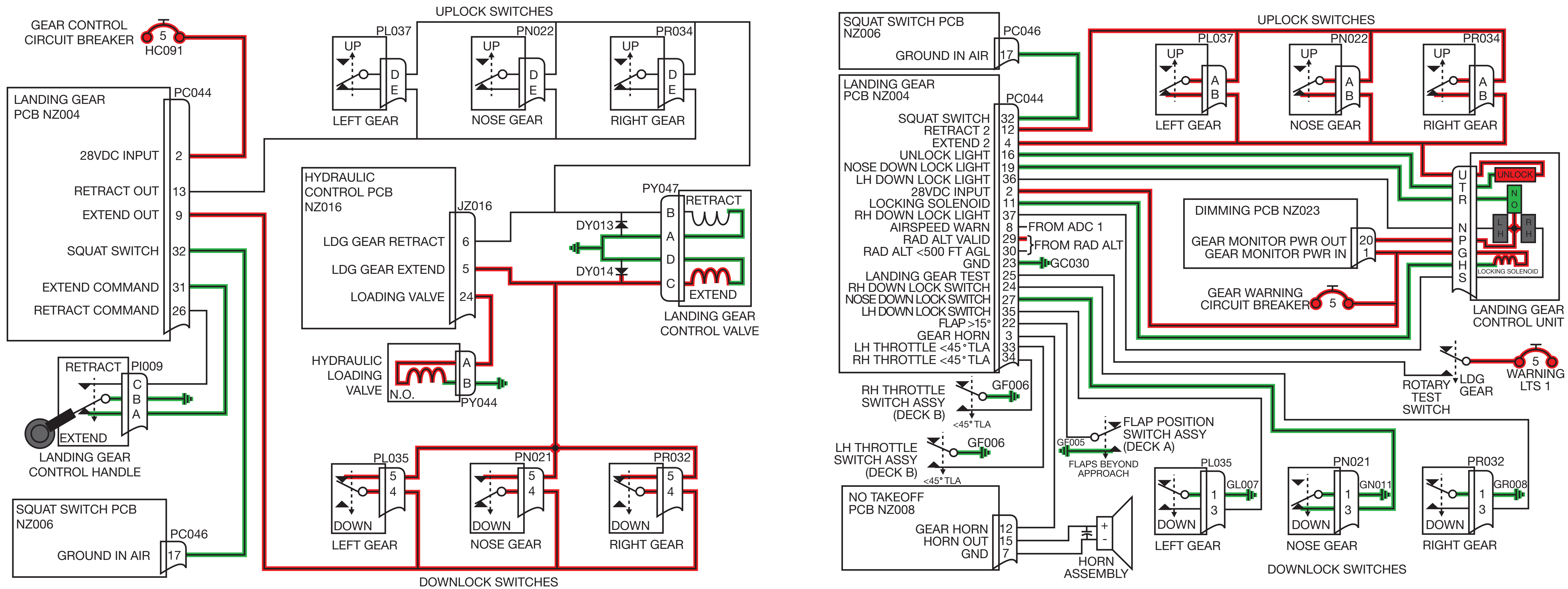


Figure 32-7. Landing Gear Control and Landing Gear Warning (Extending with Nose Gear Down and Locked)

32

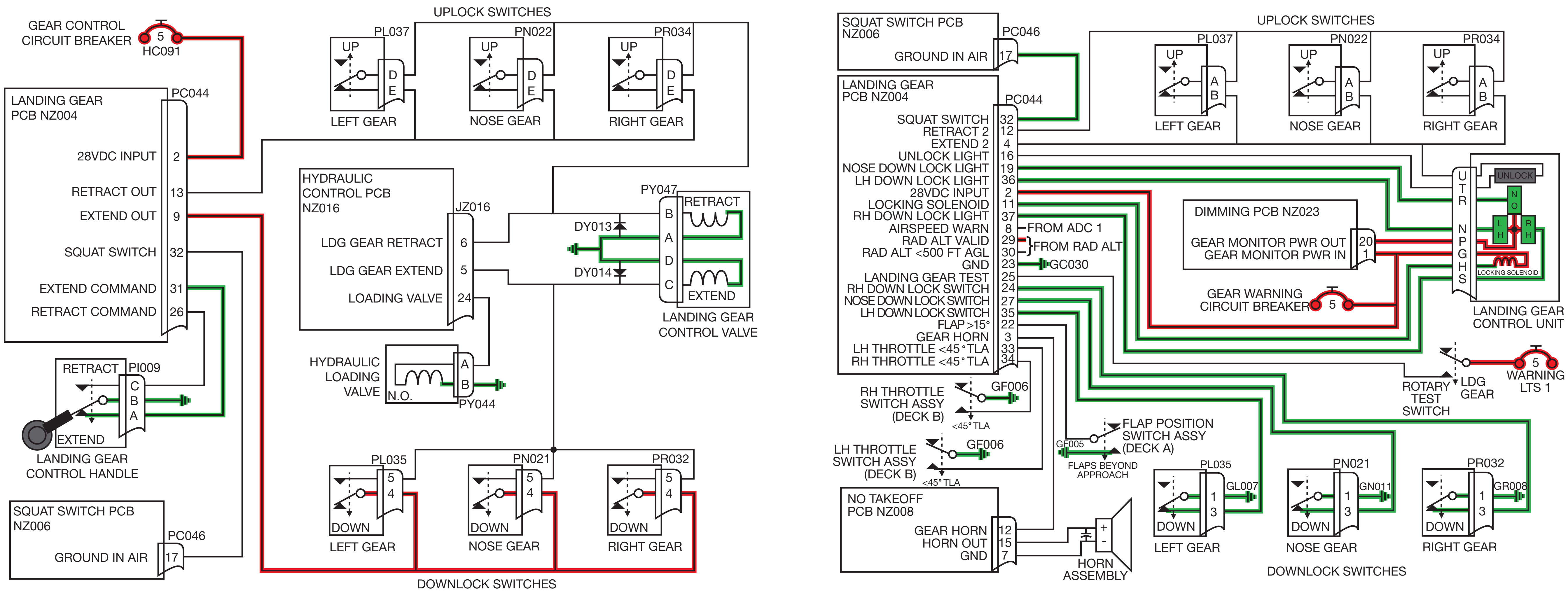
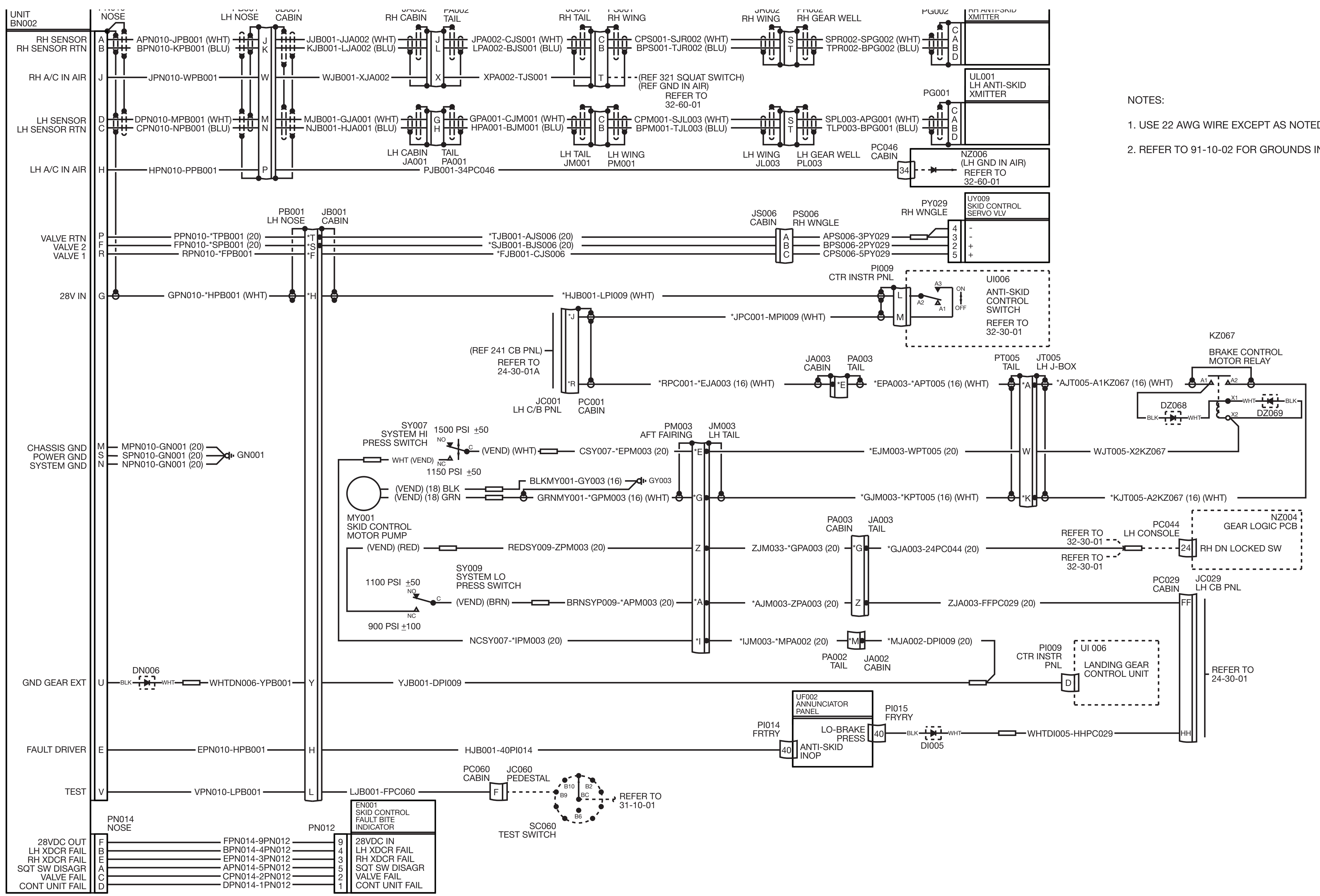


Figure 32-8. Landing Gear Control and Landing Gear Warning (Extended)

32

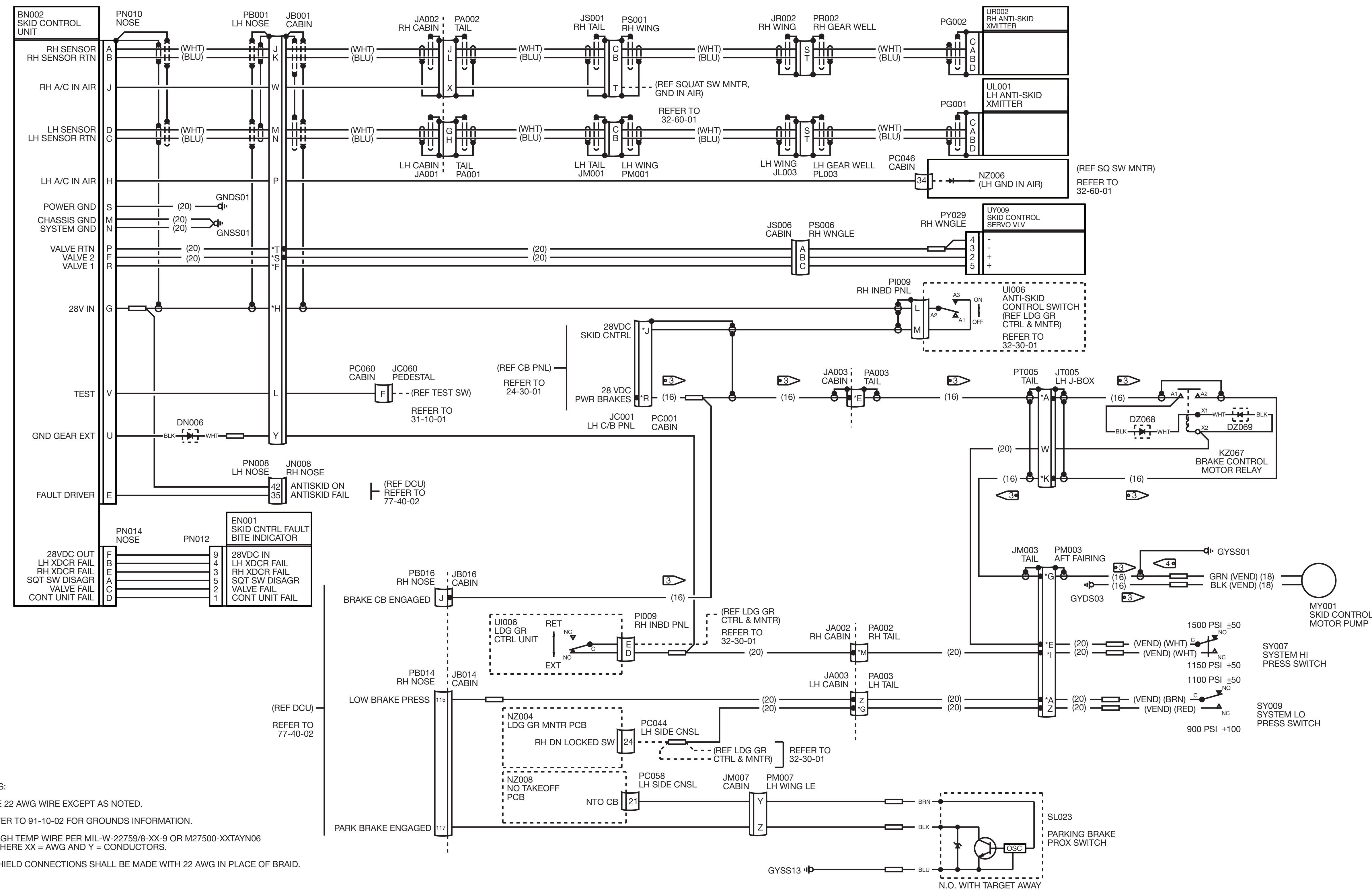


NOTES:

1. USE 22 AWG WIRE EXCEPT AS NOTED.

2. REFER TO 91-10-02 FOR GROUNDS INFORMATION.

Figure 32-9. Antiskid System - Units 5001-6000



NOTES:

1. USE 22 AWG WIRE EXCEPT AS NOTED.

2. REFER TO 91-10-02 FOR GROUNDS INFORMATION.

3 HIGH TEMP WIRE PER MIL-W-22759/8-XX-9 OR M27500-XXTAYN06 WHERE XX = AWG AND Y = CONDUCTORS.

4 SHIELD CONNECTIONS SHALL BE MADE WITH 22 AWG IN PLACE OF BRAID.

Figure 32-10. Antiskid System - Units 6001 and Subsequent

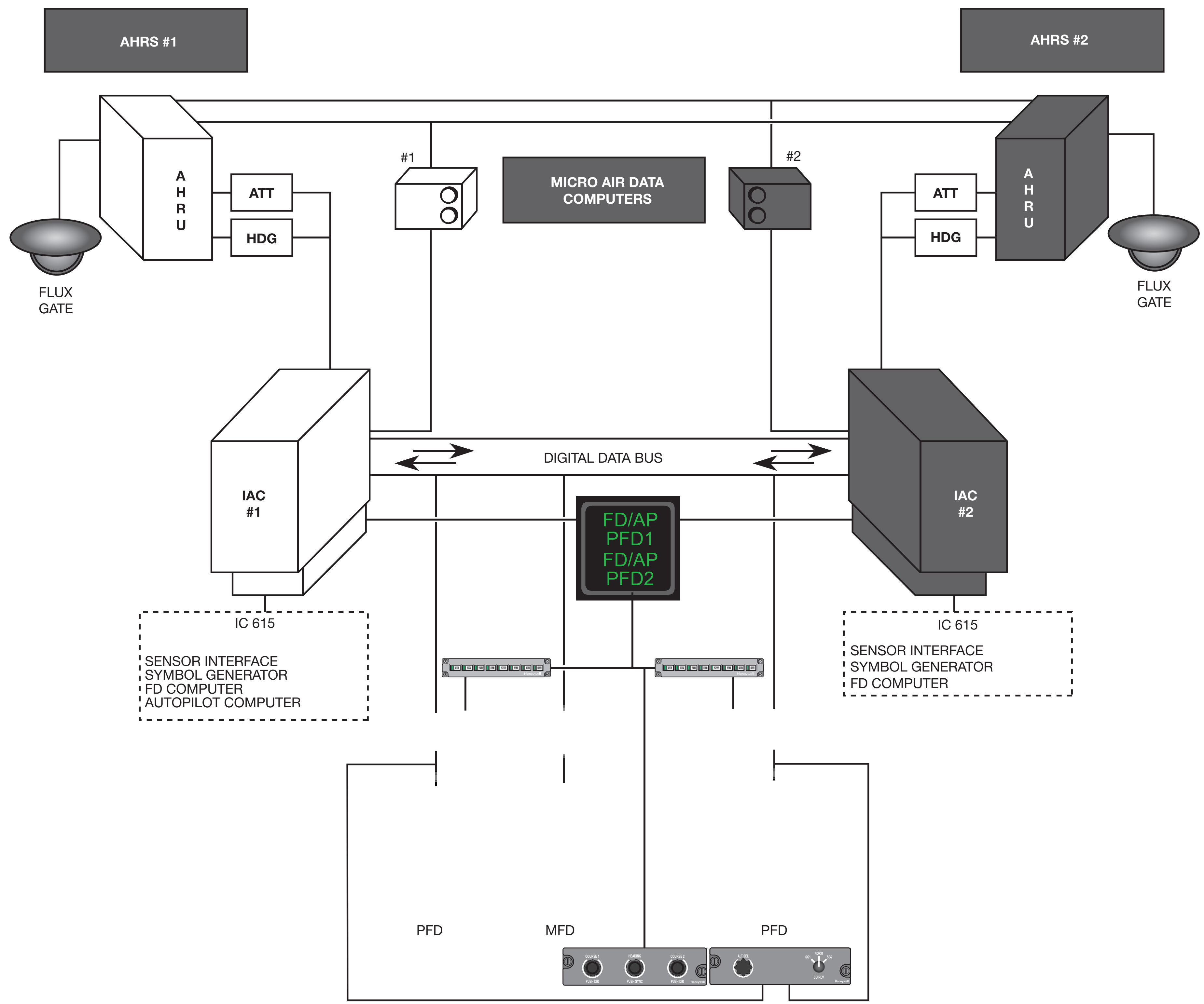


Figure 34-1. Navigation System (XLS)

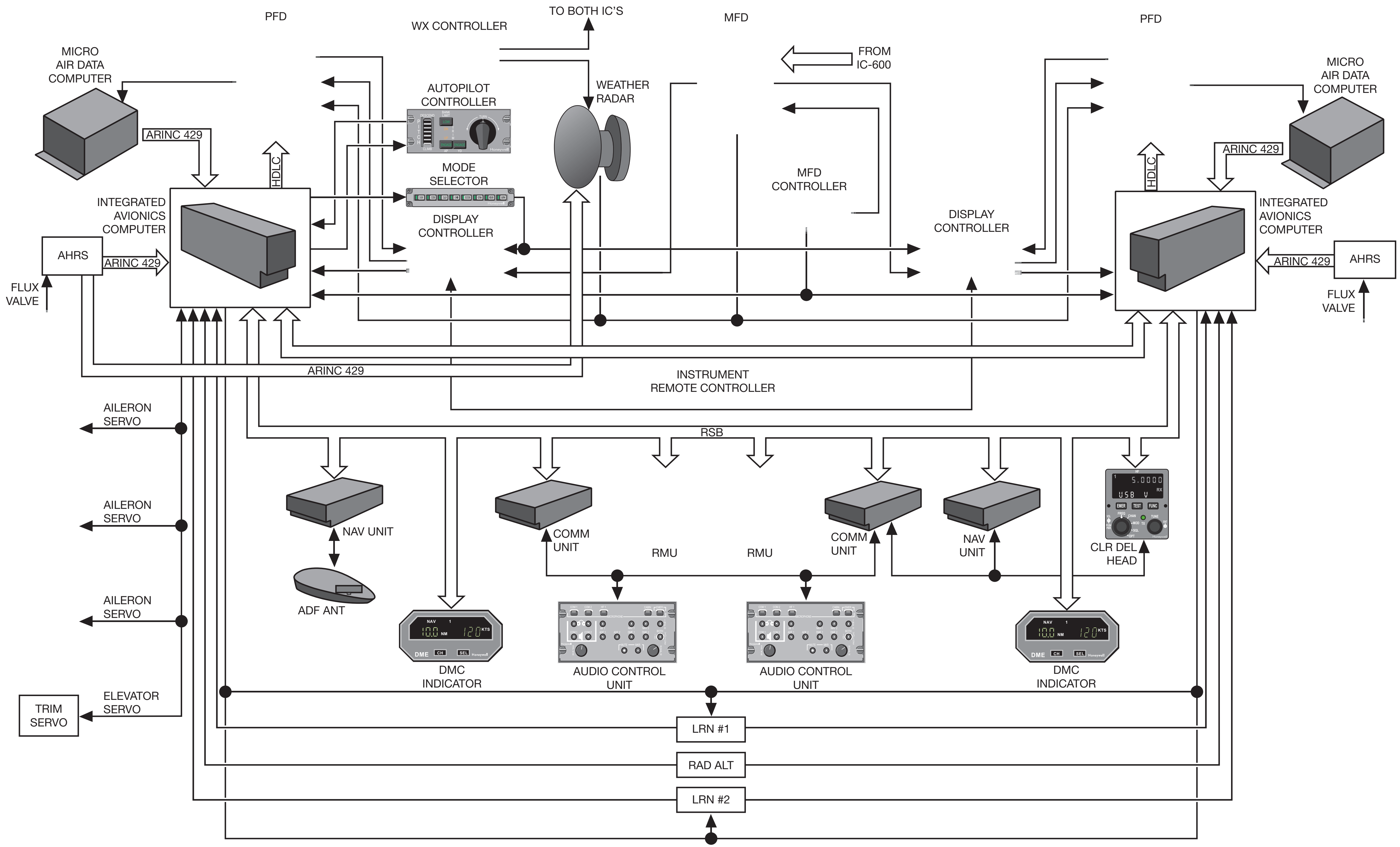


Figure 34-2. P1000 Avionics Diagram (XLS)

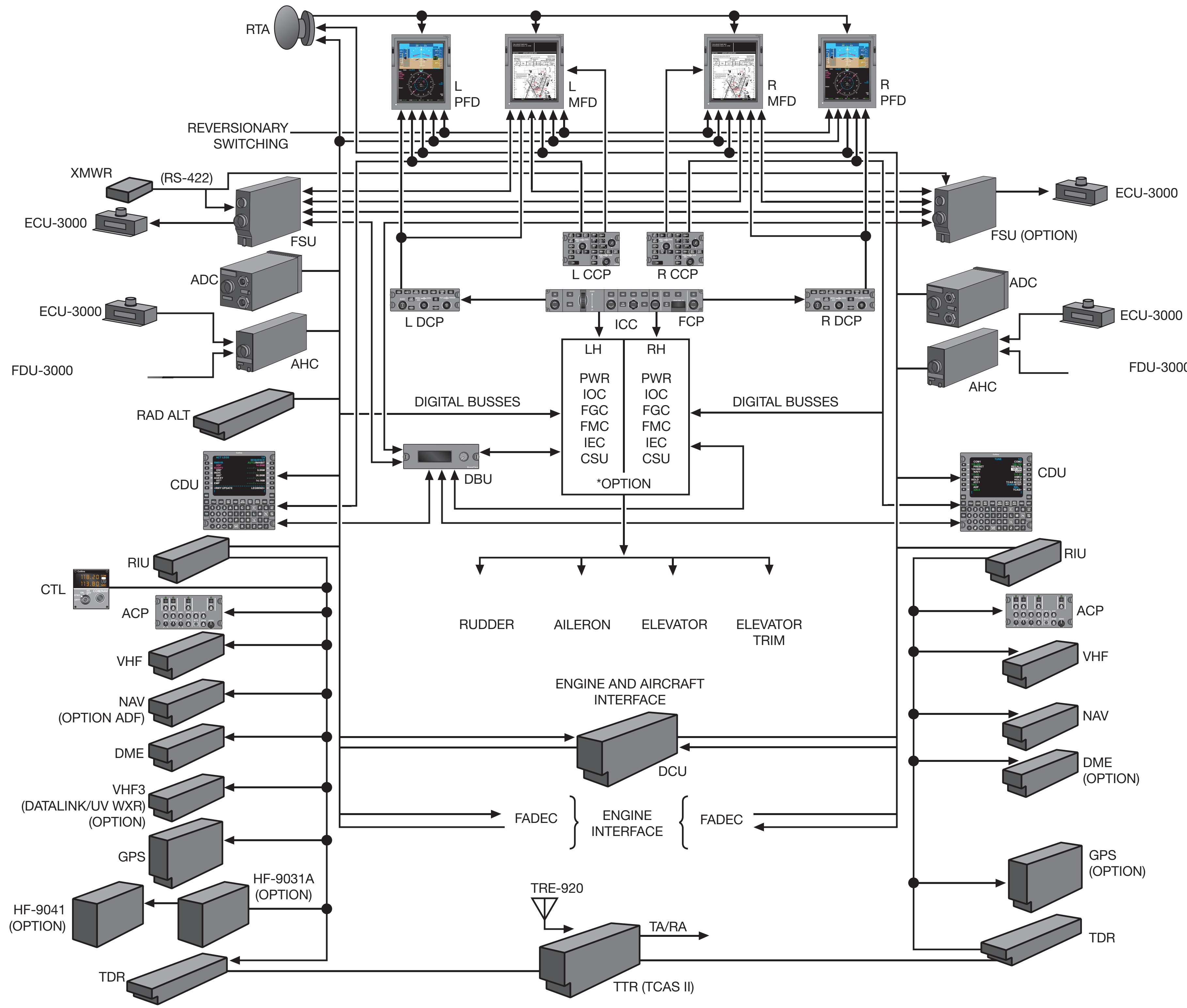


Figure 34-3. Proline 21 Avionics System Diagram (XLS+)

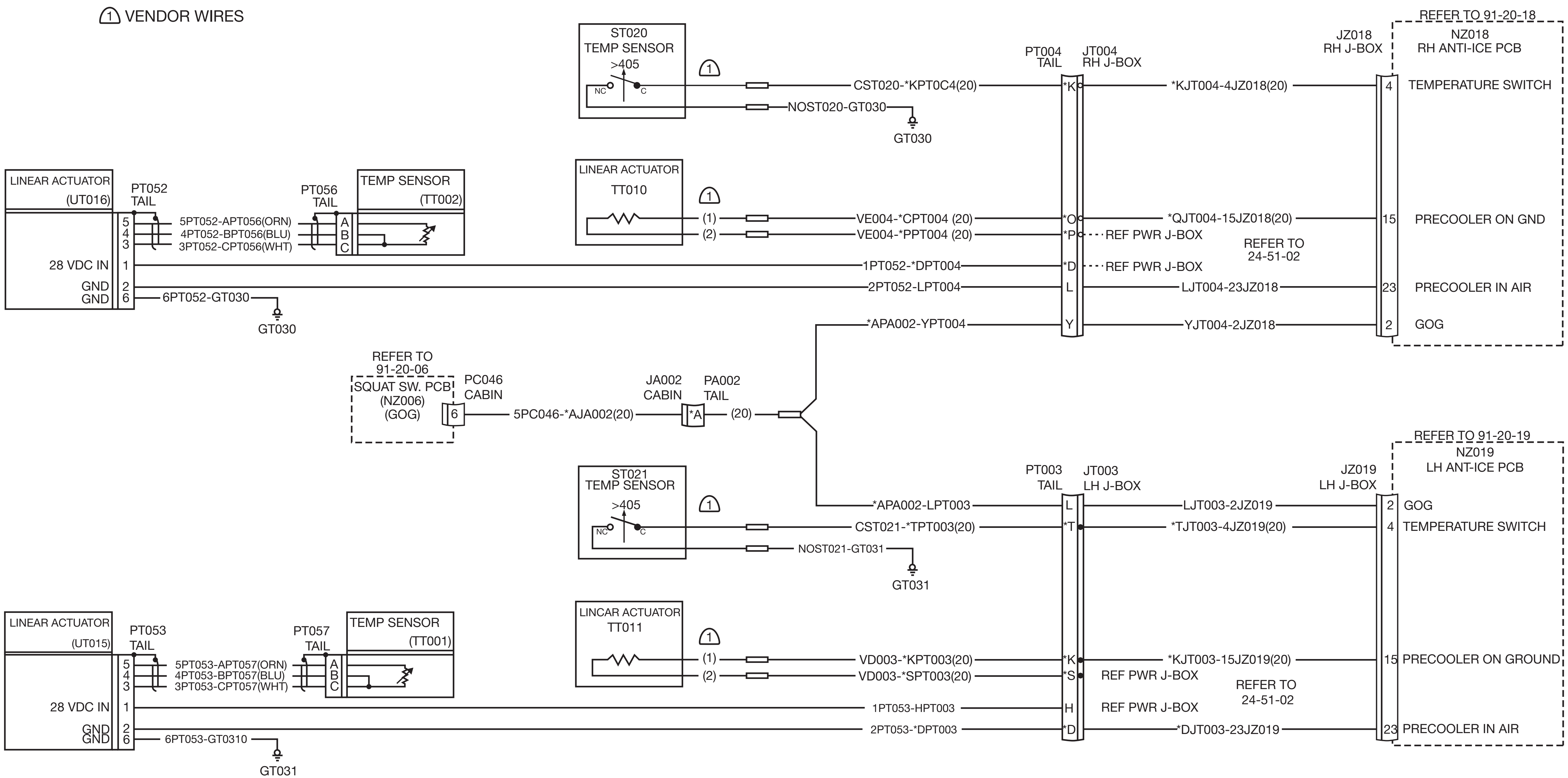
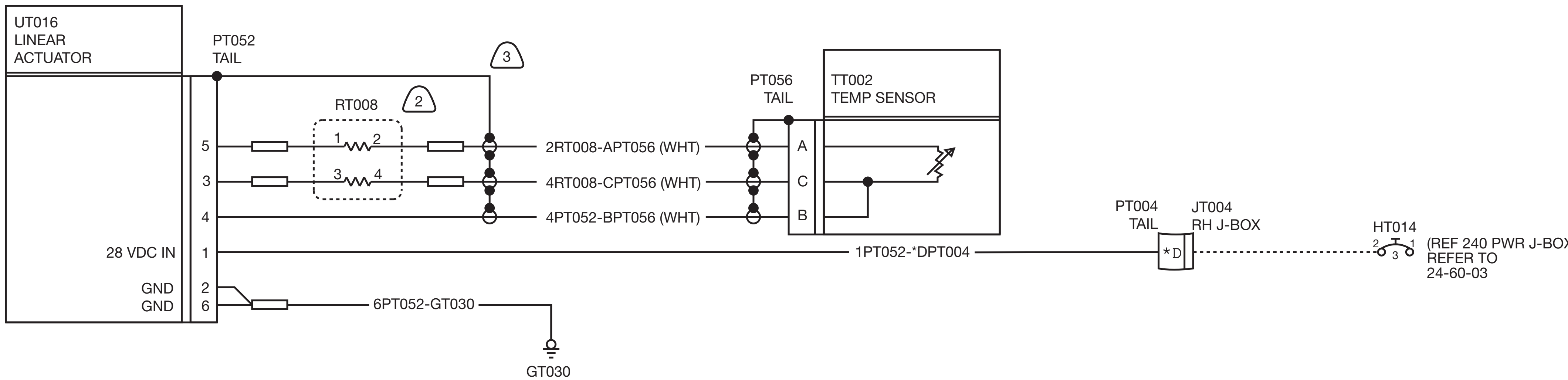


Figure 36-1. Bleed Air Precooler - Units 5001-5500

36



NOTES:

1. REFER TO 91-10-02 FOR GROUND INFORMATION.
2. THE DIFFERENCE IN RESISTANCE BETWEEN THE TWO RESISTORS IN THE ASSEMBLY SHALL BE LESS THAN 100 OHMS.
3. SHIELD TERMINATIONS SHALL BE MADE WITH 22 AWG WIRES IN PLACE OF BRAID.
4. USE 22 AWG WIRE EXCEPT AS NOTED.

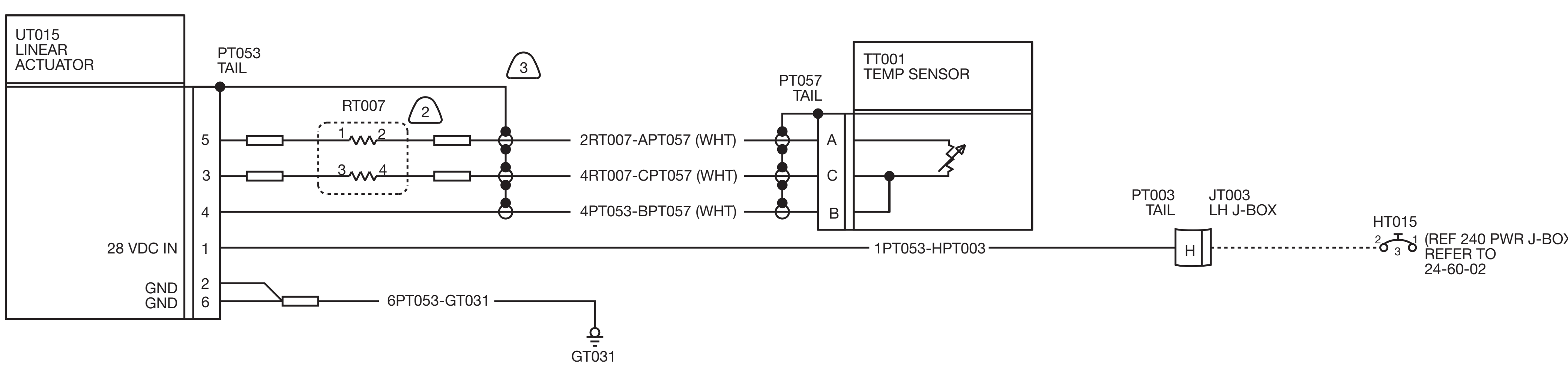
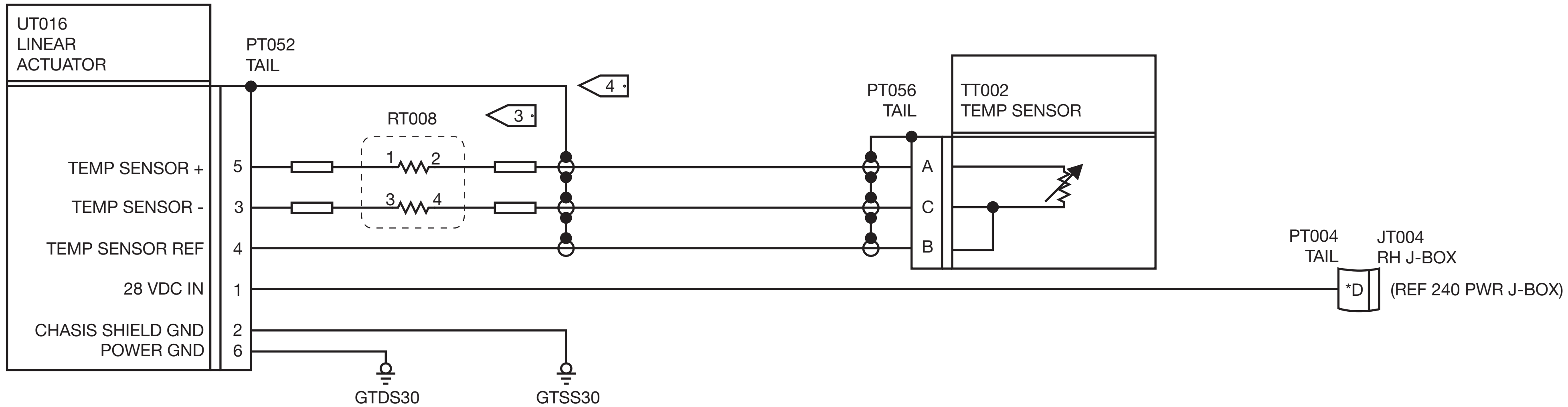


Figure 36-2. Bleed Air Precooler - Units 5501-5616, 5618-6000



NOTES:

3 IN THE ASSEMBLY SHALL BE LESS THAN 100 OHMS.

4 SHIELD TERMINATIONS SHALL BE WITH 22 AWG WIRES IN PLACE OF BRADED

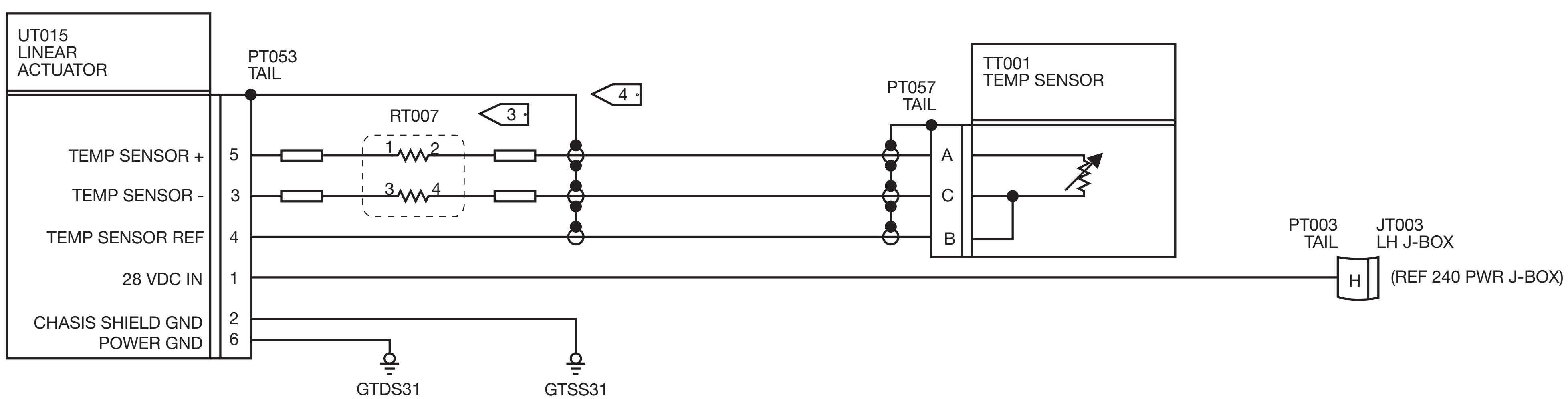


Figure 36-3. Bleed Air Precooler - Units 5617, 6001 and Subsequent

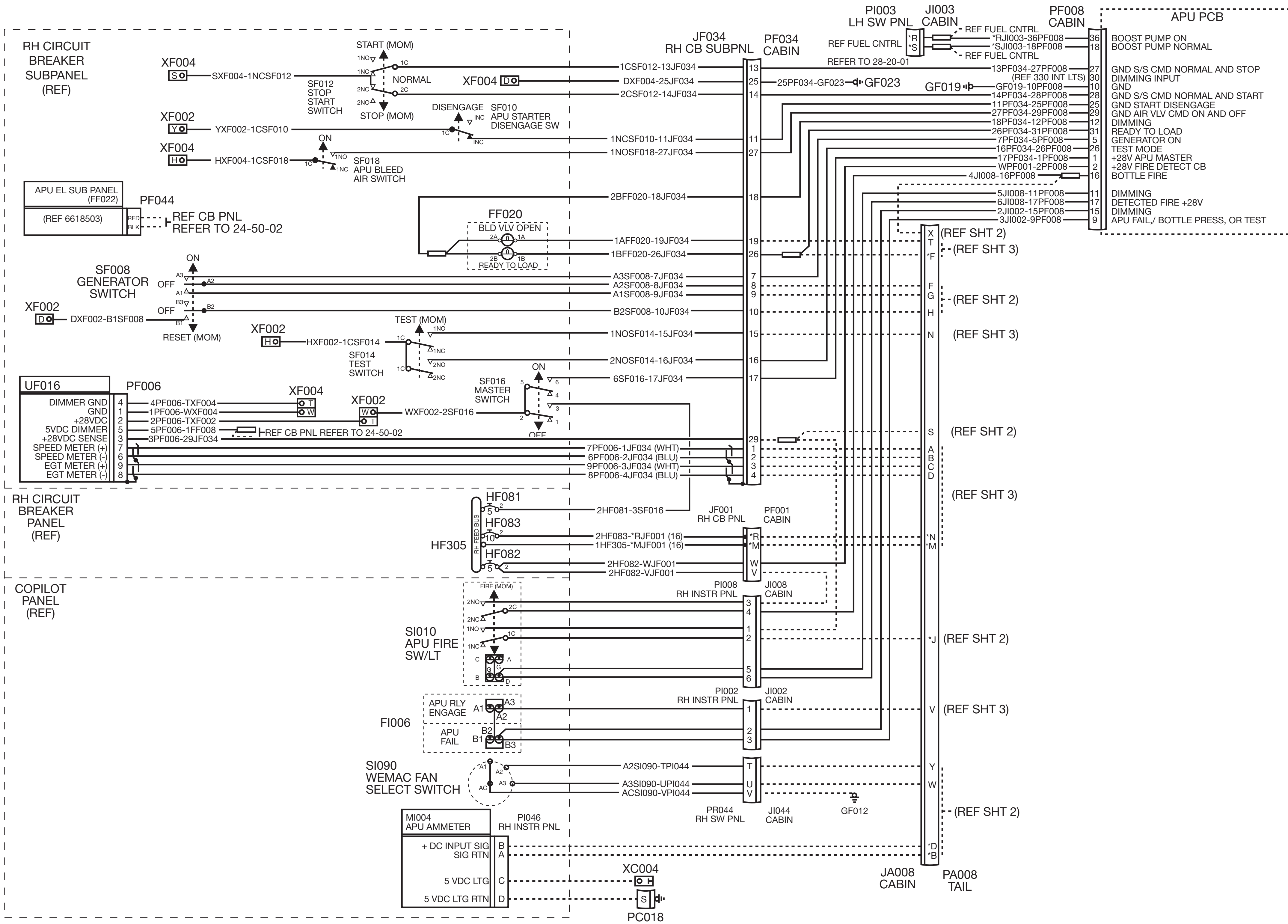


Figure 49-1. APU Installation (Sheet 1 of 3)

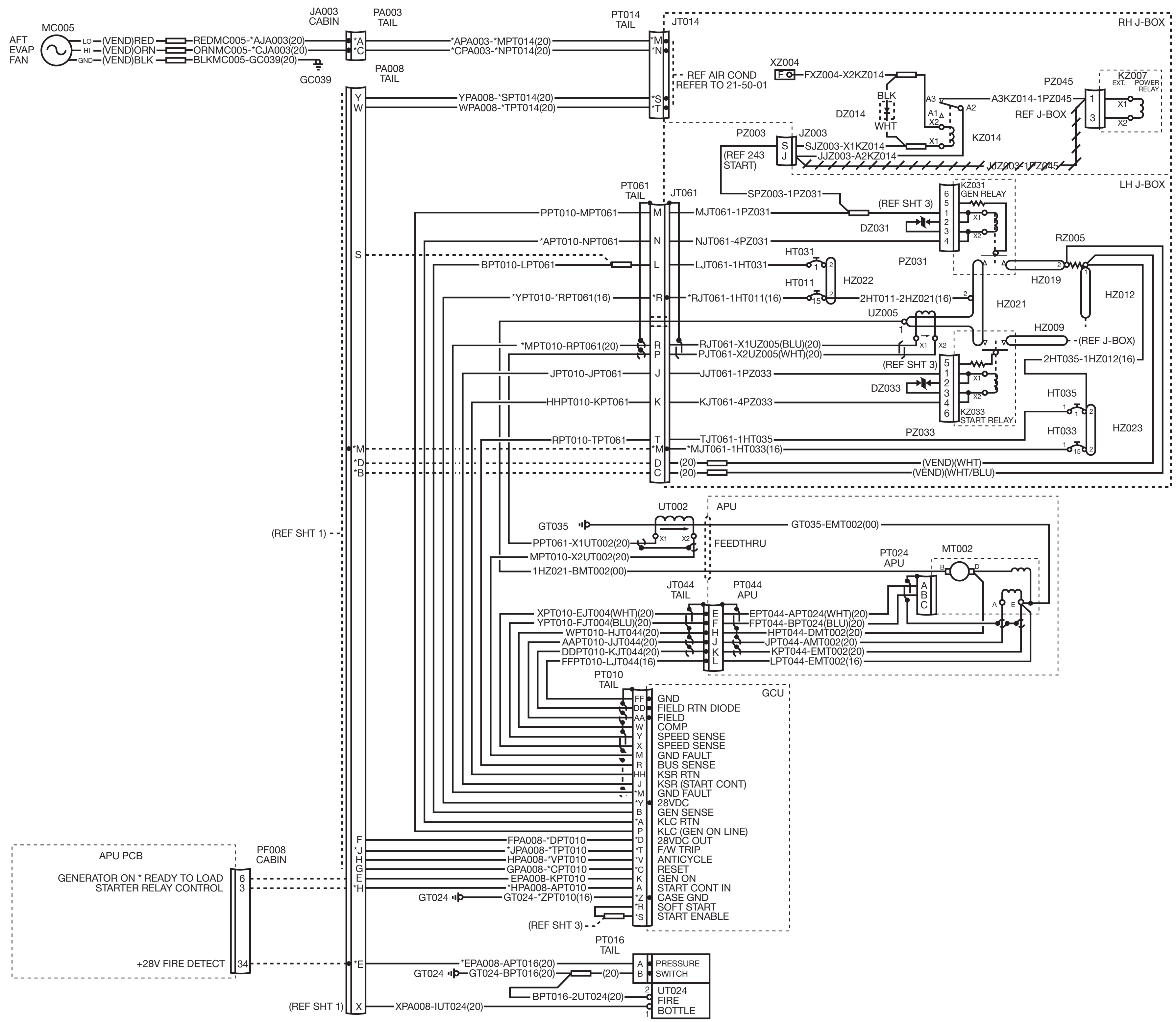


Figure 49-1. APU Installation (Sheet 2 of 3)

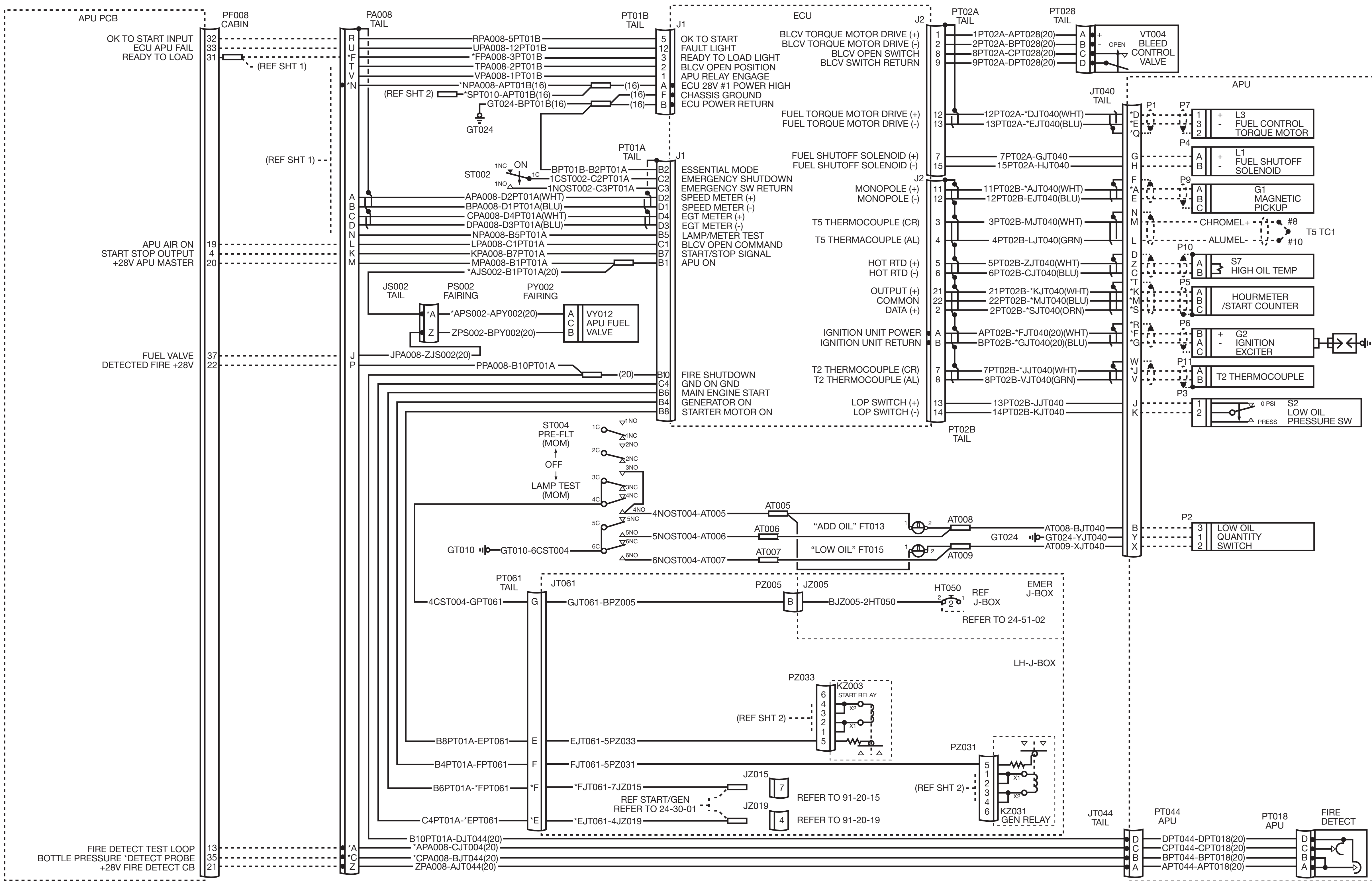


Figure 49-1. APU Installation (Sheet 3 of 3)

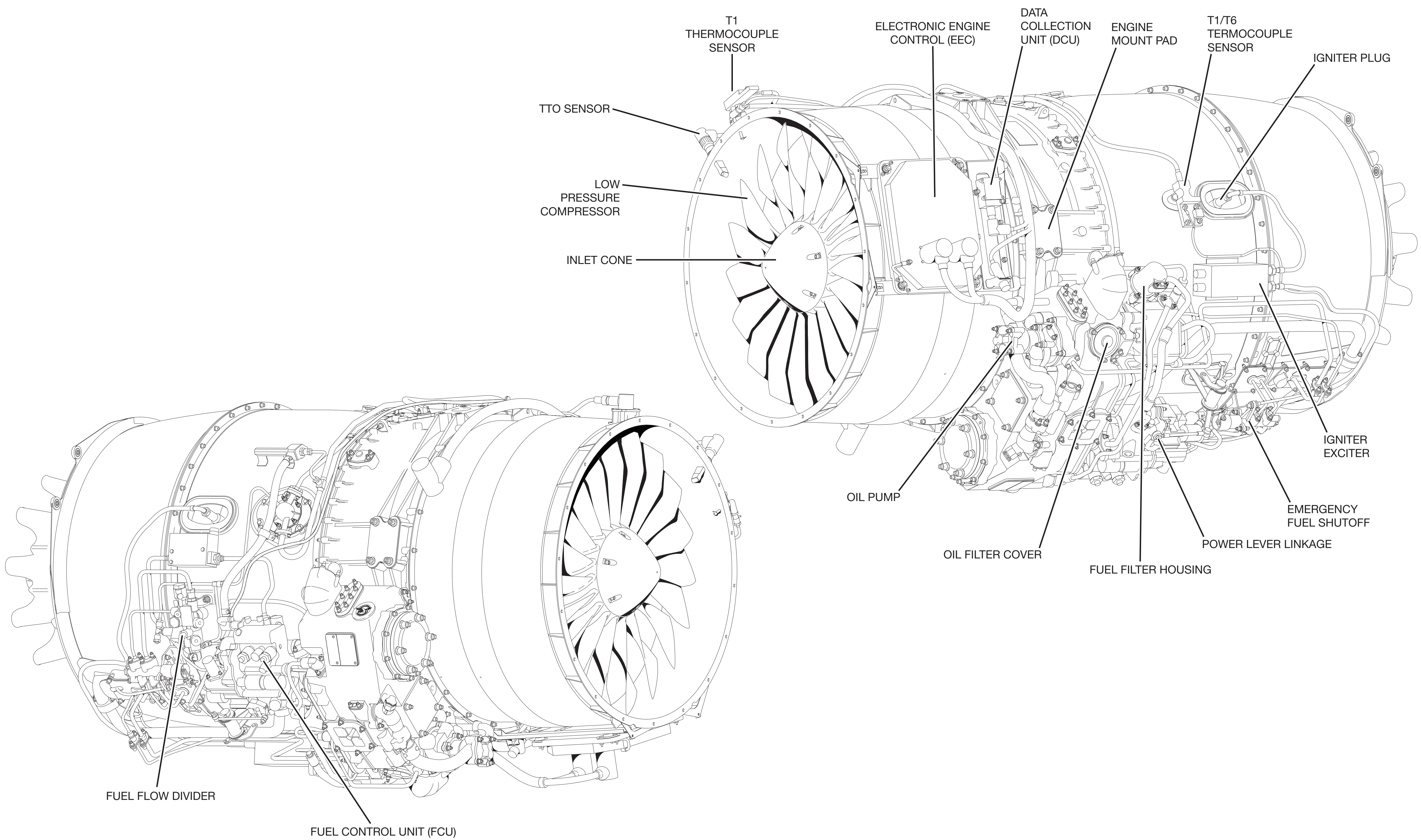


Figure 71-1. PW545 A/B Engine

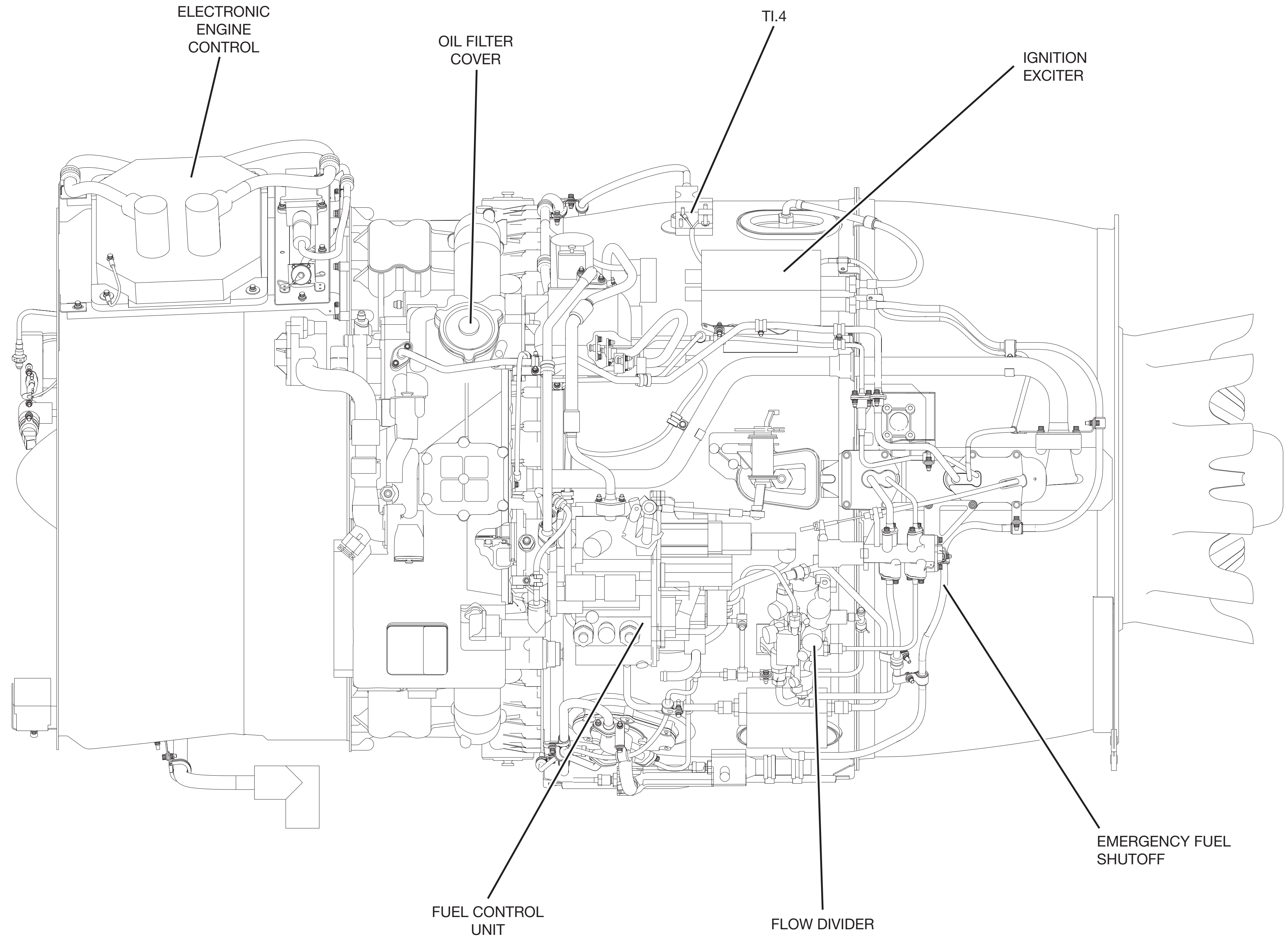


Figure 71-2. PW545 A/B Engine (Bottom View)

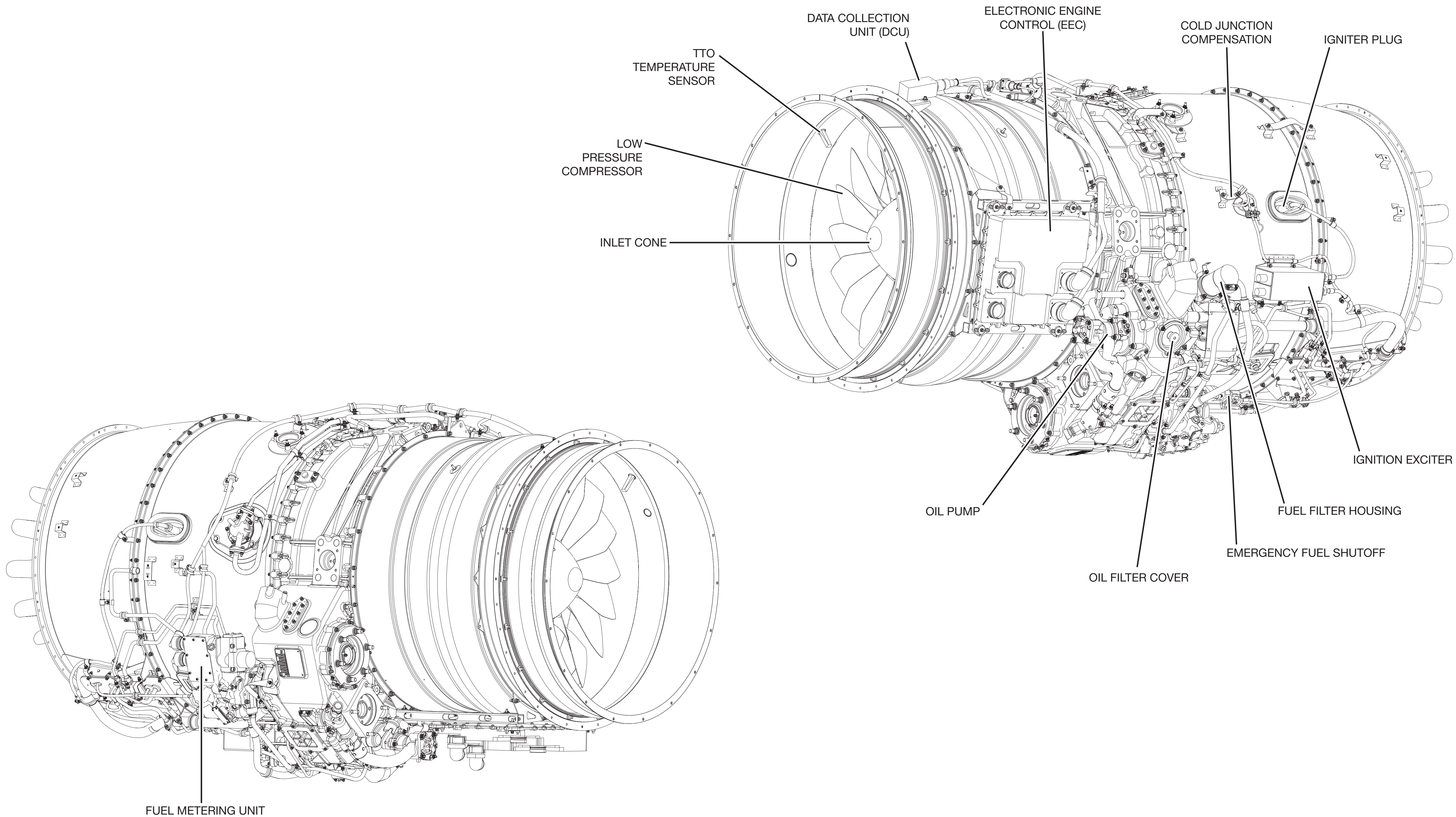


Figure 71-3. PW545C Engine

71-80

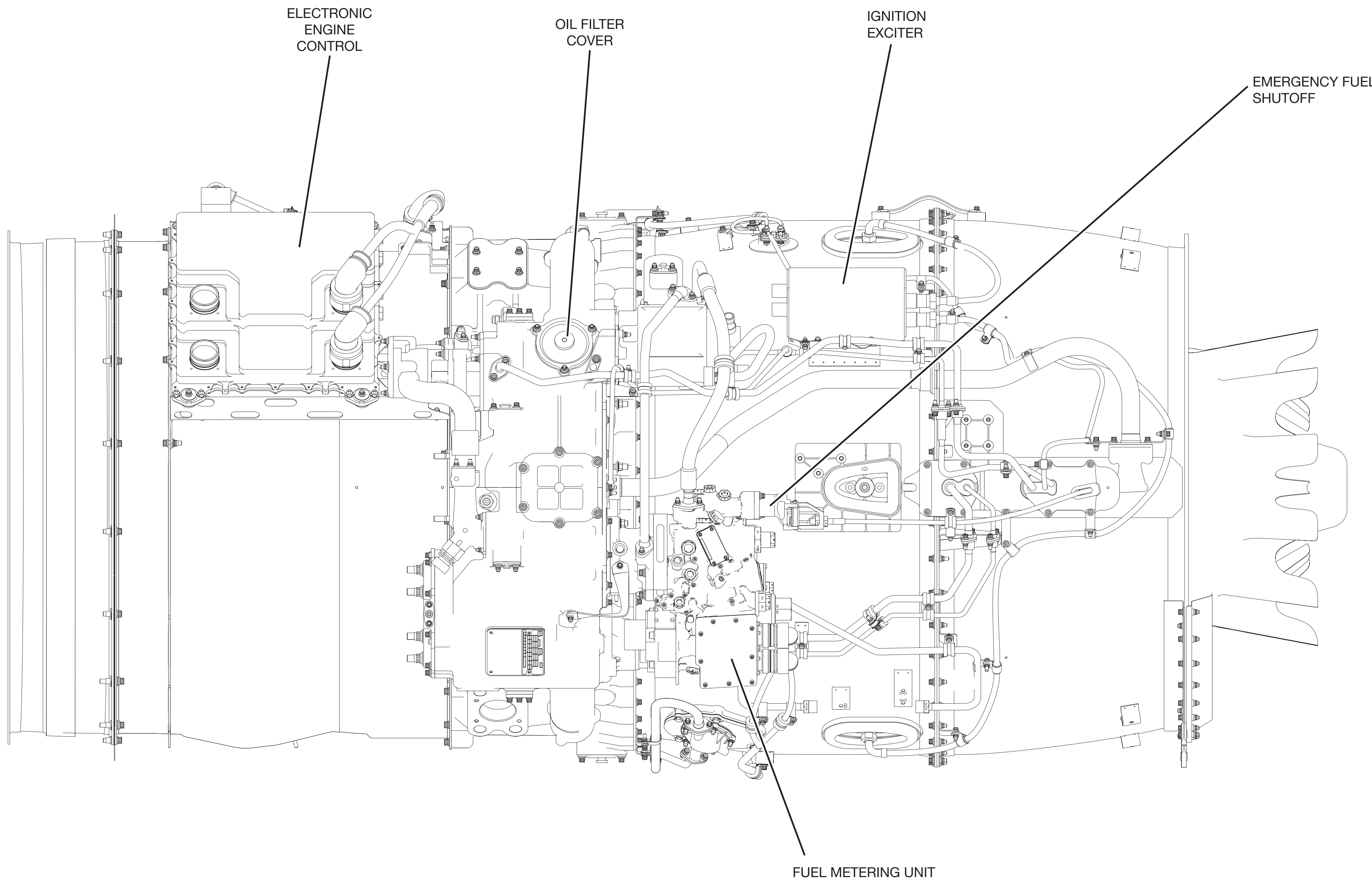


Figure 71-4. PW545C Engine

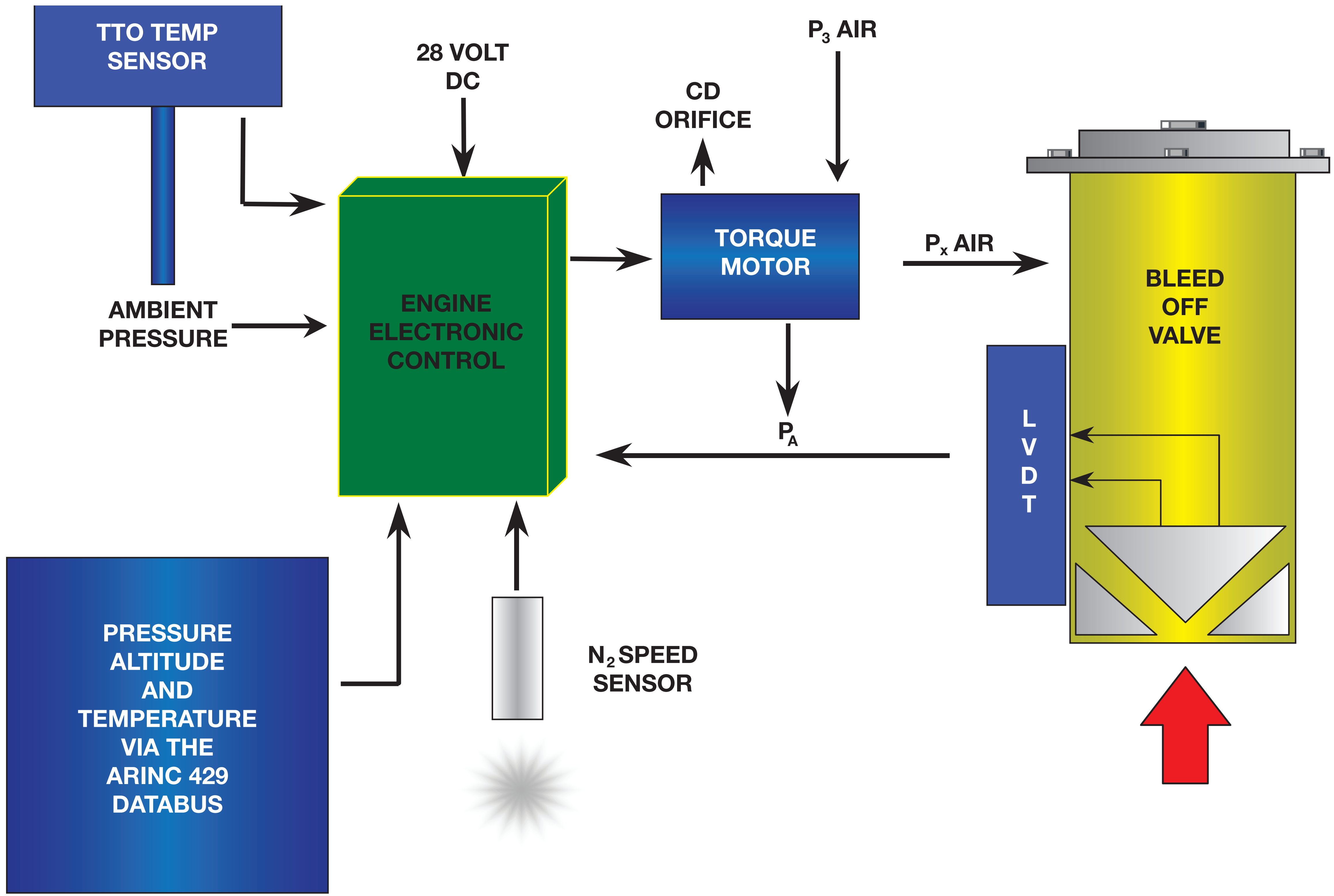


Figure 71-5. Bleed Valve Control Schematic

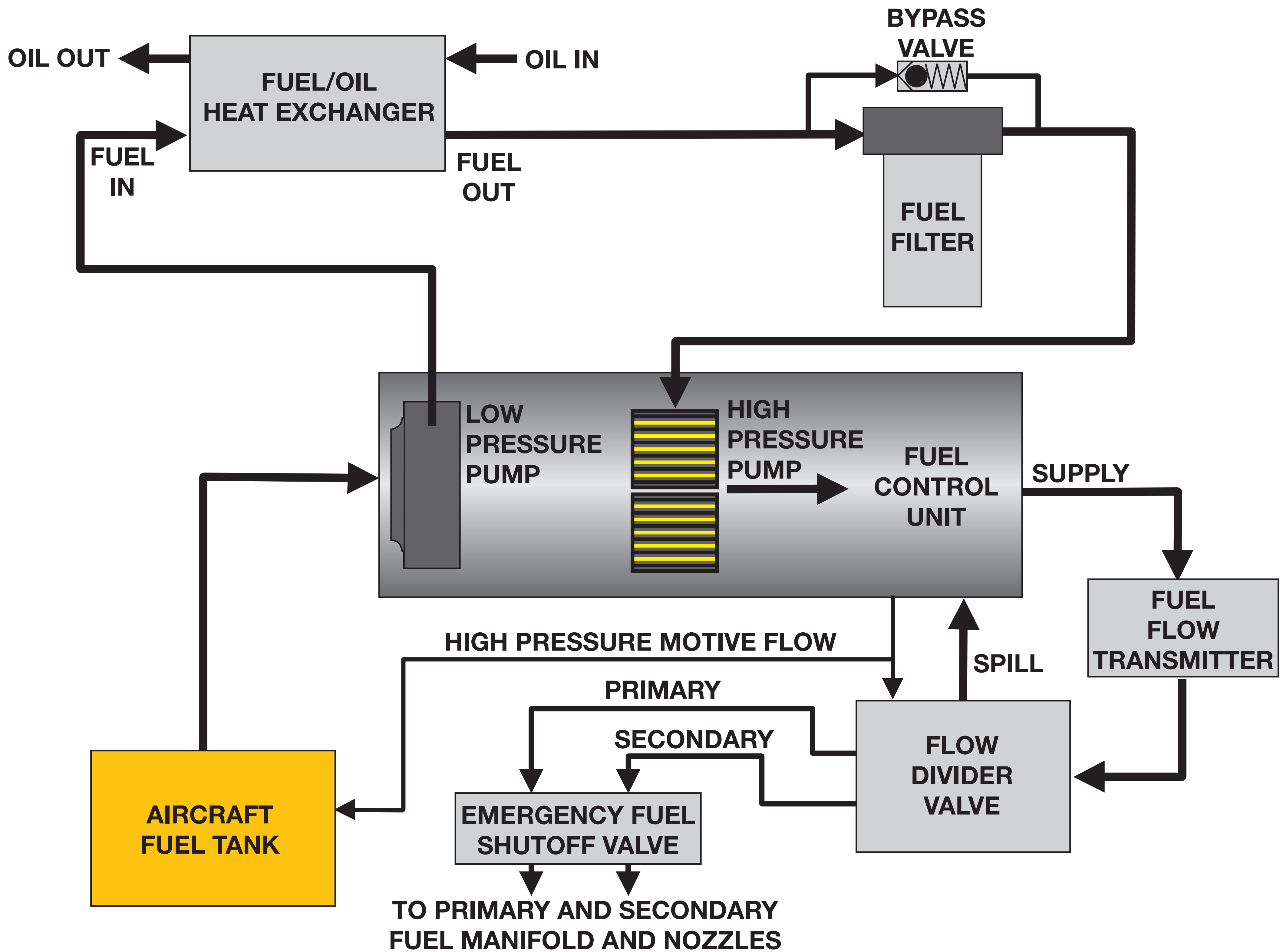


Figure 71-6. Engine Fuel System (545A/B)

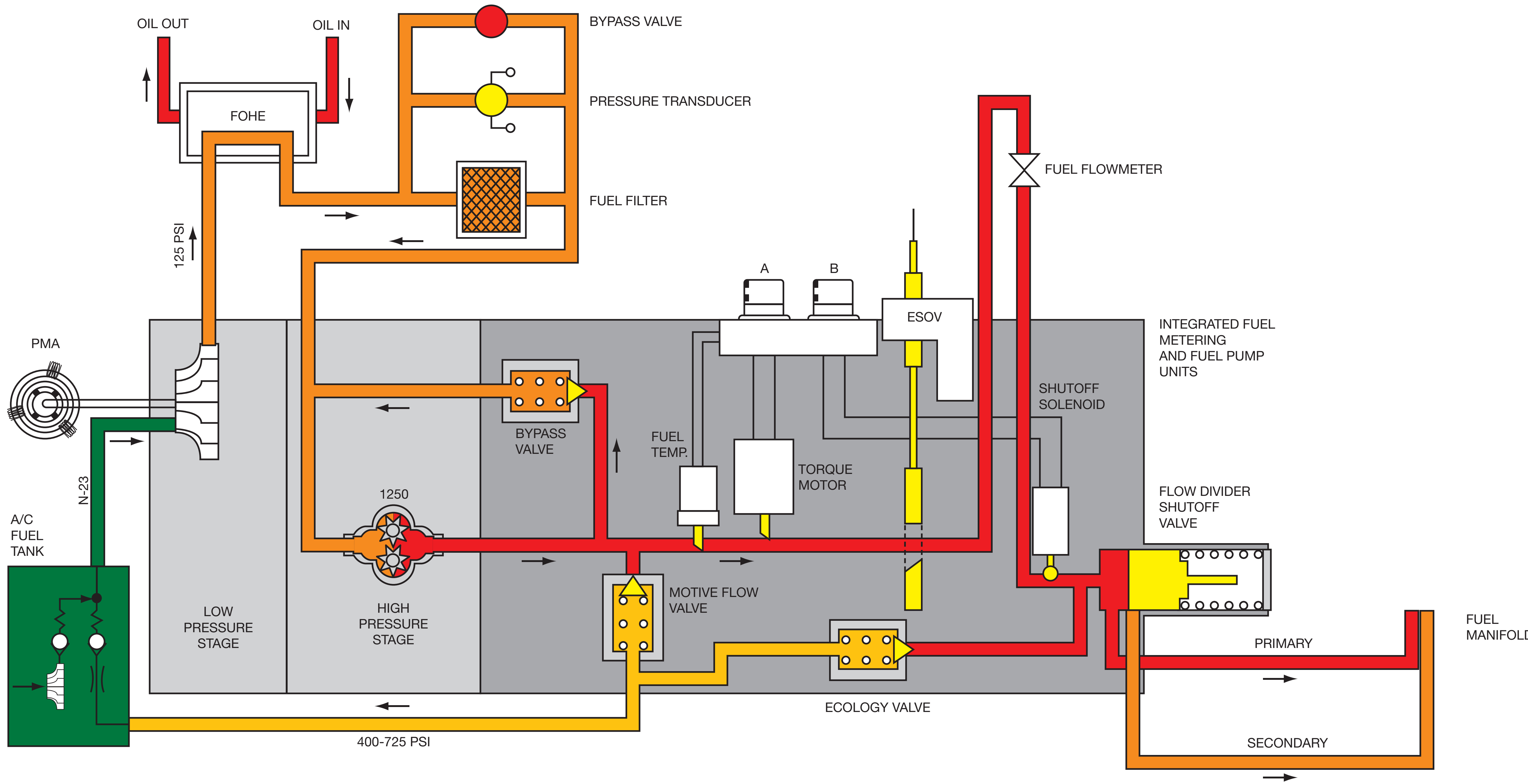
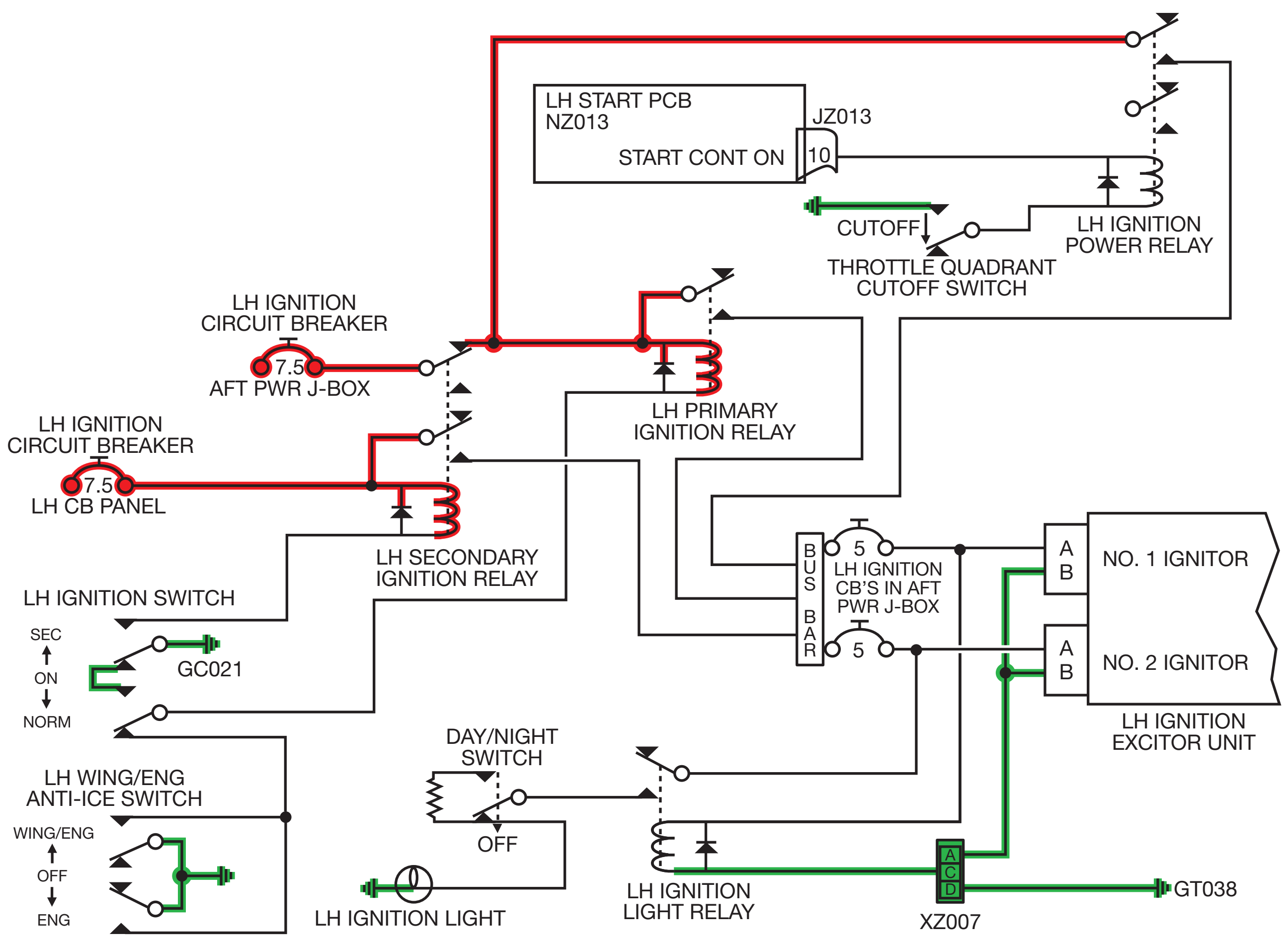
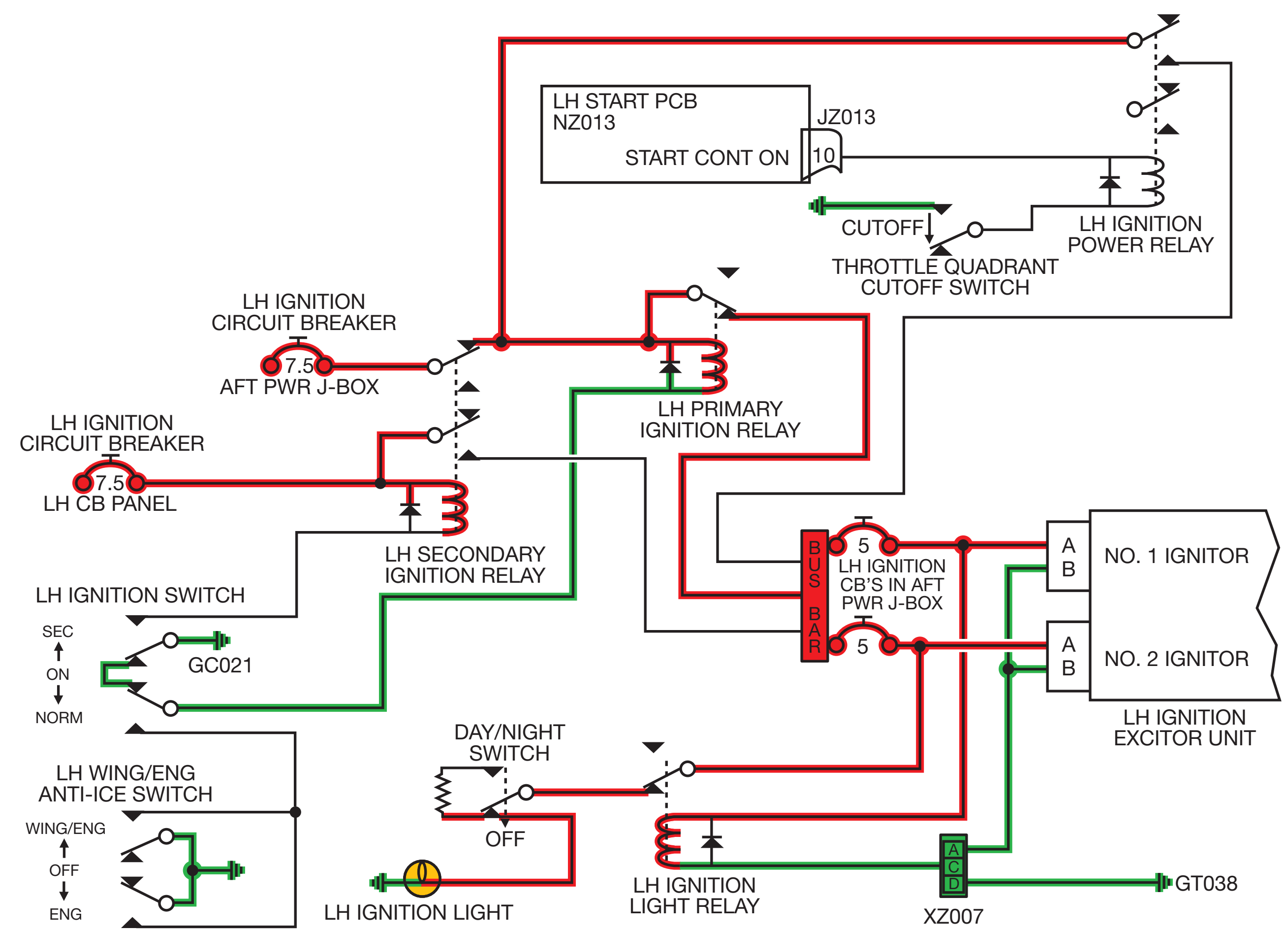


Figure 71-7. Engine Fuel System (545C)



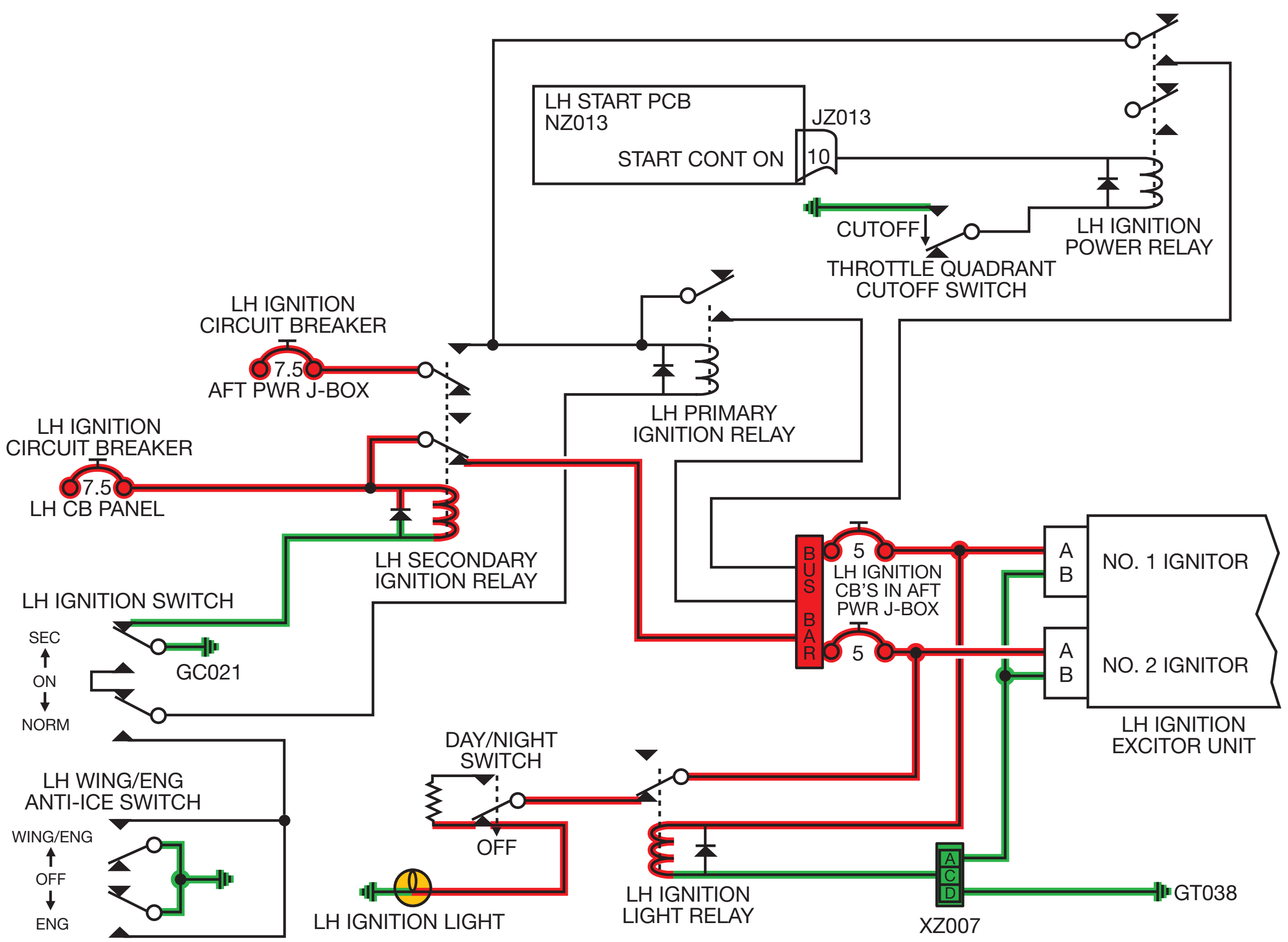
IGNITION SWITCH IN NORMAL



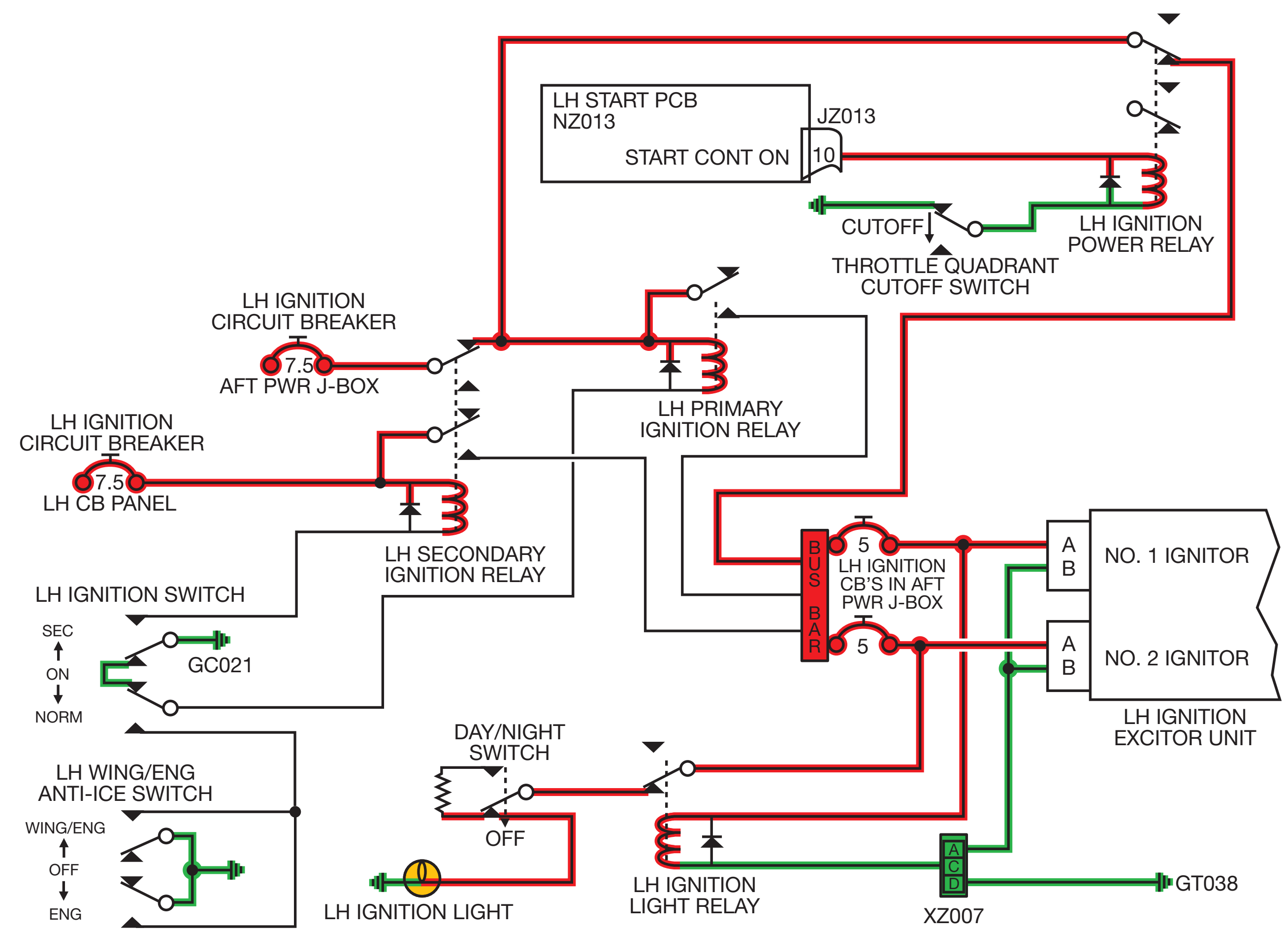
IGNITION SWITCH IN NORMAL

71-80

Figure 74-1. LH Ignition System - Units 5001-6000



IGNITION SWITCH IN SECONDARY



IGNITION SWITCH IN NORM DURING ENGINE START

Figure 74-2. LH Ignition System - Units 5001-6000

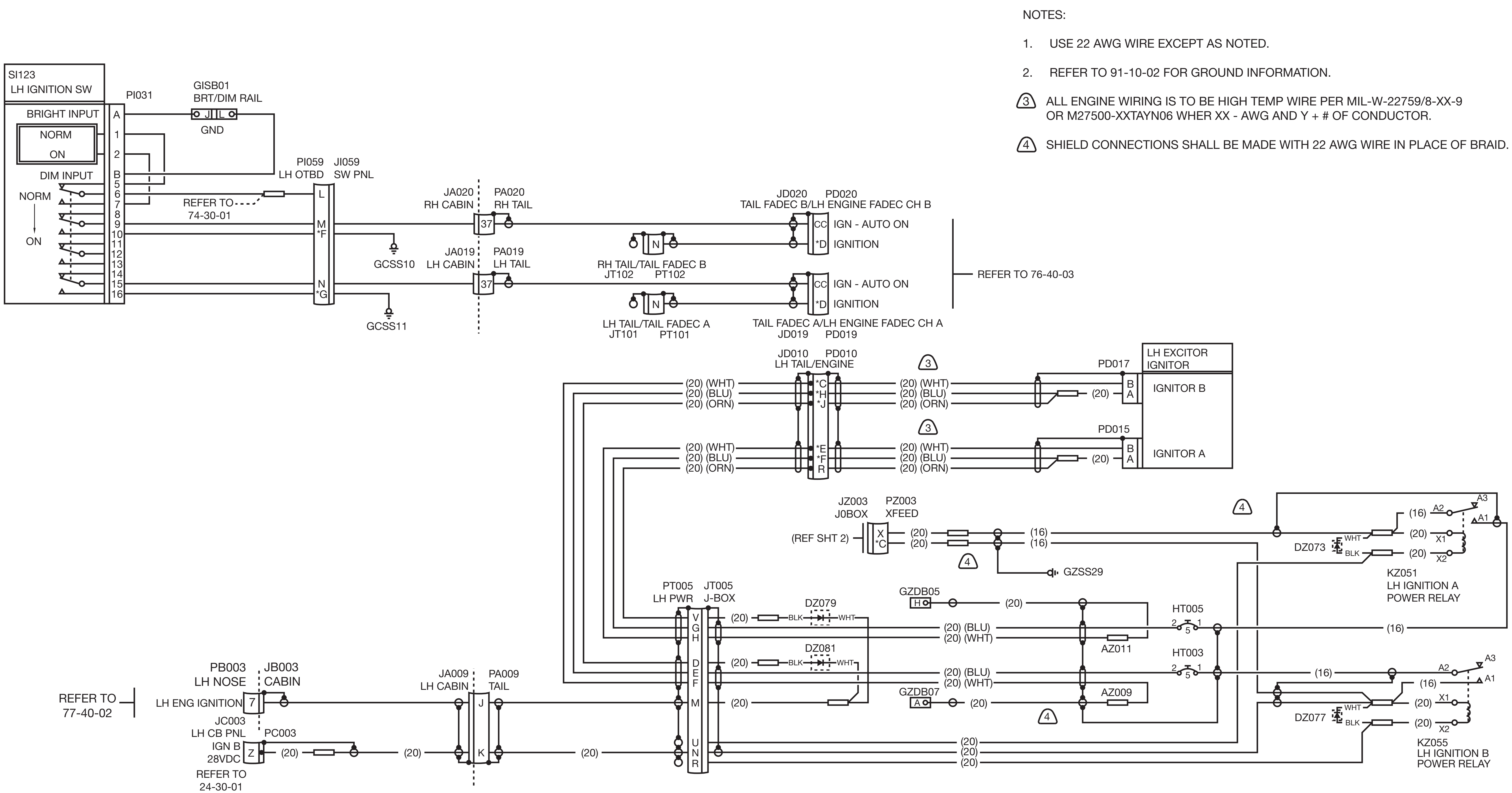


Figure 74-3. LH Ignition System - Units 6001 and Subsequent

71-80

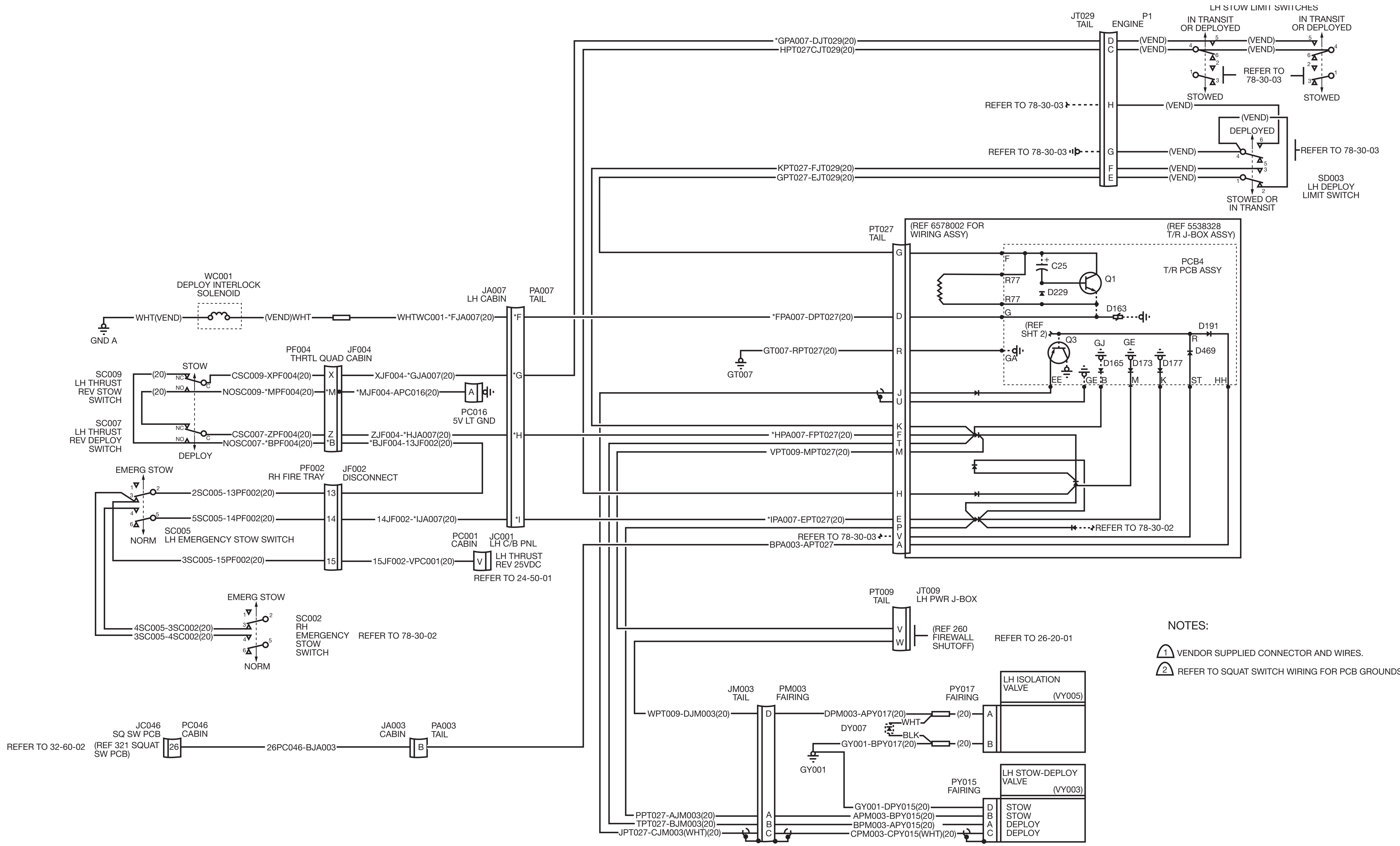
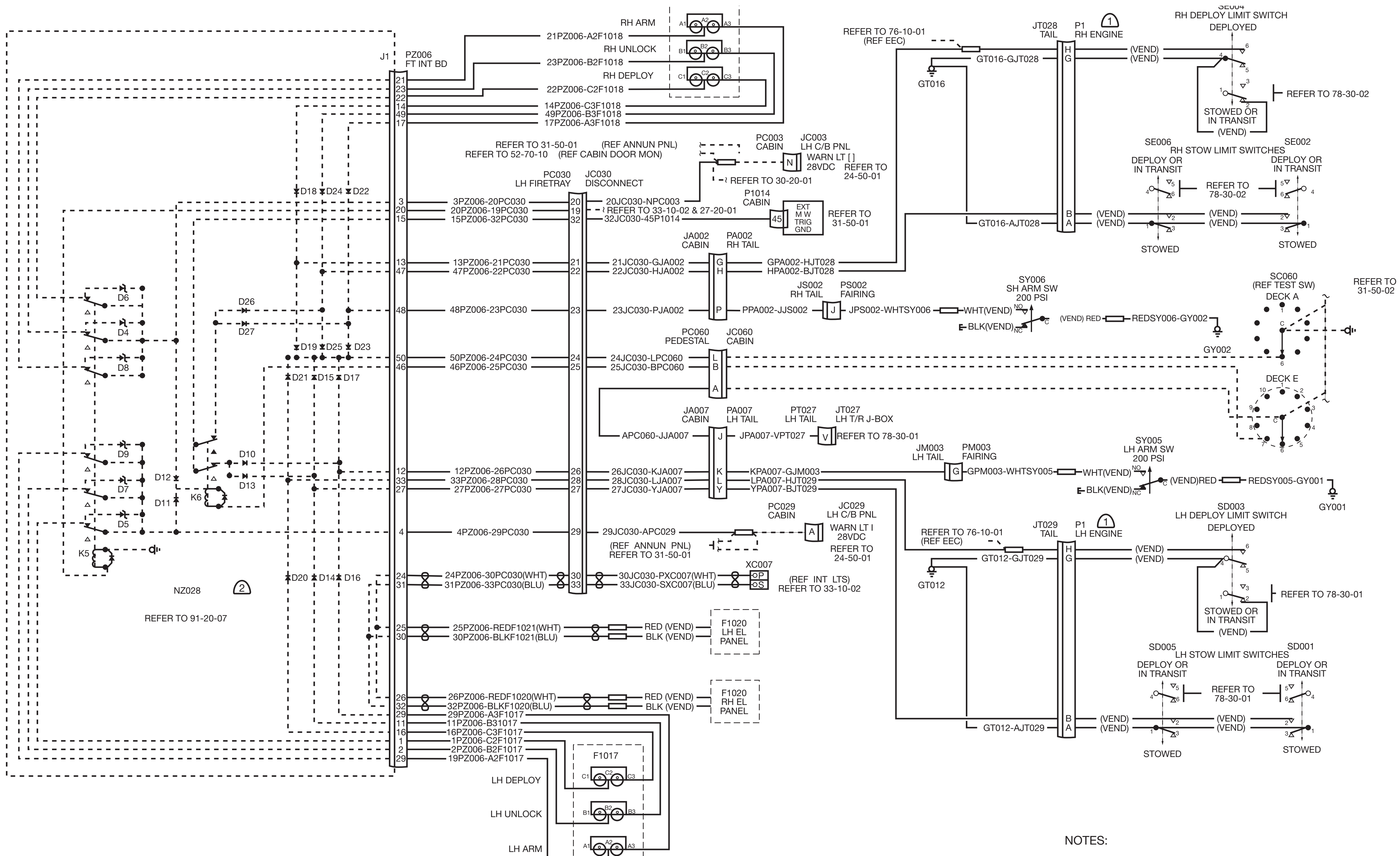


Figure 78-1. Left Thrust Reverser

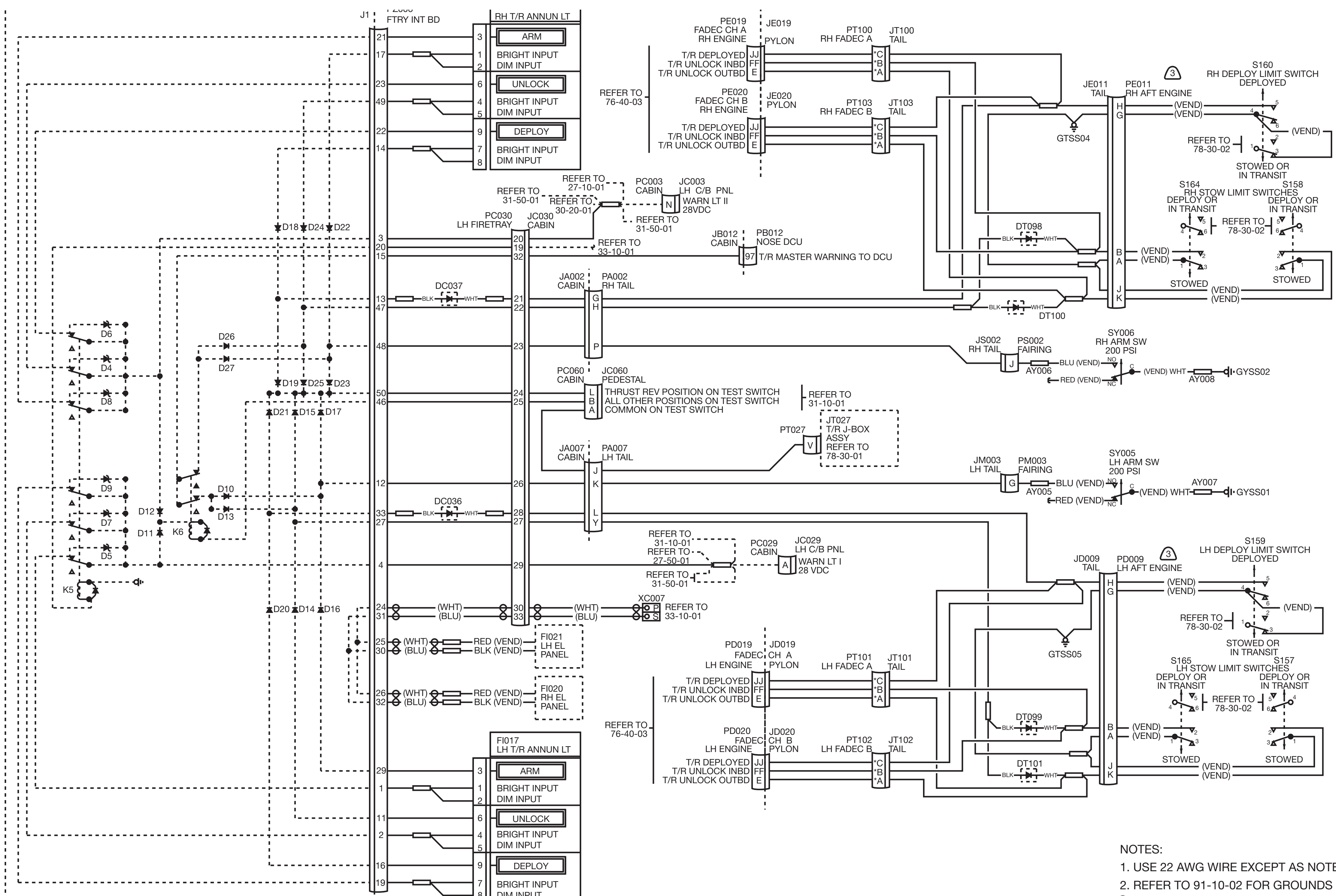
71-80



NOTES:

1. VENDOR SUPPLIED CONNECTOR AND WIRES.
2. REFER TO SQUAT SWITCH WIRING FOR PCB GROUNDS.

Figure 78-2. Thrust Reverser Annunciation - Units 5001-6000



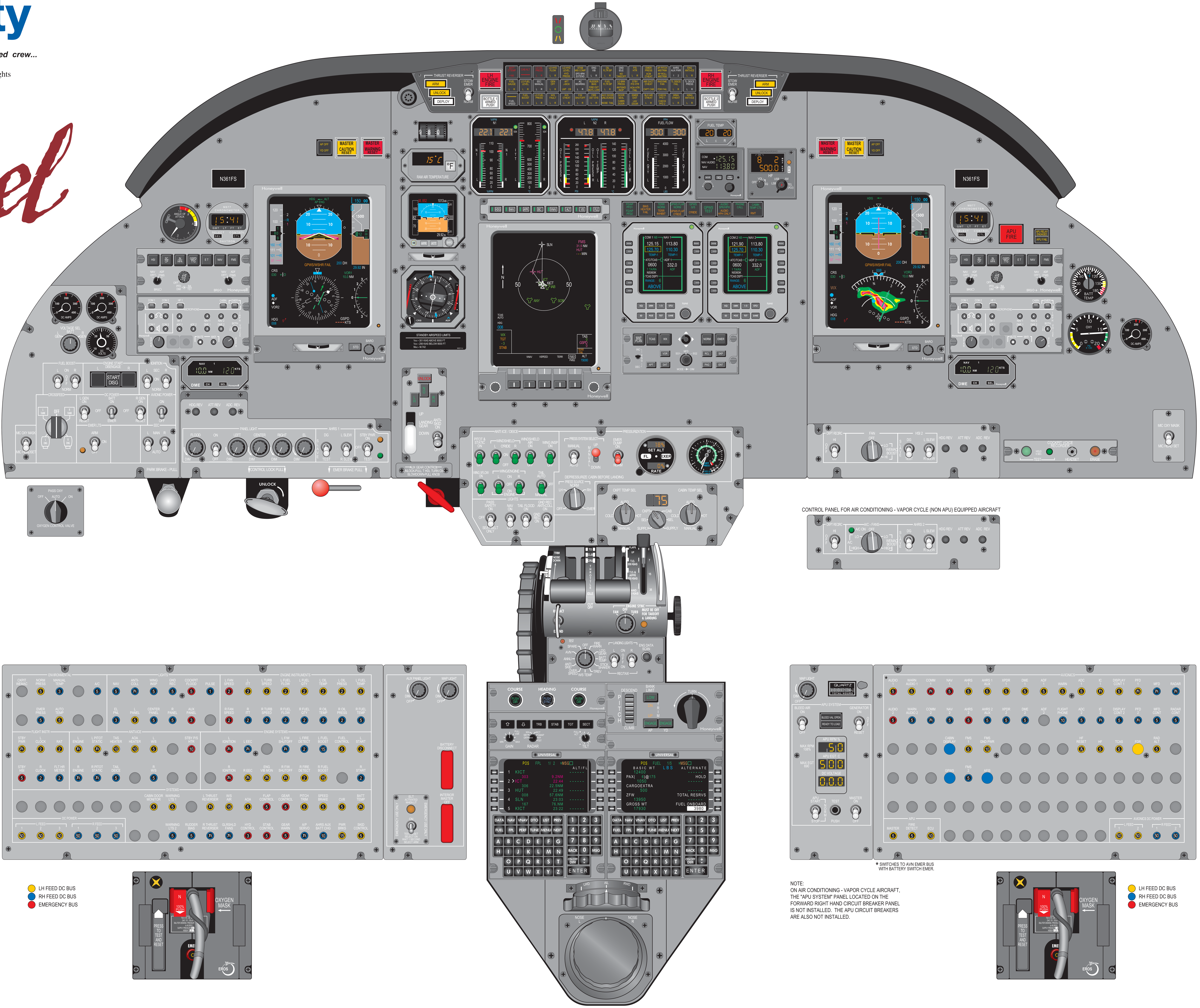
NOTES:
1. USE 22 AWG WIRE EXCEPT AS NOTED.
2. REFER TO 91-10-02 FOR GROUNDS INFORMATION.
3 VENDOR SUPPLIED CONNECTOR AND WIRES.

Figure 78-3. Thrust Reverser Annunciation - Units 6001 and Subsequent

FlightSafety international

the best safety device in any aircraft is a well trained crew...

CITATION Excel



050103

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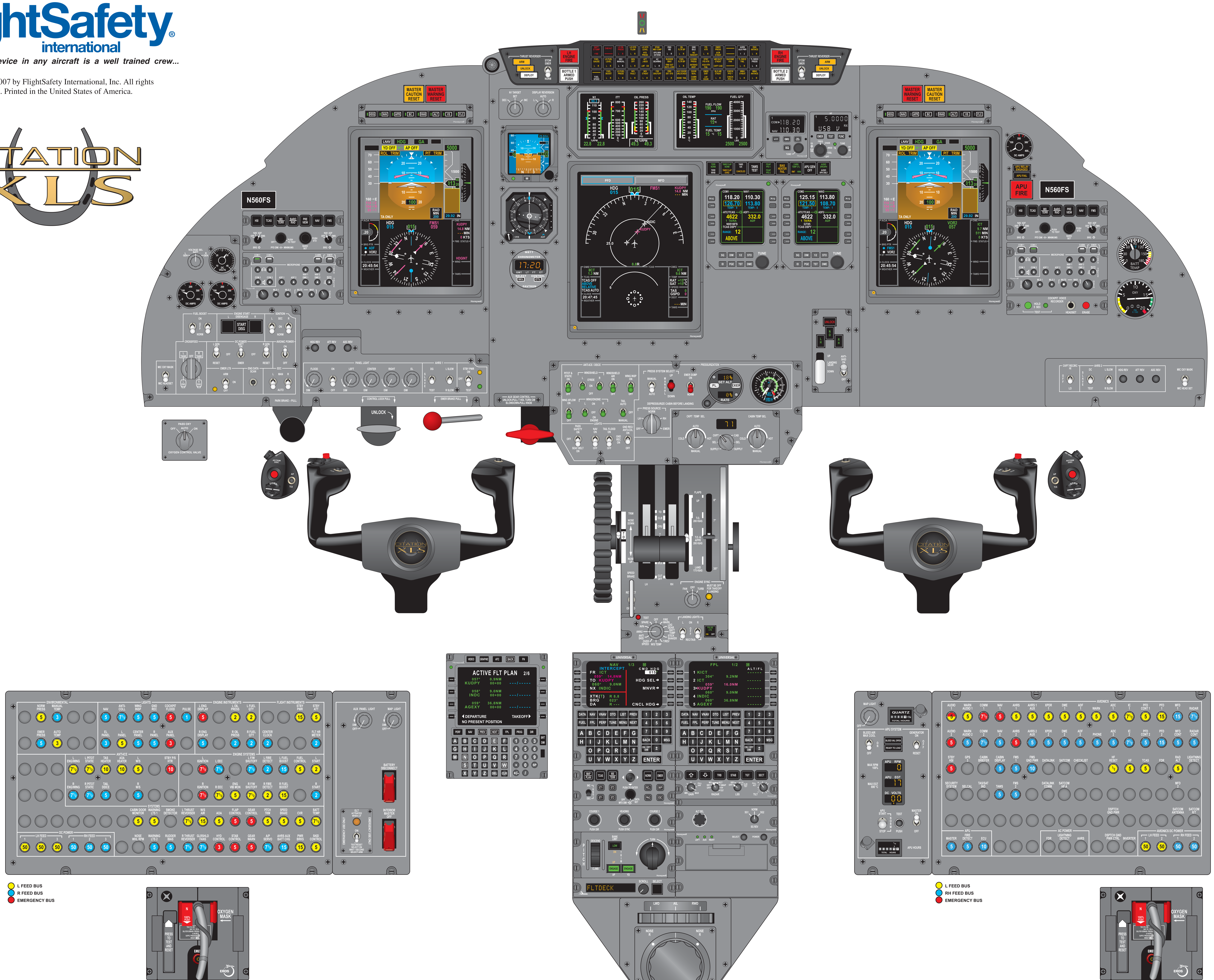
the best safety device in any aircraft is a well trained crew...

CITATION XLS



July 11, 2007

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the best safety device in any aircraft is a well trained crew...

Citation XLS+

July 16, 2008

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FlightSafety international

XL/XLS/XLS+ SERIES WALKAROUNDS

XL/XLS WALKAROUND

XLS+ WALKAROUND

Revision 0

Courses for the Citation XL/XLS/XLS+ Series are taught at the following FlightSafety Maintenance Learning Centers:

Wichita Cessna Maintenance Learning Center
2021 S. Eisenhower
Wichita, KS 67209
(316) 361-3900
(800) 491-9796
FAX (316) 361-3899

NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Pilot Manuals and Maintenance Manuals. It is to be used for familiarization and training purposes only.

At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

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INSERT LATEST REVISED PAGES, DESTROY SUPERSEDED PAGES

LIST OF EFFECTIVE PAGES

Dates of issue for original and changed pages are:

Original.......... 0 ....January 2016

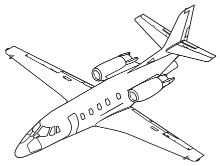
NOTE:

For printing purposes, revision numbers in footers occur at the bottom of every page that has changed in any way (grammatical or typographical revisions, reflow of pages, and other changes that do not necessarily affect the meaning of the manual).

THIS PUBLICATION CONSISTS OF THE FOLLOWING:

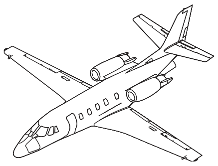
Page No.	*Revision No.	Page No.	*Revision No.
Cover	0	XL/XLS WA-1—XL/XLS WA-52	0
i—vi	0	XLS+ WA-1—XLS+ WA-50	0

*Zero in this column indicates an original page.



XL/XLS WALKAROUND

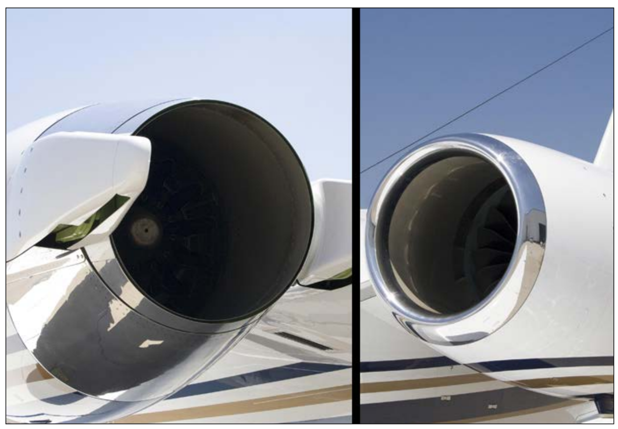
The following section is a pictorial walkaround. Each item listed in the exterior power-off preflight inspection is displayed. The general photographs contain circled numbers that correspond to specific steps displayed on the subsequent pages.



FlightSafety international

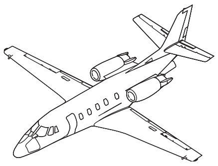
PREFLIGHT INSPECTION

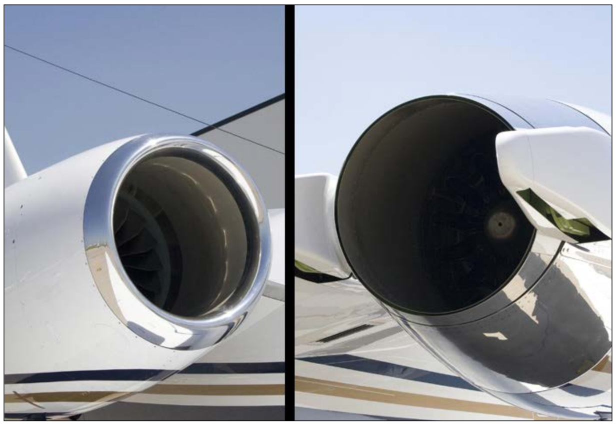
1. **BATTERY** - CONNECTED



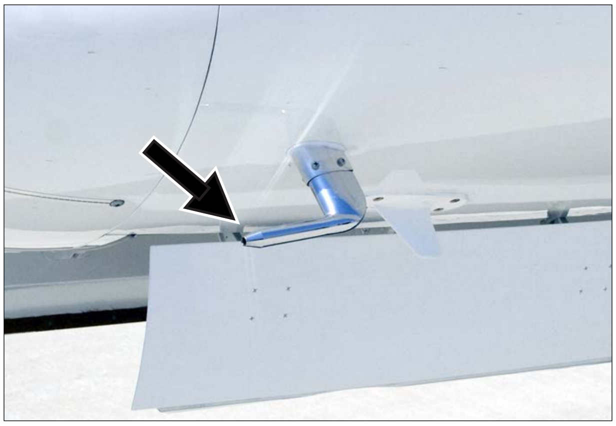
2. **ENGINE COVERS (4) -** REMOVED

WALKAROUND -XL/ XLS

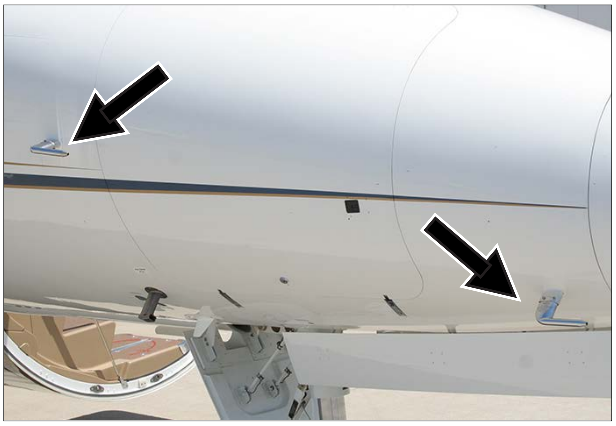




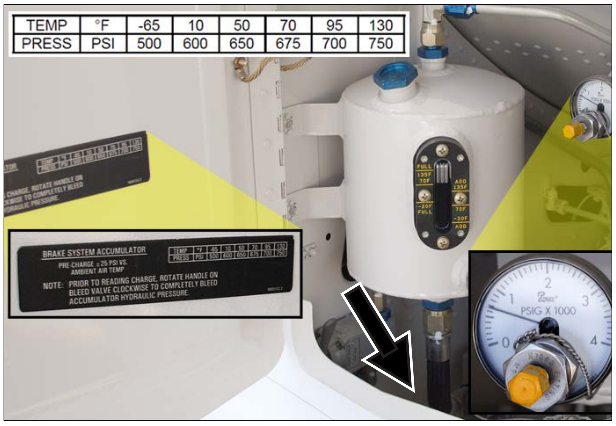
2. ENGINE COVERS (4) - REMOVED



3. PITOT COVERS (3) - REMOVED

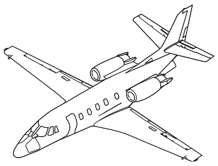


3. PITOT COVERS (3) - REMOVED

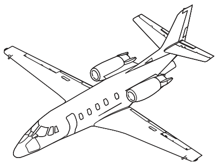


4. POWER BRAKE ACCUMULATOR CHARGE
- PER PLACARD

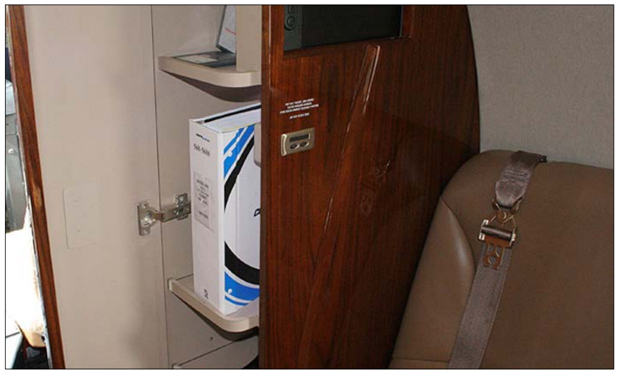
5. POWER BRAKE RESERVOIR SIGHT GAGE
- CHECK



PRELIMINARY COCKPIT INSPECTION

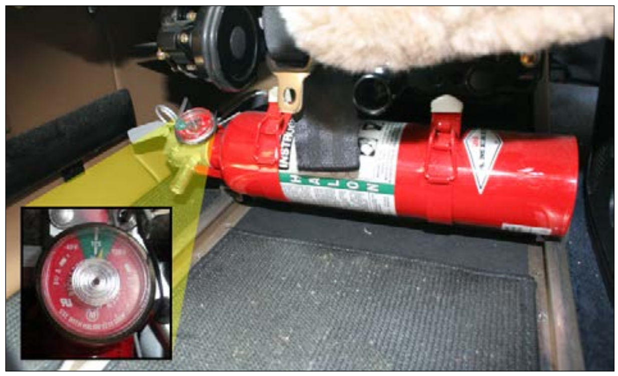


1. **DOCUMENTS** - CHECK ABOARD
   a. TO BE DISPLAYED IN THE AIRPLANE AT ALL TIMES:
      1) AIRWORTHINESS AND REGISTRATION CERTIFICATES
      2) RADIO STATION LICENSE(S) (IF REQUIRED).



1. **DOCUMENTS** - CHECK ABOARD
   b. TO BE CARRIED IN THE AIRPLANE AT ALL TIMES:
      1) FAA APPROVED AIRPLANE FLIGHT MANUAL
      2) HONEYWELL PRIMUS 1000 PILOT'S MANUAL
      3) APPLICABLE FMS PILOT'S MANUAL (IF REQUIRED)

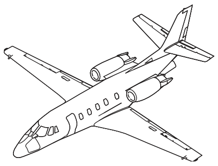
2. **FLASHLIGHT** - ABOARD



3. **PORTABLE FIRE EXTINGUISHER** - SERVICED AND SECURE (UNDER COPILOT'S SEAT)

4. **MICROPHONES, HEADSETS, OXYGEN MASKS AND SMOKE GOGGLES** - ABOARD AND PROPERLY STOWED

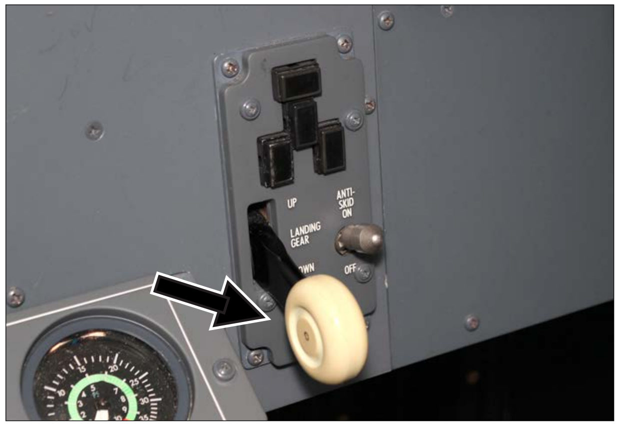
4. **MICROPHONES, HEADSETS, OXYGEN MASKS AND SMOKE GOGGLES** - ABOARD AND PROPERLY STOWED



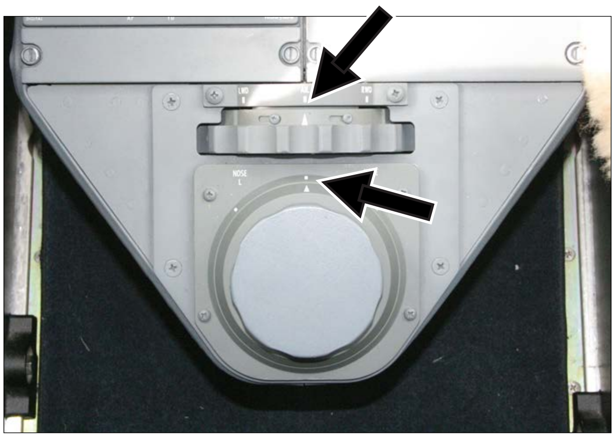
4. **MICROPHONES, HEADSETS, OXYGEN MASKS AND SMOKE GOGGLES -** ABOARD AND PROPERLY STOWED

5. **OXYGEN PRESSURE -** CHECK PER TABLE

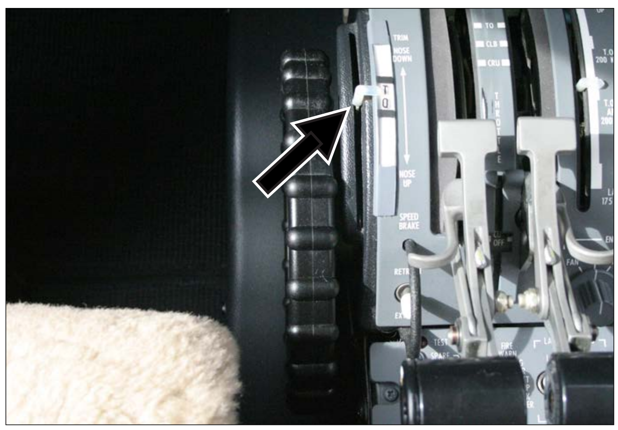
6. **CONTROL LOCK -** UNLOCKED



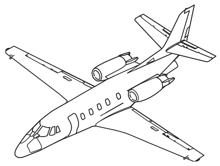
7. **LANDING GEAR HANDLE -** DOWN



8. **RUDDER AND AILERON TRIM -** POSITION PITCH TRIM TAB INDICATOR WITHIN TAKEOFF TRIM RANGE AND AILERON AND RUDDER TRIM TABS IN NEUTRAL



8. **RUDDER AND AILERON TRIM -** POSITION PITCH TRIM TAB INDICATOR WITHIN TAKEOFF TRIM RANGE AND AILERON AND RUDDER TRIM TABS IN NEUTRAL



9. FLAP HANDLE - AGREES WITH FLAP POSITION

10. CIRCUIT BREAKERS - IN

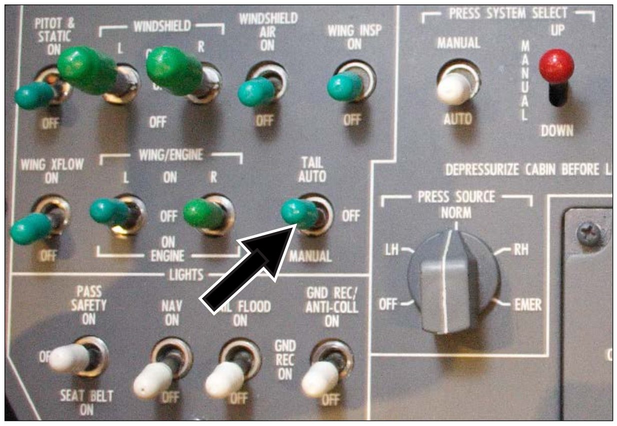
10. CIRCUIT BREAKERS - IN

11. GENERATOR SWITCHES - ON
(OFF, IF EXTERNAL POWER IS TO BE USED FOR START)

12. ALL OTHER SWITCHES AND BUTTONS
- OFF/NORM/AUTO

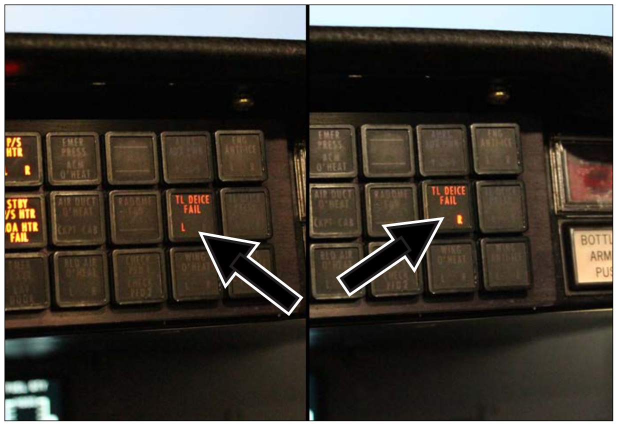
13. THROTTLES - CUT OFF

14. BATT SWITCH - ON (24 VOLTS MINIMUM)



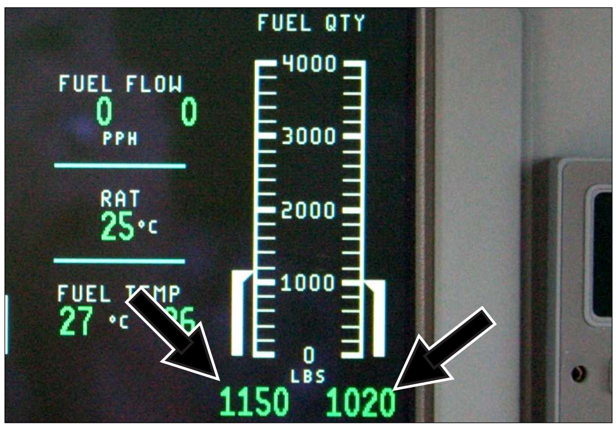
15. TAIL DEICE - CHECK

15.a. TAIL DEICE - AUTO



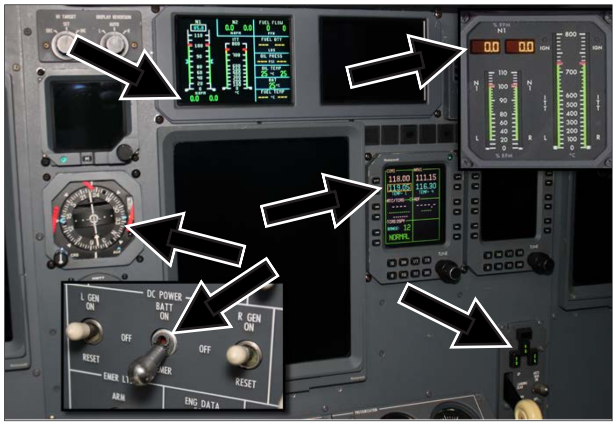
15.b. ANNUNCIATOR PANEL - AMBER TL DEICE FAIL L, THEN R (ALLOW 12 SECONDS FOR BOTH TO TEST)

15.c. TAIL DEICE - OFF



16. FUEL QUANTITY AND BALANCE - CHECK

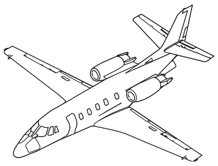
17. EXT/INT/EMER LIGHTS SWITCHES
- ON/CHECK/OFF OR AS REQUIRED



18. BATT SWITCH - EMER, CHECK N1 INDICATORS, RMU 1, STANDBY HSI AND LANDING GEAR INDICATOR RECEIVING POWER.

19. BATT SWITCH - ON

20. APU - CONSIDER USE



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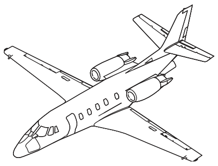
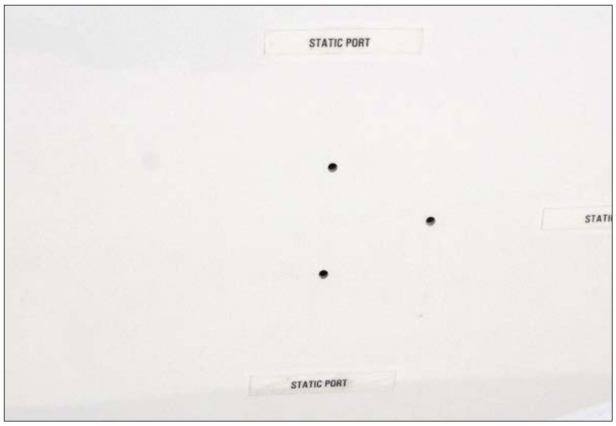
HOT ITEMS/LIGHTS

WALKAROUND -XL/ XLS



1. **HOT ITEMS/LIGHTS -** ON AND CHECK

1.a. LEFT, RIGHT AND STANDBY STATIC PORTS
- CLEAR AND WARM

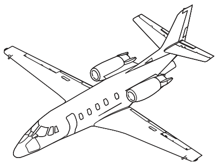


1.a. LEFT, RIGHT AND STANDBY STATIC PORTS
- CLEAR AND WARM

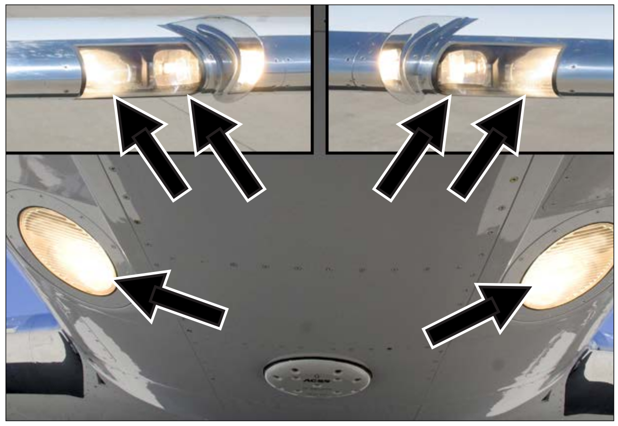
1.b. LEFT, RIGHT AND STANDBY PITOT TUBES
- CLEAR AND HOT

1.b. LEFT, RIGHT AND STANDBY PITOT TUBES
- CLEAR AND HOT

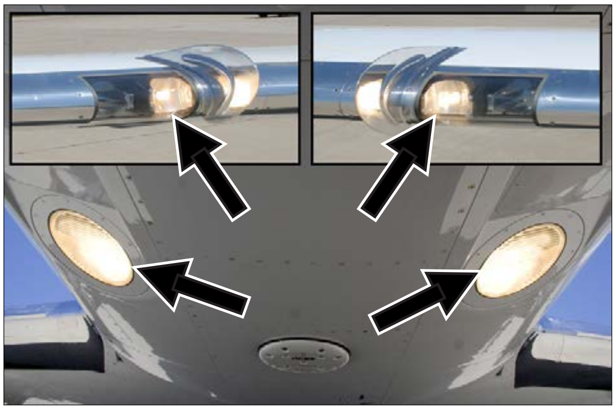
1.b. LEFT, RIGHT AND STANDBY PITOT TUBES
- CLEAR AND HOT



1.c. TRUE AIRSPEED TEMEPRATURE PROBE - CLEAR



1.d. LANDING LIGHTS - ALL ON
(IF NOT OBSERVED FROM COCKPIT)



1.e. REC/TAXI LIGHTS - ALL ON
(IF NOT OBSERVED FROM COCKPIT)

1.f. ANGLE-OF-ATTACK VANE - FREE AND HOT

1.g. GROUND RECOGNITION LIGHT - ON AND FLASHING
(IF NOT OBSERVED FROM COCKPIT)

1.h. RIGHT WING INSPECTION, NAVIGATION, AND ANTI-COLLISION LIGHTS - ON
(IF NOT OBSERVED FROM COCKPIT)

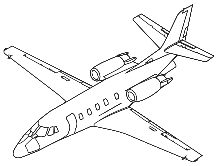
1.h. RIGHT WING INSPECTION, NAVIGATION, AND ANTI-COLLISION LIGHTS - ON (IF NOT OBSERVED FROM COCKPIT)

1.l. TAIL NAVIGATION LIGHT - ON

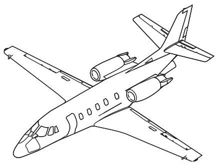
1.j. NAVIGATION, ANTI-COLLISION, AND LEFT WING INSPECTION LIGHTS - ON

1.j. NAVIGATION, ANTI-COLLISION, AND LEFT WING INSPECTION LIGHTS - ON

1.k. HOT ITEMS/LIGHTS AND BATTERY SWITCHES - OFF

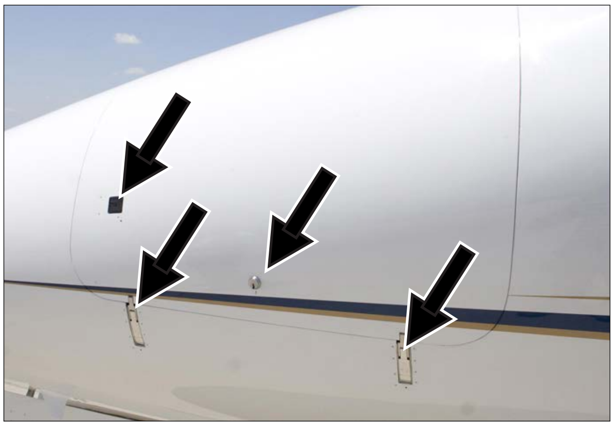


LEFT NOSE

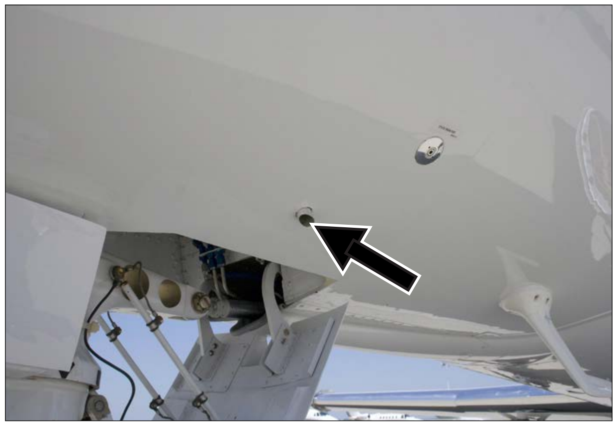


2. LEFT NOSE - CHECK

2.a. ANTISKID FAULT DISPLAY UNIT (BITE INDICATOR)
- CHECK AND RESET IF REQUIRED. VERIFY AMBER ANTISKID INOP ANNUNCIATOR EXTRINGUISHED DURING BEFORE STARTENG ENGINES CHECKLIST.



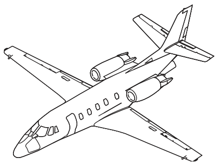
2.b. ACCESSORY DOOR - SECURE AND LOCKED



2.c. OVERBOARD VENT LINES - CLEAR

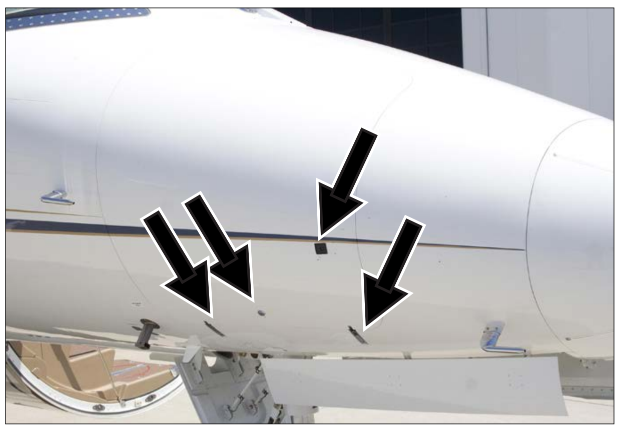
2.d. NOSE GEAR, DOORS, WHEEL AND TIRE - CONDITION

2.e. NOSE GEAR TORQUE LINK
- CHECK CONDITION (IF INSTALLED)

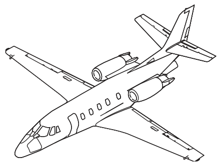


RIGHT NOSE AND FUSELAGE RIGHT SIDE

3. RIGHT NOSE AND FUSELAGE RIGHT SIDE - CHECK



3.a. ACCESSORY DOOR - SECURE AND LOCKED



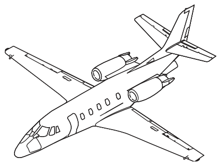
3.b. OXYGEN BLOWOUT DISC - GREEN

3.c. SINGLE POINT PRESSURE REFUEL DOOR
- LATCHED AND LOCKED

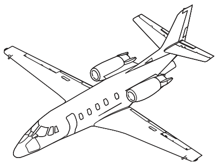
3.d. TOP AND BOTTOM ANTENNAS
- CONDITION AND SECURE

3.d. TOP AND BOTTOM ANTENNAS
- CONDITION AND SECURE

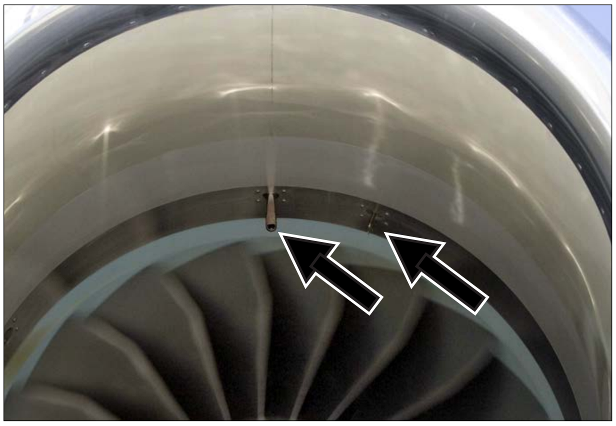
3.e. DORSAL FIN AIR INLET - CLEAR



RIGHT WING



4. **RIGHT WING -** CHECK



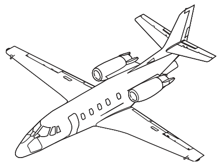
4.a. **T1 AND T0 SENSORS (IN RIGHT ENGINE INLET) -** CONDITION

4.b. **ENGINE FAN DUCT AND FAN -** CONDITION

4.c. **WING INSPECTION LIGHT -** CONDITION

4.d. **EMERGENCY EXIT -** CHECK CONDITION

4.e. **ANTI-ICE BLEED AIR COOLING AIR INLET -** CLEAR



4.f. FUEL QUICK DRAINS (5)
- DRAIN AND CHECK FOR CONTAMINATION

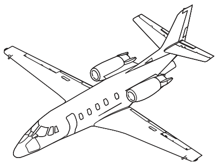
4.g. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE
- CONDITION AND SECURE

4.g. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE
- CONDITION AND SECURE

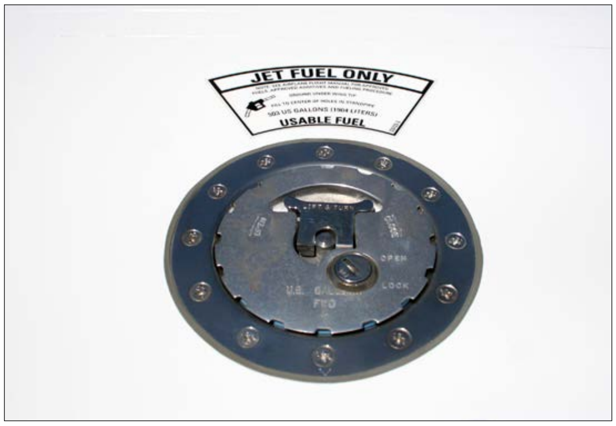
4.h. WHEEL WELL - CONDITION/NO LEAKS

4.i. VORTEX GENERATORS (26) - CHECK
(NO MORE THAN 3 MAY BE MISSING ON ENTIRE PLANE)

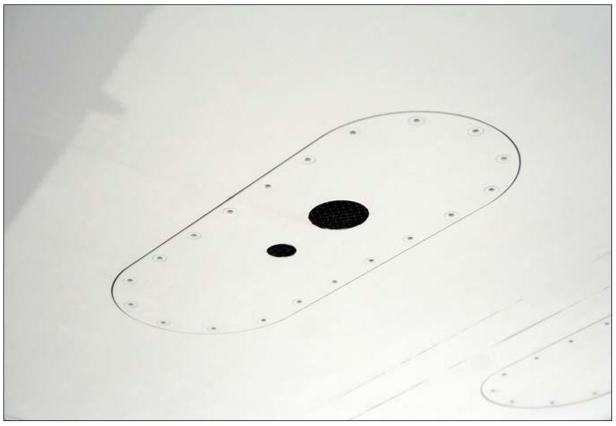
4.j. BOUNDARY LAYER ENERGIZERS (11)
- CHECK (NONE MAY BE MISSING)



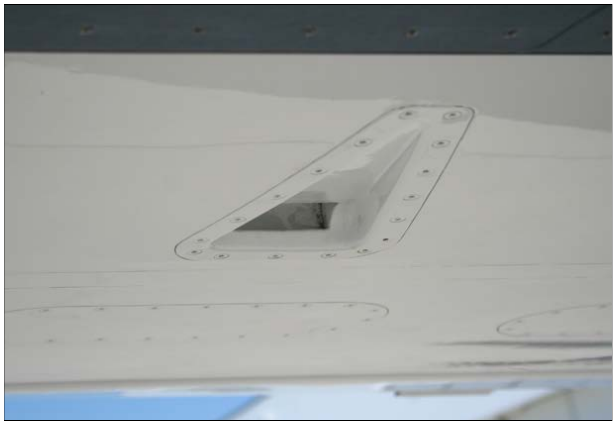
4.k. HEATED LEADING EDGE
- CONDITION AND VENT CLEAR



4.l. FUEL FILLER CAP - SECURE

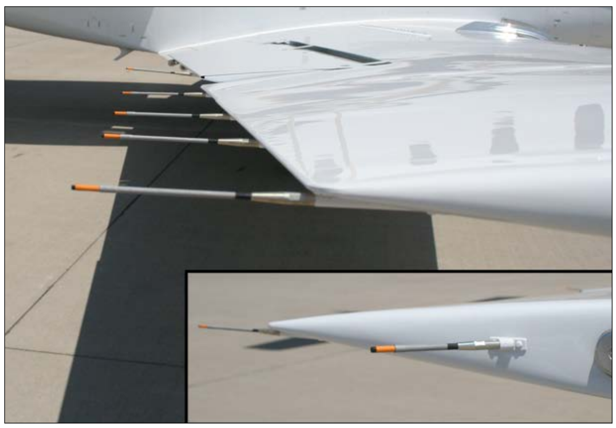


4.m. FUEL TANK RELIEF VALVES - CONDITION/NO LEAKS

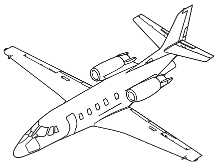


4.n. FUEL TANK VENT - CLEAR

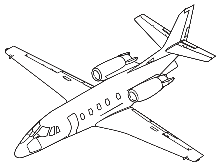
4.o. RECOGNITION, LANDING, NAVIGATION, AND STROBE LIGHTS - CONDITION



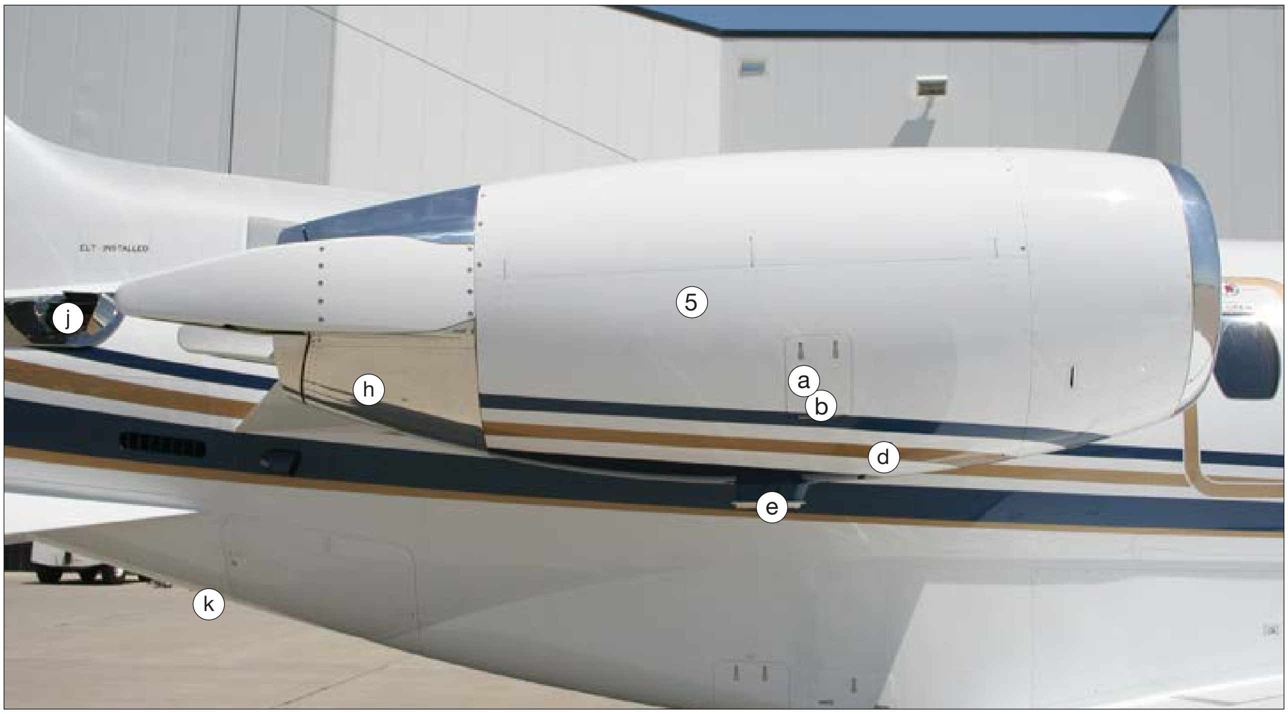
P. STATIC WICKS (6) - CHECK

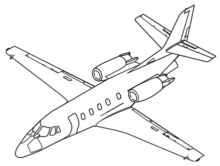


4.q. AILERON, SPEED BRAKES, AND FLAPS
- CONDITION AND SECURE



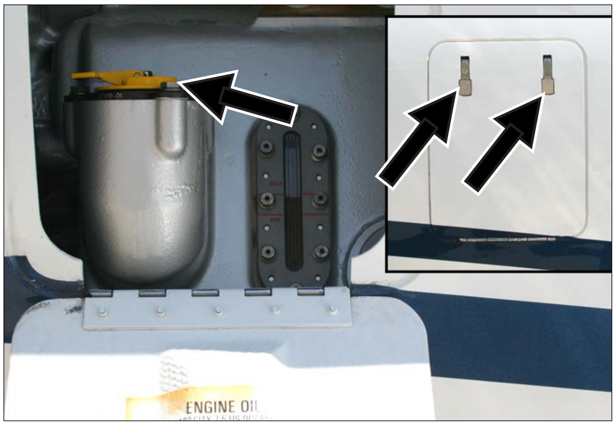
RIGHT NACELLE/PYLON





5. RIGHT NACELLE/PYLON - CHECK

5.a. OIL LEVEL - CHECK

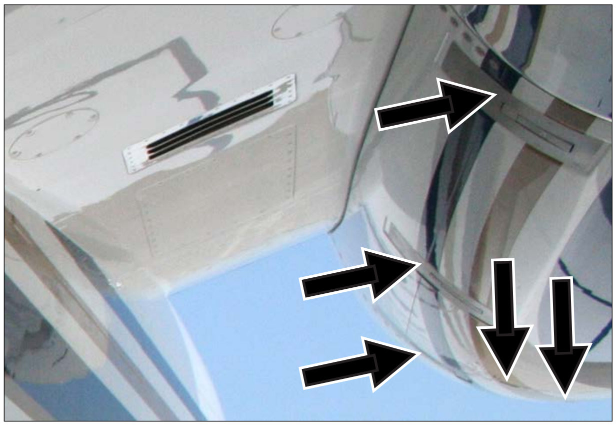


5.b. FILLER CAP AND ACCESS DOOR - SECURE

5.c. PYLON PRE-COOLER EXHAUST DUCT - CLEAR

5.d. GENERATOR AND ALTERNATOR COOLING AIR EXHAUST - CLEAR

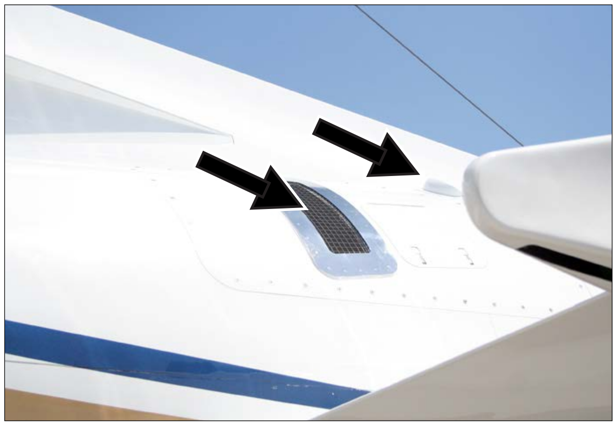
5.e. ENGINE FLUID DRAIN MAST - CLEAR



5.f. COWLING - SECURED/ALL LATCHES (5) LATCHED

5.g. ENGINE EXHAUST AND BYPASS DUCTS
- CONDITION AND CLEAR

5.h. THRUST REVERSER BUCKETS
- CONDITION AND STOWED

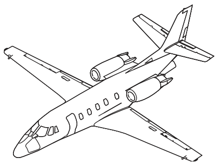


5.i. APU ENGIEN AND GENERATOR COOLING INLET
- CLEAR (LOCATED ABOVE RIGHT ENGINE PYLON)

5.j. APU EXHAUST - CLEAR
(LOCATED ABOVE RIGHT ENGINE PYLON)

5.k. APU DRAIN - CLEAR
(LOCATED ON BOTTOM RIGHT SIDE OF TAILCONE)

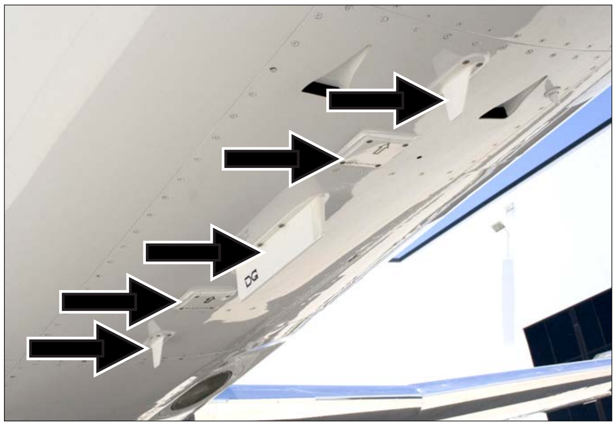
RIGHT AFT FUSELAGE



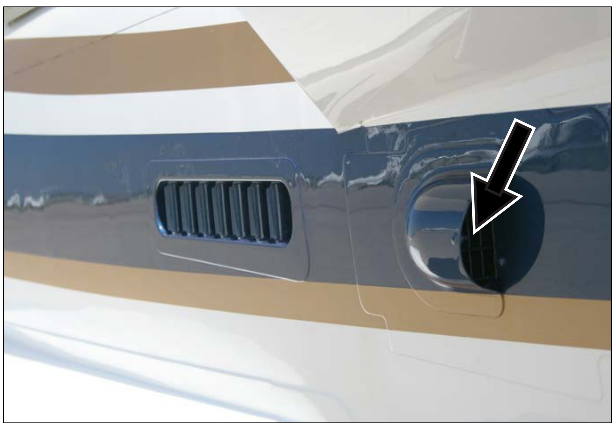
6. RIGHT AFT FUSELAGE - CHECK

6.a. HYDRAULIC AND TOILET (IF INSTALLED) SERVICE DOORS - SECURE

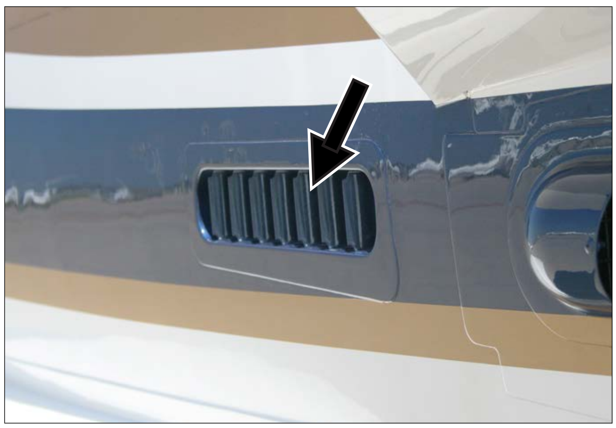
6.b. HYDRAULIC DRAIN MAST - NO LEAKS



6.c. HYDRAULIC AND TOILET SERVICE DOORS - SECURE

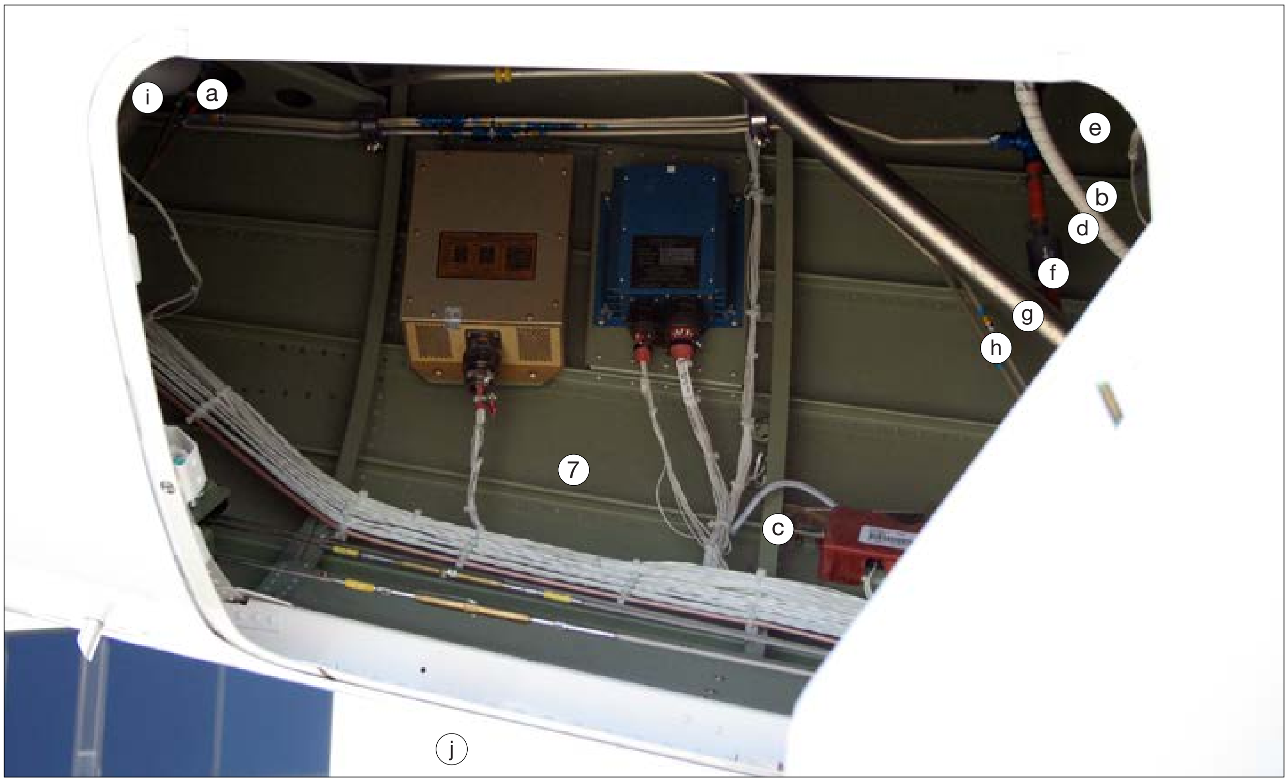


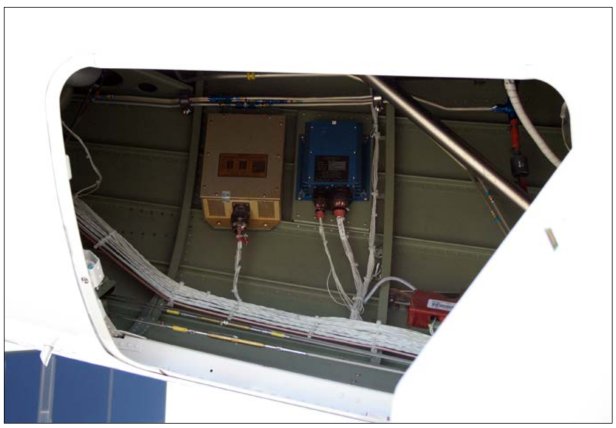
6.d. TAILCONE POSITIVE PRESSURE INLET - CLEAR

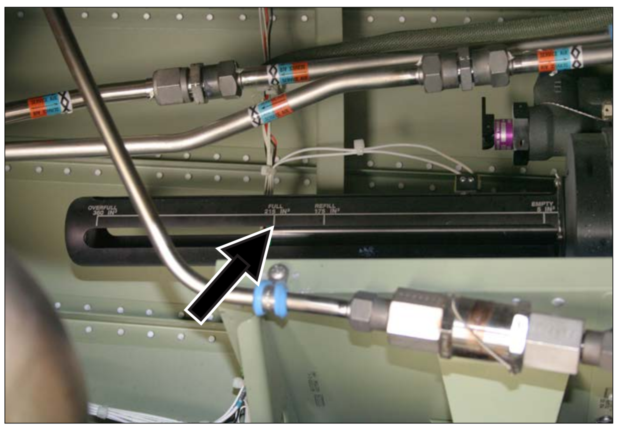


6.e. ACM EXHAUST - CLEAR

TAILCONE COMPARTMENT

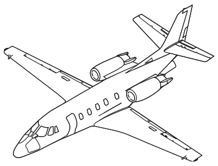


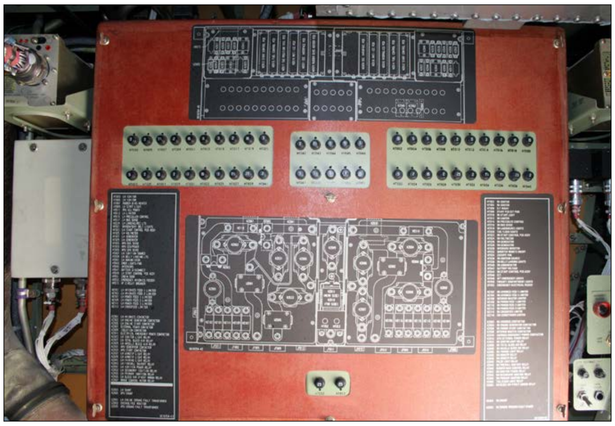




7. TAILCONE COMPARTMENT - CHECK

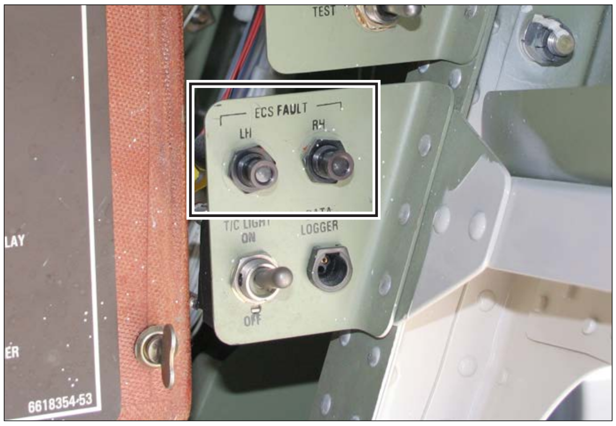
7.a. HYDRAULIC FUILD QUANTITY - CHECK



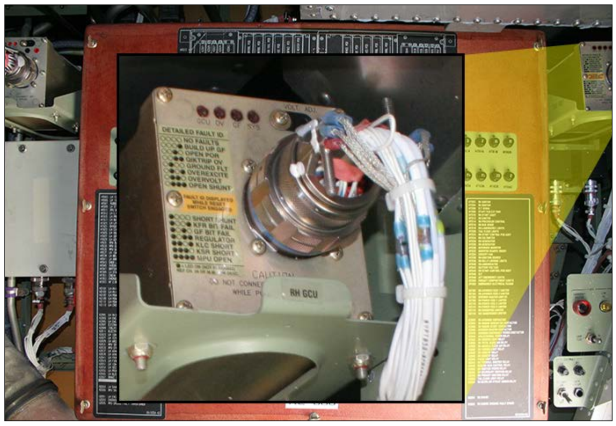


7.b. **AFT JUNCTION BOX CIRCUIT BREAKERS -** IN

7.c. **EXTERNAL POWER RECEPTACLE CIRCUIT BREAKER -** IN



7.d. **ECS INDICATORS -** CHECK AND RESET IF REQUIRED
VERIFY ENGINE(S) OPERATE IN AUTO MODE AFTER ENGINE START

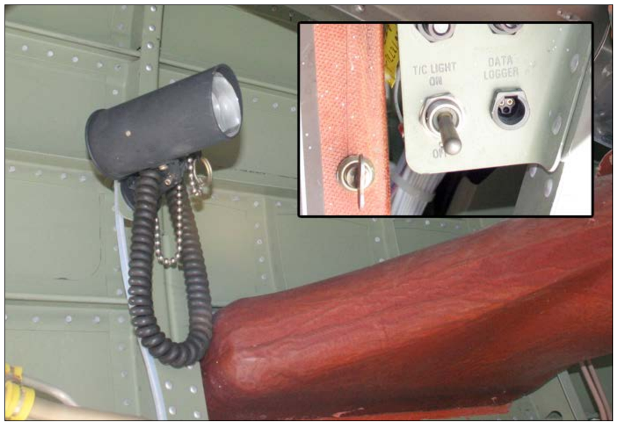


7.e. **LH AND RH GCU** - CHECK AND RESET IF REQUIRED

7.f. **ENGINE CHIP DETECTOR (IF INSTALLED) -** TEST

7.g. **APU SERVICE PANEL SWITCH -** LAMP TEST
(AMBER LIGHT SHOULD ILLUMINATE)

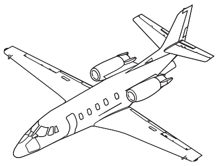
7.h. APU SERVICE PANEL SWITCH - PRE FLT
IF THE AMBER LOW OIL LIGHT IS ILLUMINATED, THE APU MAY BE OEPRATED FOR A MAXIMUM OF 20 HOURS PRIOR TO SERVICING THE AIRPLANE.



7.i. TAILCONE LIGHT - OFF

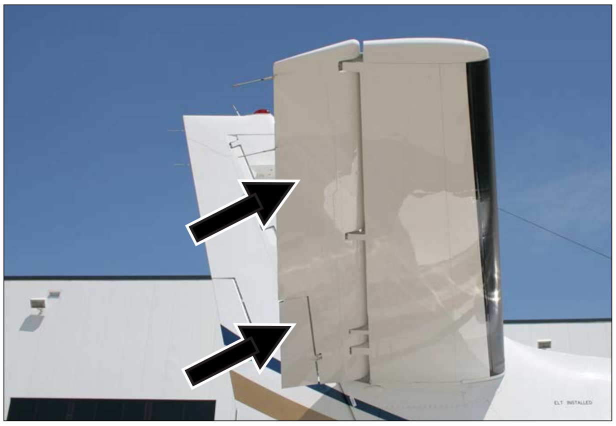
7.j. TAILCONE ACCESS DOOR - SECURE AND LOCKED

RIGHT EMPENNAGE



8. **RIGHT EMPENNAGE -** CHECK

8.a. **RIGHT HORIZONTAL STABILIZER DEICE BOOT -** CONDITION AND SECURE



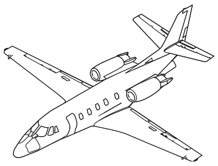
8.b. **RIGHT ELEVATOR AND TRIM TAB -** CONDITION

8.c. **RUDDER AND TRIM TAB -** CONDITION

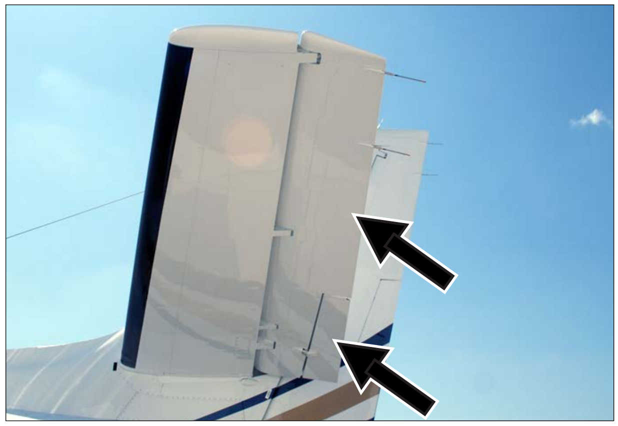
8.d. **STATIC WICKS (RUDDER, VERTICAL STABILIZER AND BOTH ELEVATORS) (8) -** CHECK

8.e. **TAIL STRAKES -** CONDITION AND SECURE

LEFT EMPENNAGE



9. LEFT EMPENNAGE - CHECK



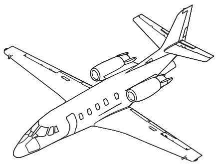
9.a. LEFT ELEVATOR AND TRIM TAB - CONDITION

9.b. LEFT HORIZONTAL STABILIZER DEICE BOOT
- CONDITION AND SECURE

9.c. HORIZONTAL STABILIZER POSITION INDEX
- CHECK; AGREES WITH FLAP POSITION

9.d. EXTERNAL POWER SERVICE DOOR - SECURE

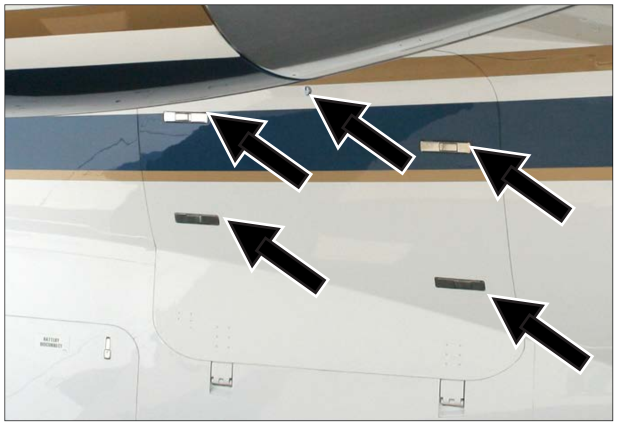
BAGGAGE COMPARTMENT



10. BAGGAGE COMPARTMENT - CHECK

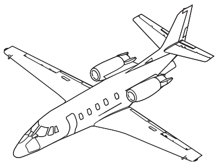
10.a. BAGGAGE COMPARTMENT - SECURE

10.b. BAGGAGE COMPARTMENT LIGHT - OFF

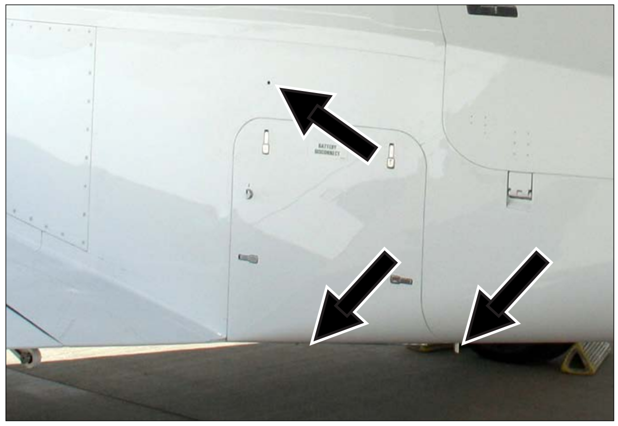


10.c. BAGGAGE COMPARTMENT ACCESS DOOR
- SECURE AND LOCKED

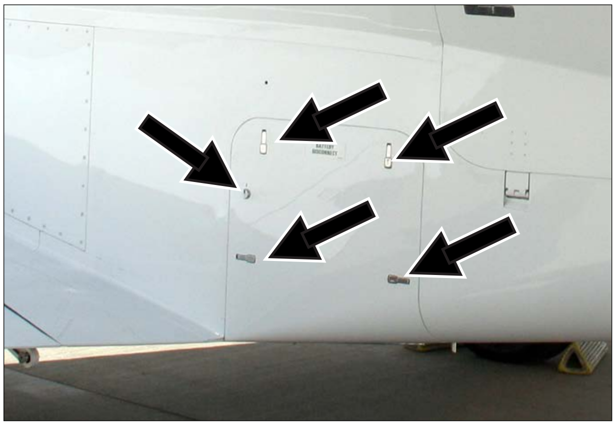
LEFT AFT FUSELAGE



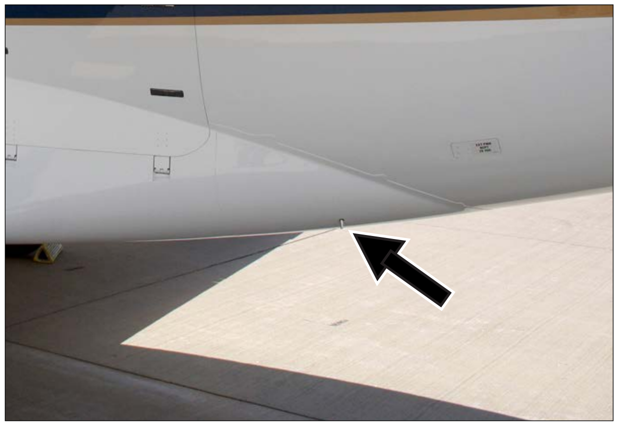
11. LEFT AFT FUSELAGE - CHECK



11.a. BATTERY COOLING INTAKE AND VENT LINES
- CLEAR



11.b. BATTERY COMPARTMENT ACCESS DOOR
- SECURE AND LOCKED

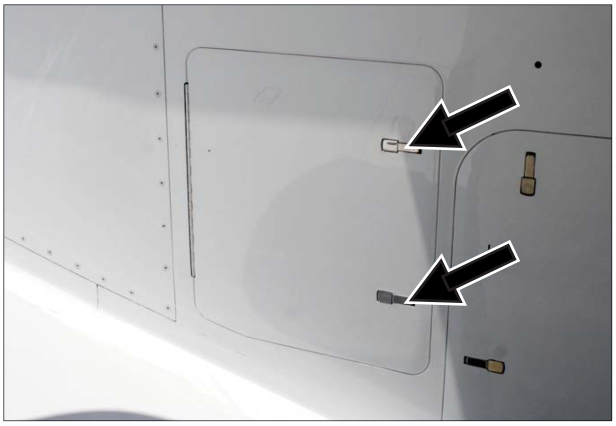


11.c. BRAKE RESERVOIR OVERBOARD VENT LINE
- CLEAR

BRAKE COMPARTMENT

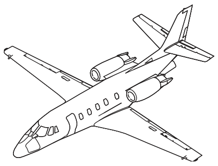
12. **BRAKE COMPARTMENT -** SECURE

12.a. **BRAKE AND GEAR PNEUMATIC GAGE**
- PER PLACARD



12.b. **BRAKE COMPARTMENT ACCESS DOOR -** SECURE

LEFT NACELLE/PYLON

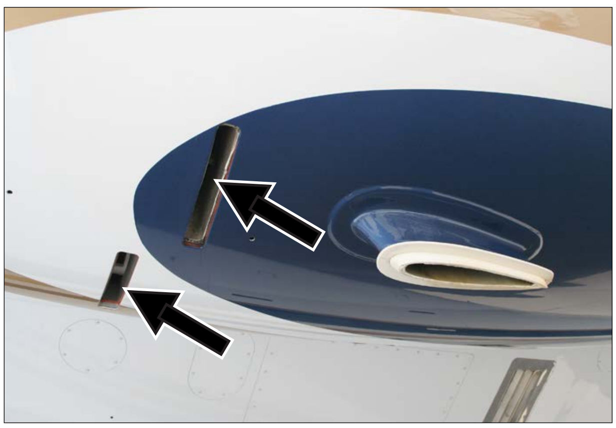


13. LEFT NACELLE/PYLON - CHECK

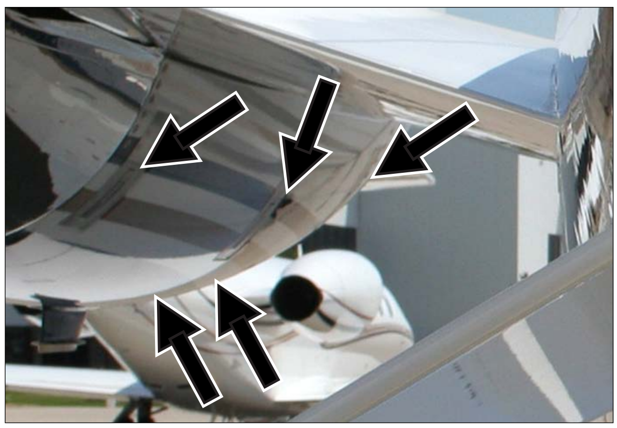
13.a. THRUST REVERSER BUCKETS
- CONDITION AND STOWED

13.b. ENGINE EXHAUST AND BYPASS DUCTS
- CONDITION AND CLEAR

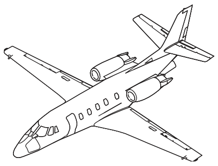
13.c. ENGINE FLUID DRAN MAST - CLEAR



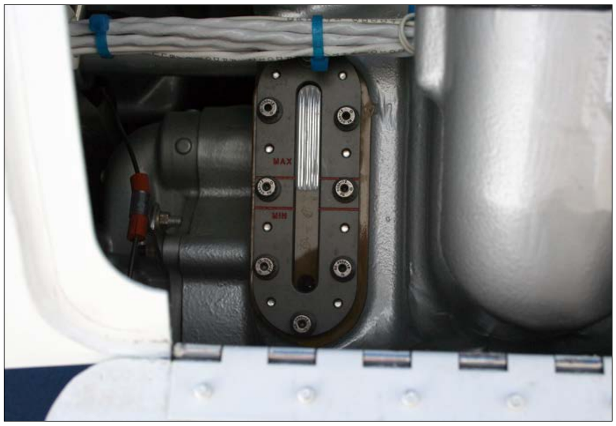
13.d. GENERATOR AND ALTERNATOR COOLING AIR EXHAUST - CLEAR



13.e. COWLING - SECURED/ALL LATCHES (5) LATCHED

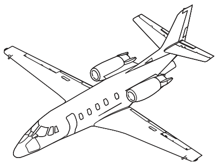


13.f. PYLON PRE-COOLER EXCHANGER EXHAUST DUCT - CLEAR



13.g. OIL LEVEL - CHECK

13.h. FILLER CAP AND ACCESS DOOR - SECURE



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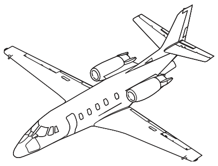
LEFT WING

WALKAROUND -XL/ XLS



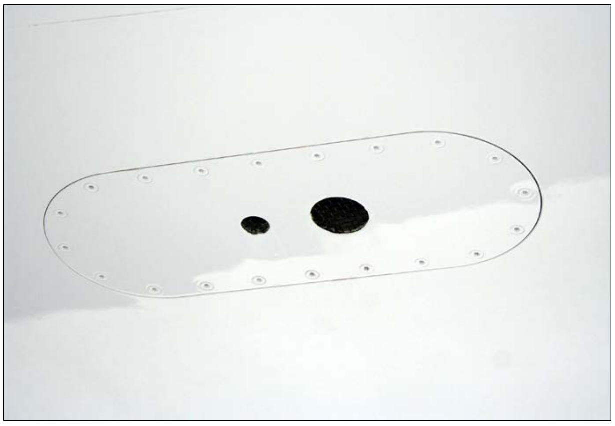
14. LEFT WING - CHECK

14.a. FLAP, SPEED BRAKES, AILERON AND TRIM TAB - CONDITION AND SECURE

14.b. STATIC WICKS (6) - CHECK

14.c. NAVIGATION, STROBE, LANDING AND RECOGNITION LIGHTS - CONDITION

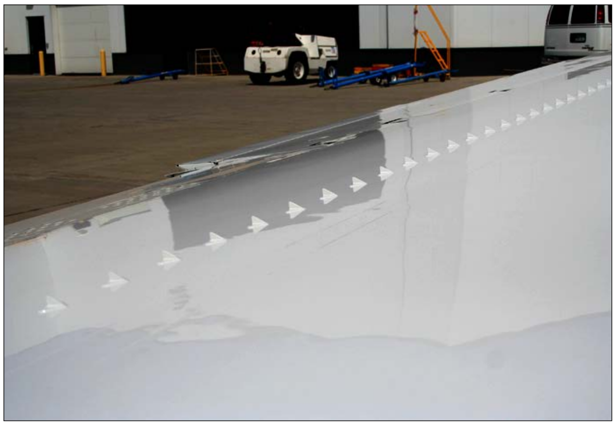
14.d. FUEL TANK VENT - CLEAR



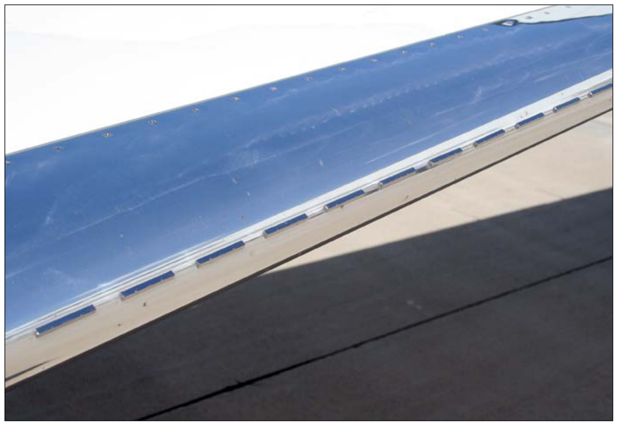
14.e. FUEL TANK RELIEF VALVES - CONDITION/NO LEAKS

14.f. FUEL FILLER CAP - SECURE

14.g. HEATED LEADING EDGE
- CONDITION AND VENT CLEAR



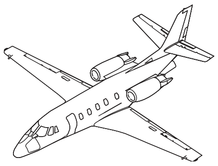
14.h. VORTEX GENERATORS (26) - CHECK (NO MORE THAN 3 MAY BE MISSING ON ENTIRE PLANE)



14.i. BOUNDARY LAYER ENERGIZERS (11) - CHECK (NONE MAY BE MISSING)

14.j. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE
- CONDITION AND SECURE

14.j. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE
- CONDITION AND SECURE



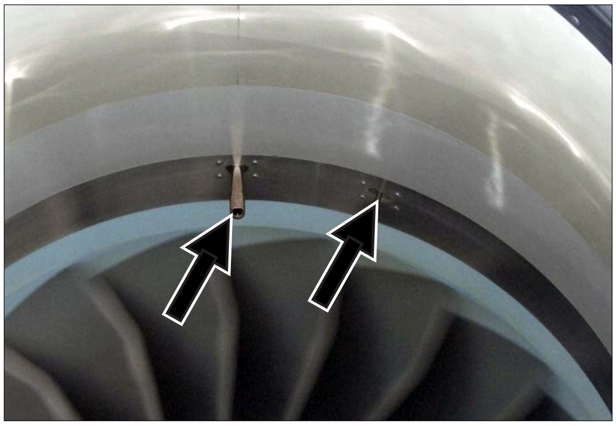
14.k. WHEEL WELL - CONDITION/NO LEAKS

14.l. FUEL QUICK DRAINS (5) - DRAIN AND CHECK FOR CONTAMINATION

**14.m. ANTI-ICE BLEED AIR COOLING AIR INLET - ** CLEAR

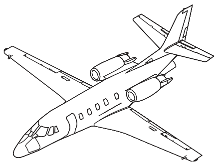
14.n. WING INSPECTION LIGHT - CONDITION

14.o. ENGINE FAN DUCT AND FAN - CONDITION

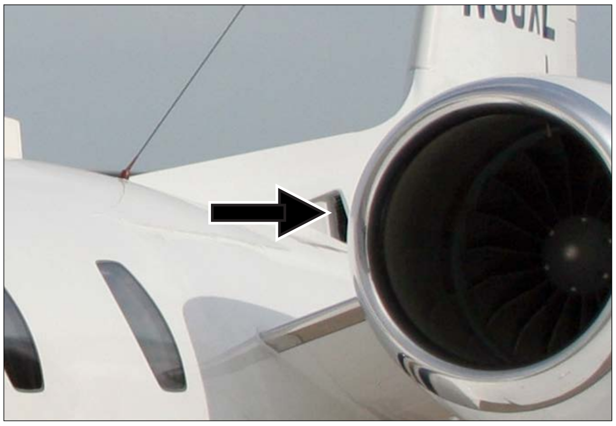


14.p. T1 AND T0 SENSORS (IN LEFT ENGINE INLET) - CONDITION

CABIN ENTRY



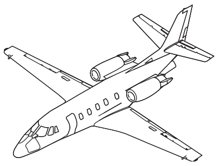
15. CABIN ENTRY - CHECK



15.a. DORSAL FIN AIR INLET - CLEAR

15.b. SECONDARY CABIN DOOR SEAL
- CHECK FOR RIPS. TEARS AND FOLDING

CABIN INSPECTION



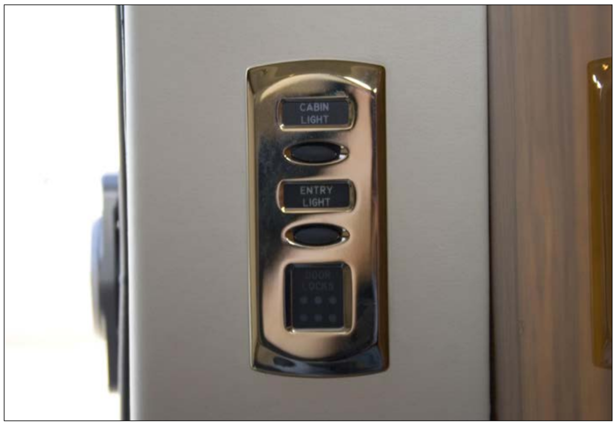
1. **LAVATORY DOORS -** STOWED OPEN

2. **EMERGENCY EXIT -** SECURE;
HANDLE LOCK PIN - REMOVE

3. **WATER BARRIER** - STOWED (IF REQUIRED)

4. **PORTABLE FIRE EXTINGUISHERS**
- SERVICED AND SECURE

5. **PASSENGER SEATS -** UPRIGHT, OUTBOARD AND POSITION AFT OR FORWARD AS REQUIRED TO CLEAR EXIT DOORS

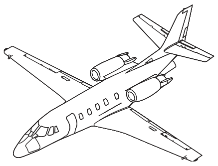


6. **DOOR ENTRY LIGHTS -** OFF

XLS+ WALKAROUND

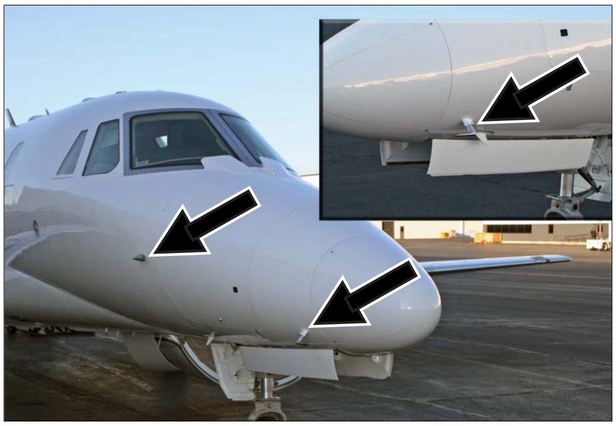
The following section is a pictorial walkaround. Each item listed in the exterior power-off preflight inspection is displayed. The general photographs contain circled numbers that correspond to specific steps displayed on the subsequent pages.

PREFLIGHT INSPECTION



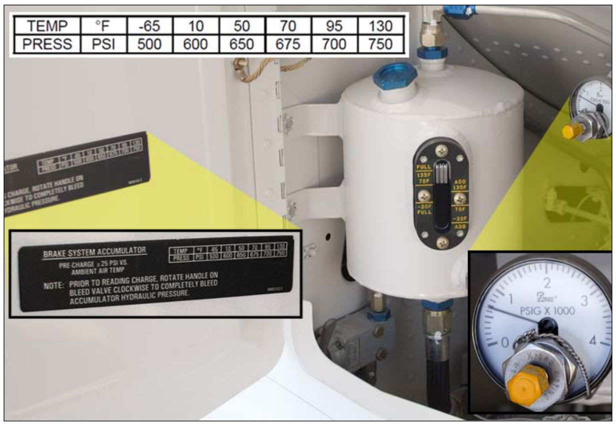
1. **BATTERY** - CONNECTED

2. **ENGINE COVERS (4) -** REMOVED



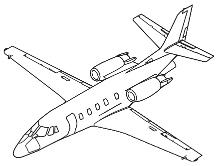
3. **PITOT COVERS (3) -** REMOVED

4. **POWER BRAKE ACCUMULATOR**
   - BLEED TO PRE-CHARGE



5. **BRAKE SYSTEM ACCUMULATOR CHARGE**
   - PER PLACARD

6. **POWER BRAKE RESERVOIR SIGHT GAGE -** CHECK



7. **BRAKE SYSTEM ACCESS DOOR -** SECURE

8. **APU INLETS, EXHAUST AND OIL LEVEL (IF STARTING APU BEFORE EXTERIOR INSPECTION)** - CHECK

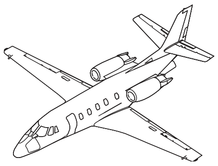
8. **APU INLETS, EXHAUST AND OIL LEVEL (IF STARTING APU BEFORE EXTERIOR INSPECTION)** - CHECK

8. **APU INLETS, EXHAUST AND OIL LEVEL (IF STARTING APU BEFORE EXTERIOR INSPECTION)** - CHECK

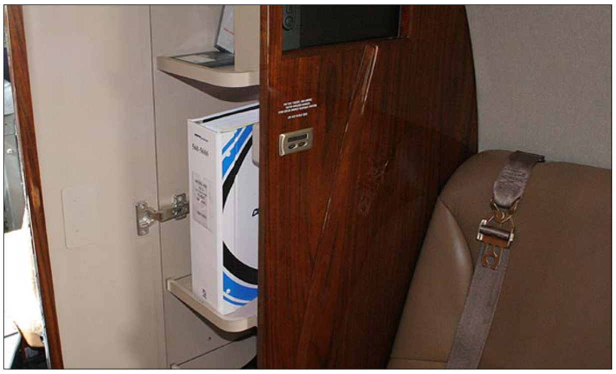
PRELIMINARY COCKPIT INSPECTION


20
7


9
14
10
22
8



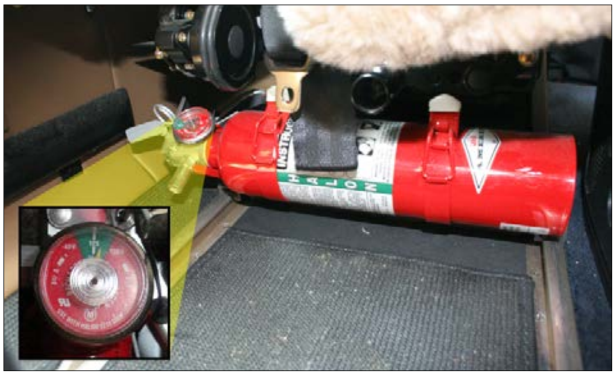
1. **DOCUMENTS** - CHECK ABOARD
   a. TO BE DISPLAYED IN THE AIRPLANE AT ALL TIMES:
      1) AIRWORTHINESS AND REGISTRATION CERTIFICATES
      2) RADIO STATION LICENSE(S) (IF REQUIRED).



1. **DOCUMENTS** - CHECK ABOARD
   b. TO BE CARRIED IN THE AIRPLANE AT ALL TIMES:
      1) FAA APPROVED AIRPLANE FLIGHT MANUAL
      2) COLLINS PRO LINE 21 AVIONICS SYSTEM OPERATOR'S GUIDE

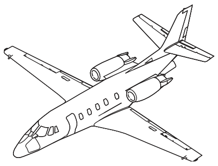
1. **DOCUMENTS** - CHECK ABOARD
   b. TO BE CARRIED IN THE AIRPLANE AT ALL TIMES:
      3) COLLINS FMS-3000 FLIGHT MANAGEMENT SYSTEM OPERATOR'S GUIDE
      4) OTHER APPLICABLE PILOT'S MANUALS AS REQUIRED IN SECTION II, OPERATING LIMITATIONS OR APPLICABLE AFM SUPPLEMENT

2. **FLASHLIGHT -** ABOARD



3. **PORTABLE FIRE EXTINGUISHER -** SERVICED AND SECURE (UNDER COPILOT'S SEAT)

4. **MICROPHONES, HEADSETS, OXYGEN MASKS AND SMOKE GOGGLES -** ON BOARD AND PROPERLY STOWED



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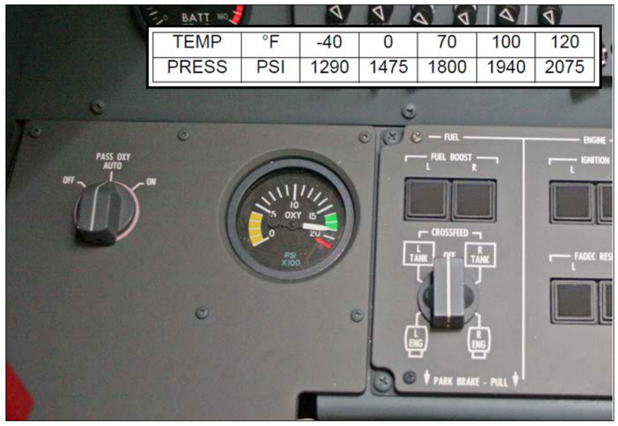
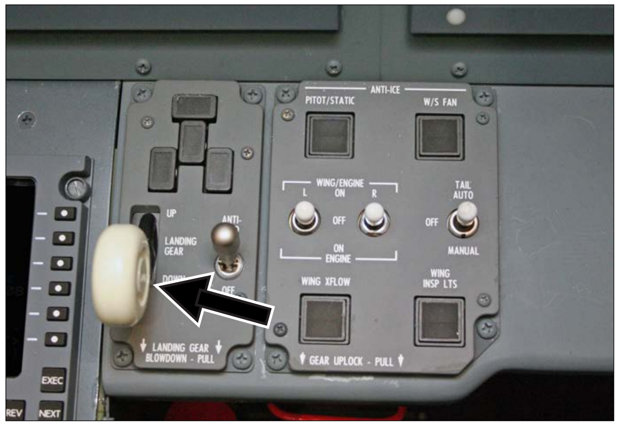
4. **MICROPHONES, HEADSETS, OXYGEN MASKS AND SMOKE GOGGLES -** ON BOARD AND PROPERLY STOWED

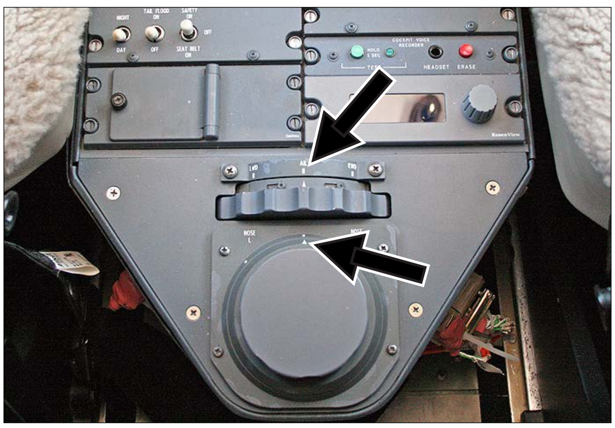


5. **OXYGEN QUANTITY -** CHECK

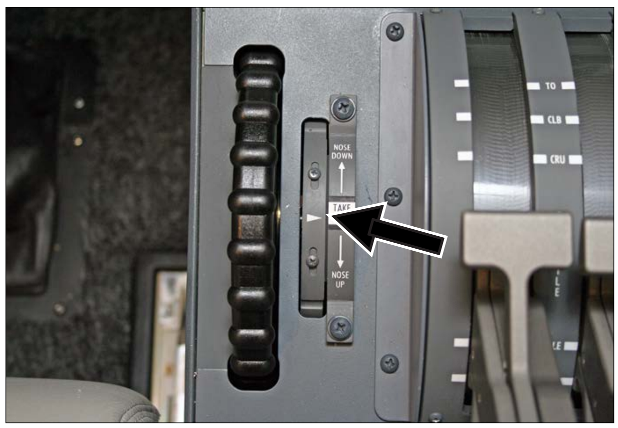
6. **CONTROL LOCK -** UNLOCKED



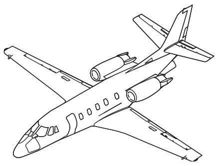
7. **LANDING GEAR HANDLE -** DOWN



8. **RUDDER AND AILERON TRIM -** NEUTRAL



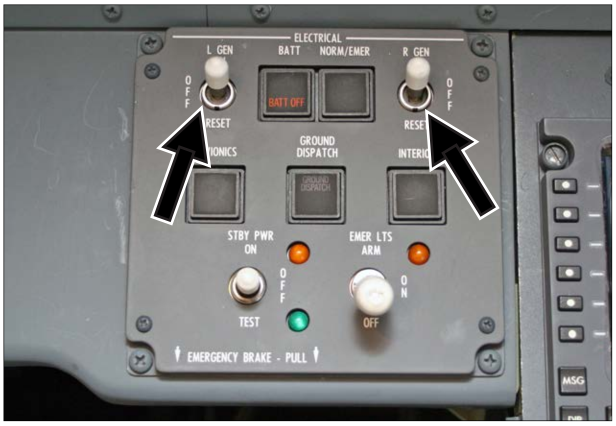
9. **ELEVATOR TRIM -** SET FOR TAKE OFF (WITHIN TO BAND)



10. FLAP HANDLE - AGREES WITH FLAP POSITION

11. CIRCUIT BREAKERS - IN

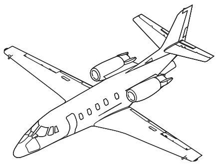
11. CIRCUIT BREAKERS - IN



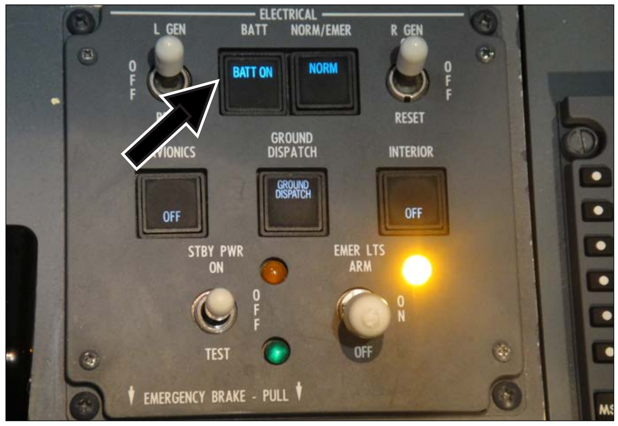
12. GENERATOR SWITCHES - ON
(OFF, IF EPU START)

13. ALL OTHER SWITCHES AND BUTTONS
- OFF/NORM/AUTO

13. ALL OTHER SWITCHES AND BUTTONS
- OFF/NORM/AUTO



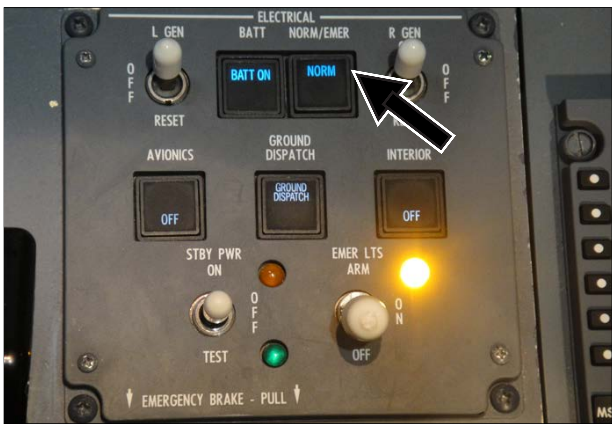
14. THROTTLES - CUT OFF



15. BATT BUTTON - BATT ON

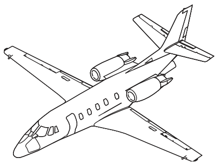
16. NORM/EMER BUTTON - NORM (24 VOLTS MINIMUM)

17. NORM/EMER BUTTON - EMER
CHECK STANDBY ENGINE DISPLAY, STANDBY RADIO CONTROL HEAD, STANDBY HSI, PILOT AND COPILOT AUDIO PANELS, CDUU1 AND LANDING GEAR DOWNLOCK ANNUNCIATORS ILLUMINATED.



18. NORM/EMER BUTTON - NORM

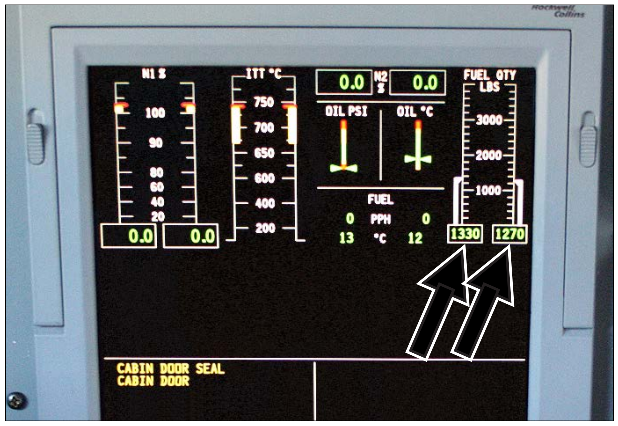
19. PARK BRAKE - SET



20. TAIL DEICE SYSTEM
a. TAIL SWITCH - AUTO

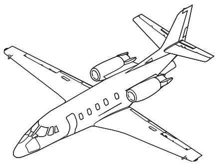
20. TAIL DEICE SYSTEM
b. CAS MESSAGES - AMBER TAIL DE-ICE FAIL L, THEN R (ALLOW 12 SECONDS FOR BOTH TO TEST)

20. TAIL DEICE SYSTEM
c. TAIL SWITCH - OFF

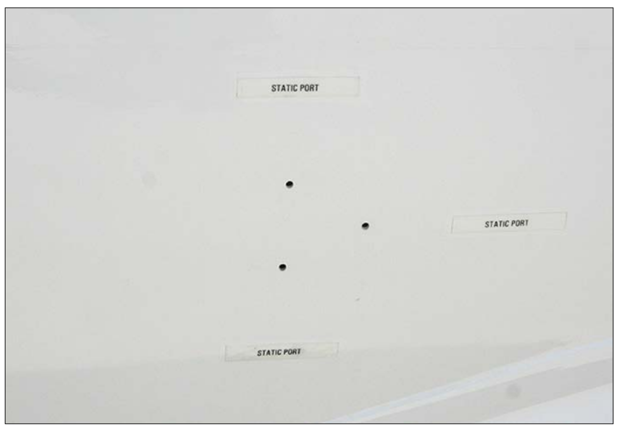


21. FUEL QUANTITY AND BALANCE - CHECK

22. EXTERIOR, INTERIOR AND EMERGENCY LIGHT SWITCHES - ON/CHECK/OFF OR AS REQUIRED

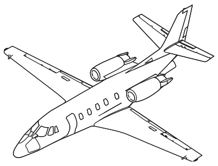


HOT ITEMS/LIGHTS



a. LEFT, RIGHT AND STANDBY STATIC PORTS
- CLEAR/WARM

a. LEFT, RIGHT AND STANDBY STATIC PORTS
- CLEAR/WARM

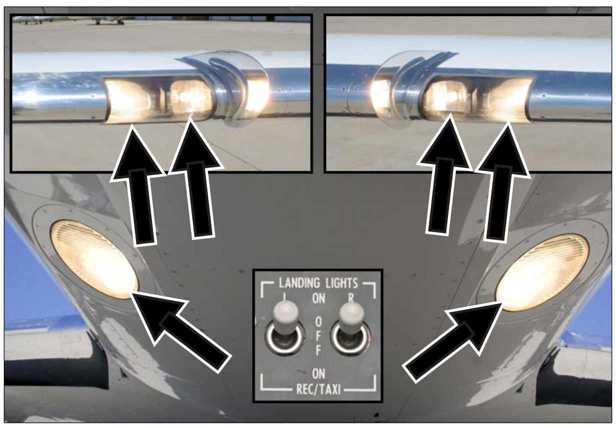


a. LEFT, RIGHT AND STANDBY STATIC PORTS
- CLEAR/WARM

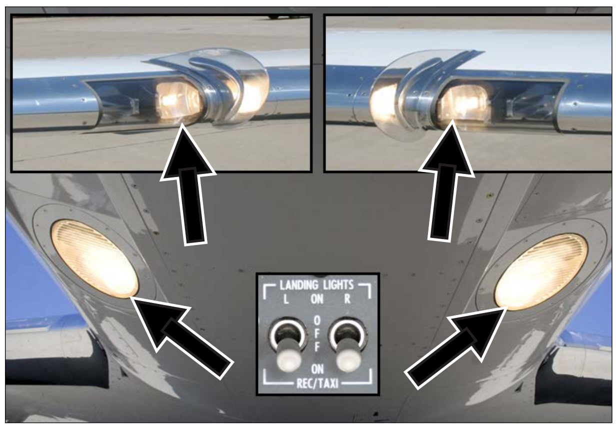
b. LEFT, RIGHT AND STANDBY PITOT TUBES
- CLEAR/HOT

b. LEFT, RIGHT AND STANDBY PITOT TUBES
- CLEAR/HOT

b. LEFT, RIGHT AND STANDBY PITOT TUBES
- CLEAR/HOT

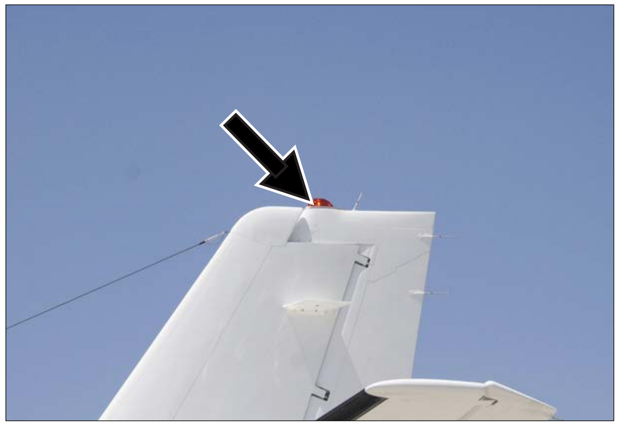


c. LANDING LIGHTS - ALL ON

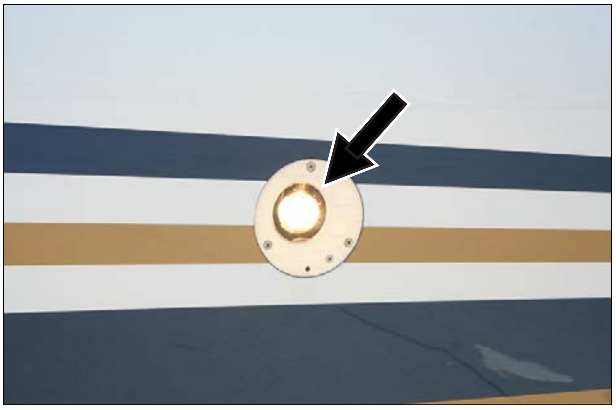


d. RECOGNITION/TAXI LIGHTS - ALL ON

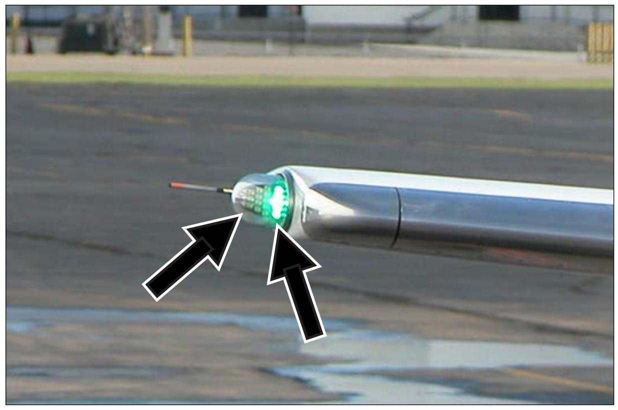
e. **ANGLE-OF-ATTACK VANE -** ROTATES/HOT



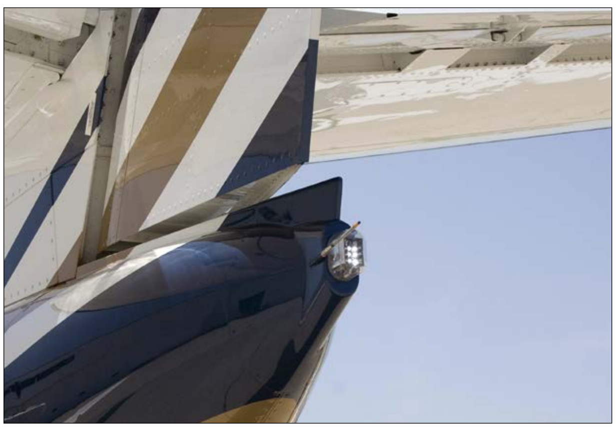
f. **GROUND RECOGNITION LIGHT -** ON/FLASHING



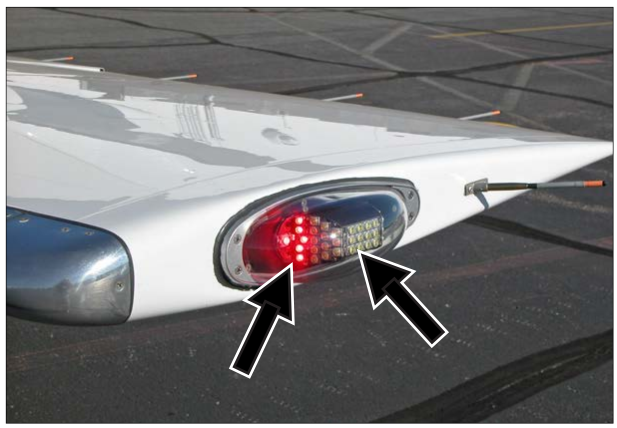
g. **RIGHT WING INSPECTION, NAVIGATION, AND ANTI-COLLISION LIGHTS -** ON



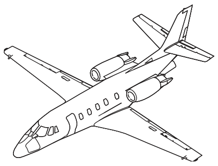
g. **RIGHT WING INSPECTION, NAVIGATION, AND ANTI-COLLISION LIGHTS -** ON



h. **TAIL NAVIGATION LIGHT -** ON



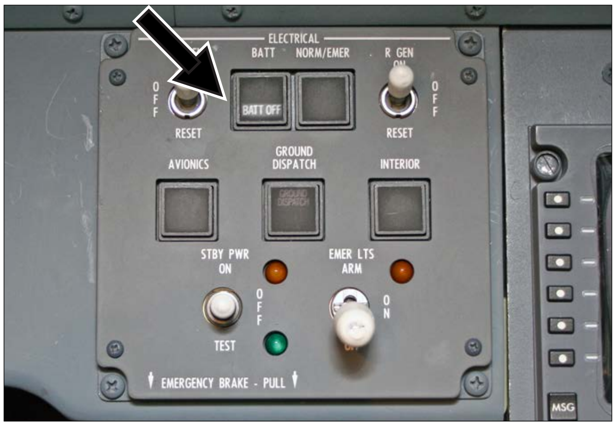
i. **LEFT NAVIGATION, ANTI-COLLISION, AND WING INSPECTION LIGHTS -** ON



i. **LEFT NAVIGATION, ANTI-COLLISION, AND WING INSPECTION LIGHTS -** ON

j. **HOT ITEMS/LIGHTS AND BATT BUTTON** - OFF/BATT OFF

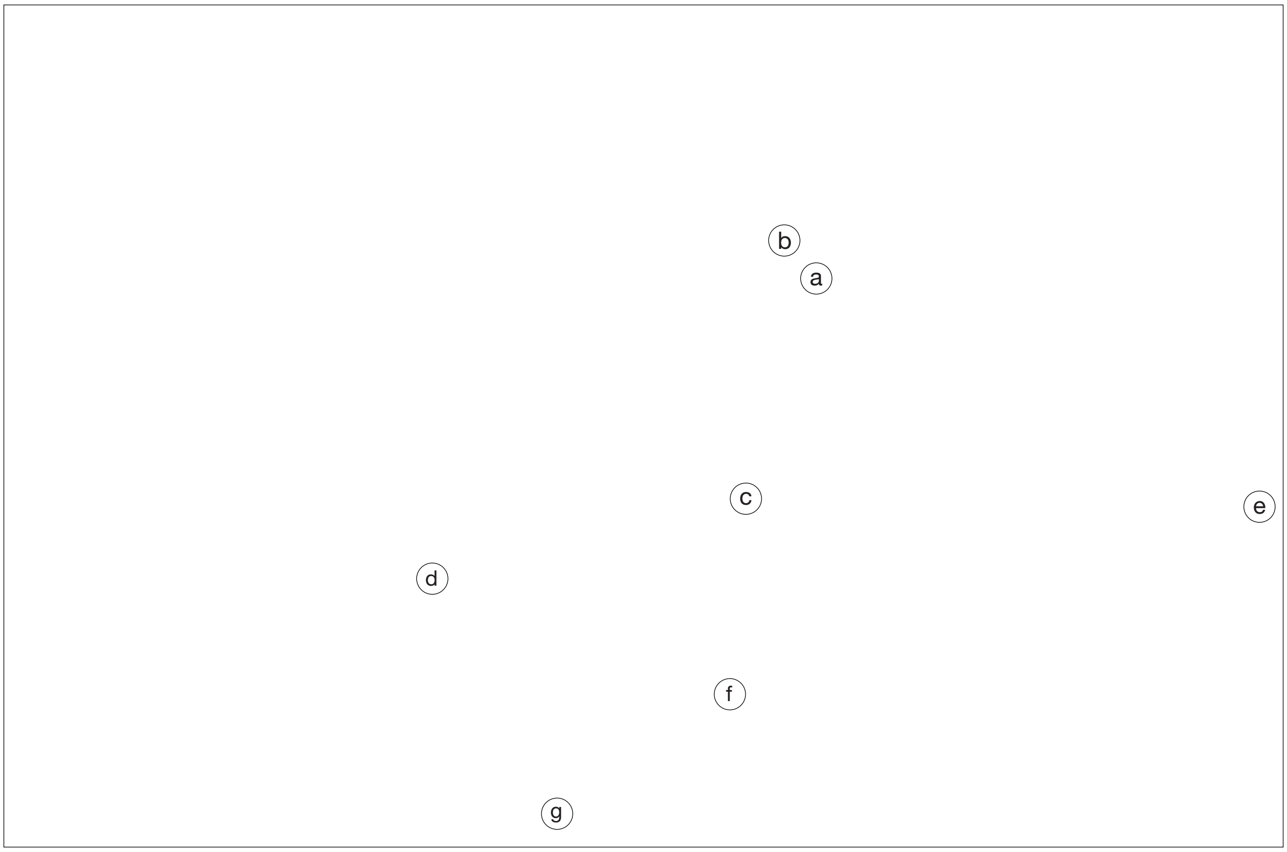
j. **HOT ITEMS/LIGHTS AND BATT BUTTON** - OFF/BATT OFF

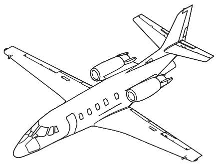


j. **HOT ITEMS/LIGHTS AND BATT BUTTON** - OFF/BATT OFF

XLS+ WALKAROUND

LEFT NOSE AND FUSELAGE LEFT SIDE

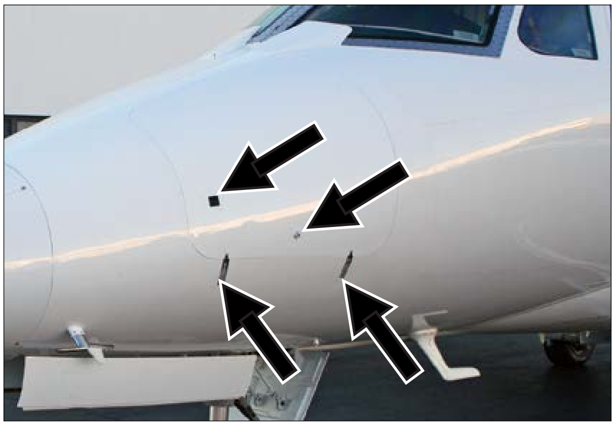




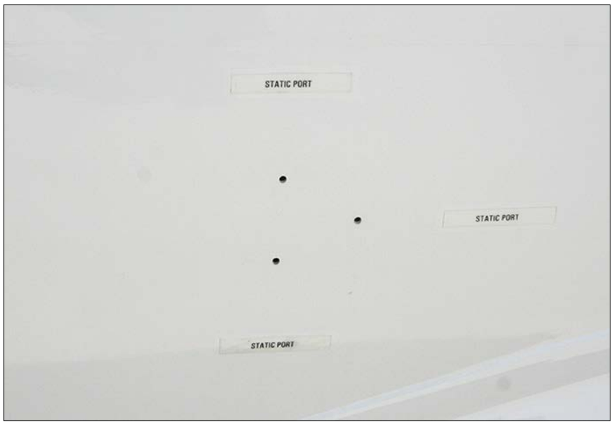
a. ANTISKID FAULT DISPLAY UNIT (BITE INDICATOR) - CHECK/RESET IF REQUIRED

b. GEAR AND BRAKE PNEUMATIC PRESSURE GAGE - PER PLACARD

c. STATIC DRAIN - CLOSED

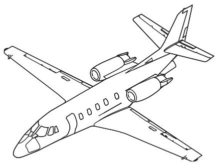


d. ACCESSORY DOOR - SECURE/LOCKED

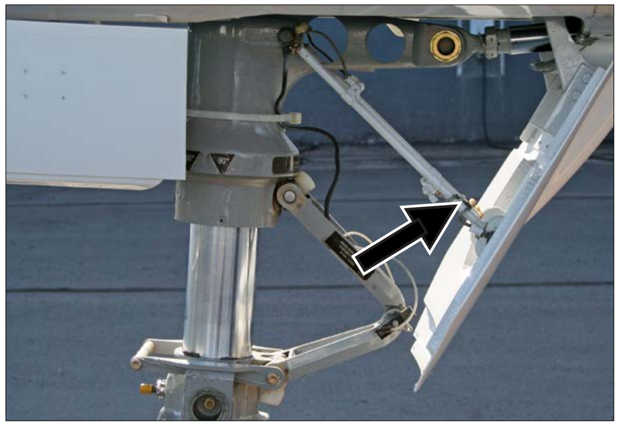


e. STATIC PORTS AND SURROUNDING FUSELAGE SKIN - CLEAN/NO DAMAGE

f. OVERBOARD VENT LINES - CLEAR



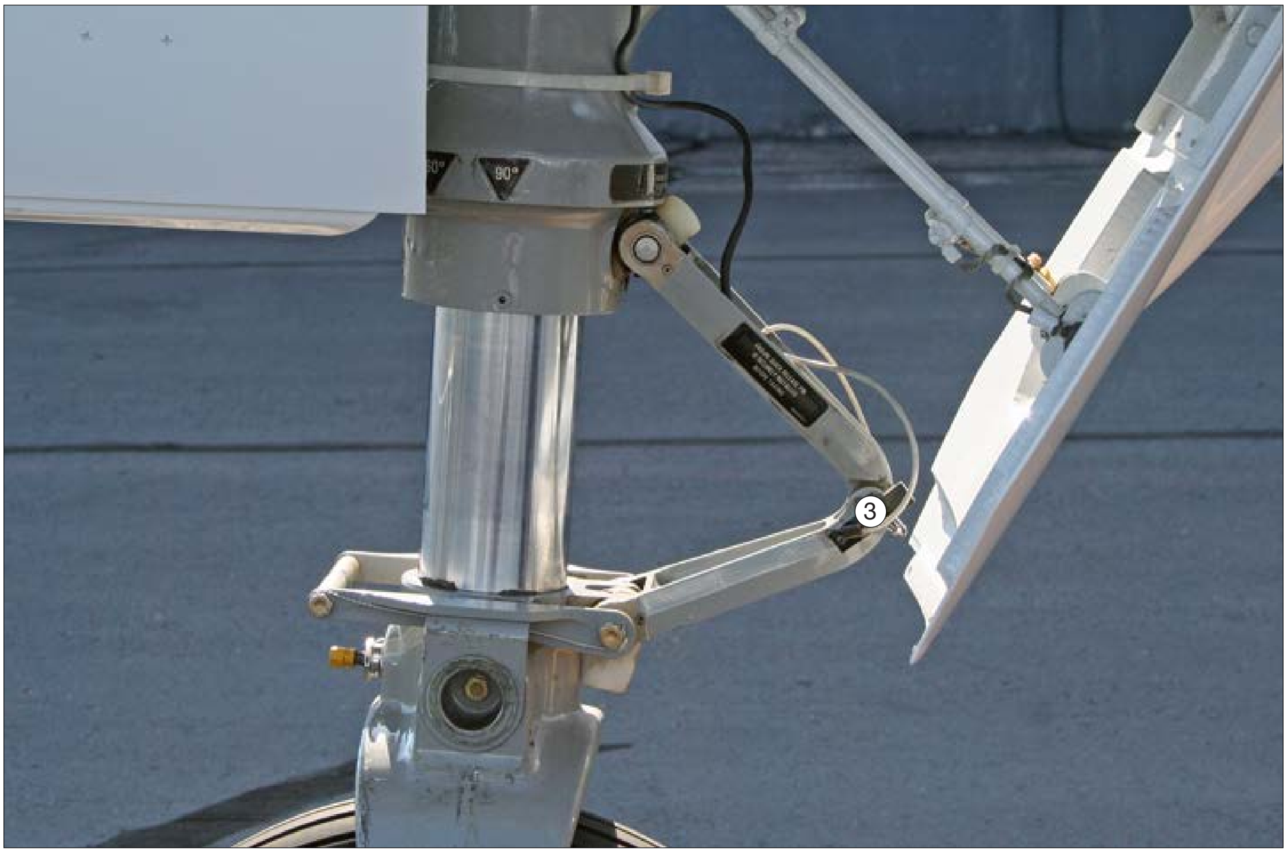
g. NOSE GEAR, DOORS, WHEEL AND TIRE - CONDITION

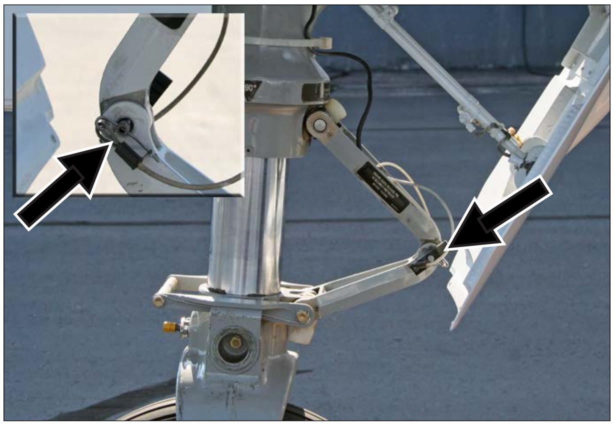


g. NOSE GEAR, DOORS, WHEEL AND TIRE - CONDITION

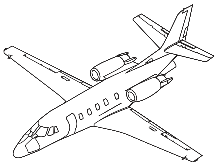
g. NOSE GEAR, DOORS, WHEEL AND TIRE - CONDITION

NOSE GEAR TORQUE LINKS

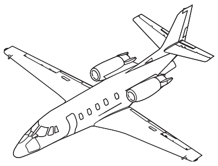




3. NOSE GEAR TORQUE LINKS
- CONNECTED, CHECK PIN INSTALLED



RIGHT NOSE AND FUSELAGE RIGHT SIDE



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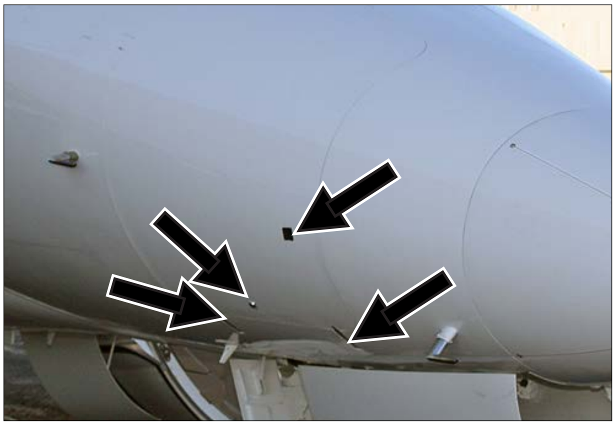
a. **OXYGEN BOTTLE SUPPLY LEVER -** ON

b. **STATIC DRAINS (2) -** CLOSED

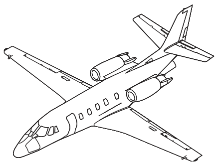
b. **STATIC DRAINS (2) -** CLOSED

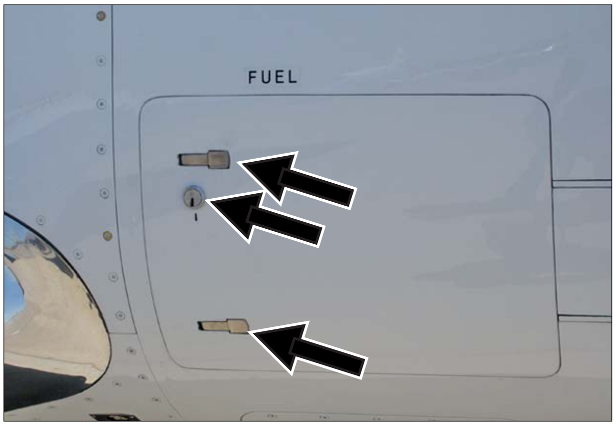


c. **ACCESSORY DOOR -** SECURE/LOCKED

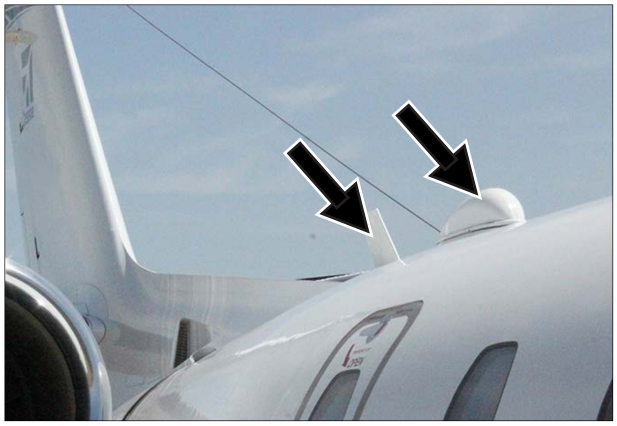
d. **OXYGEN BLOWOUT DISC -** GREEN

e. **STATIC PORTS AND SURROUNDING FUSELAGE SKIN** - CLEAN/NO DAMAGE

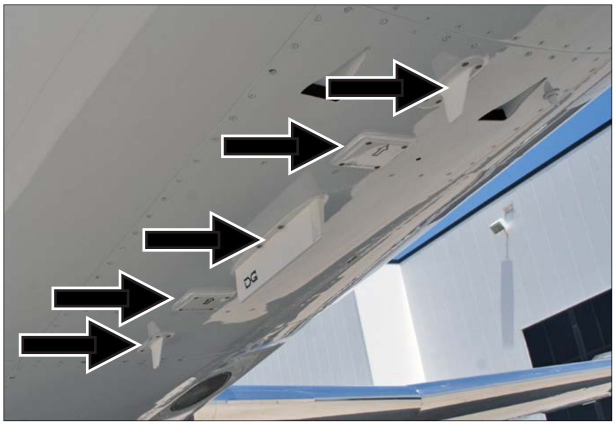




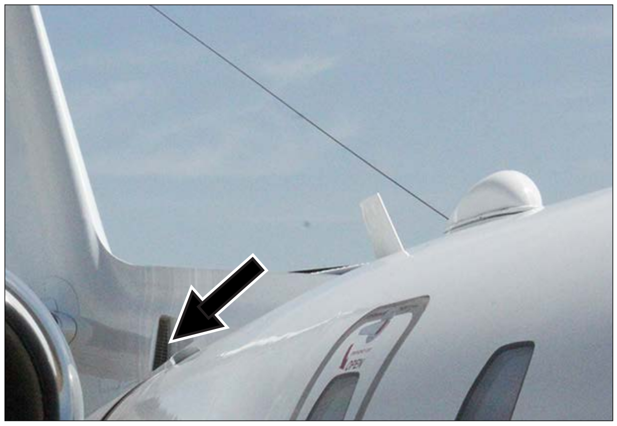
f. SINGLE POINT PRESSURE REFUEL DOOR - LATCHED/LOCKED



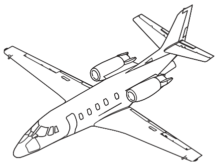
g. TOP AND BOTTOM ANTENNAS - CONDITION



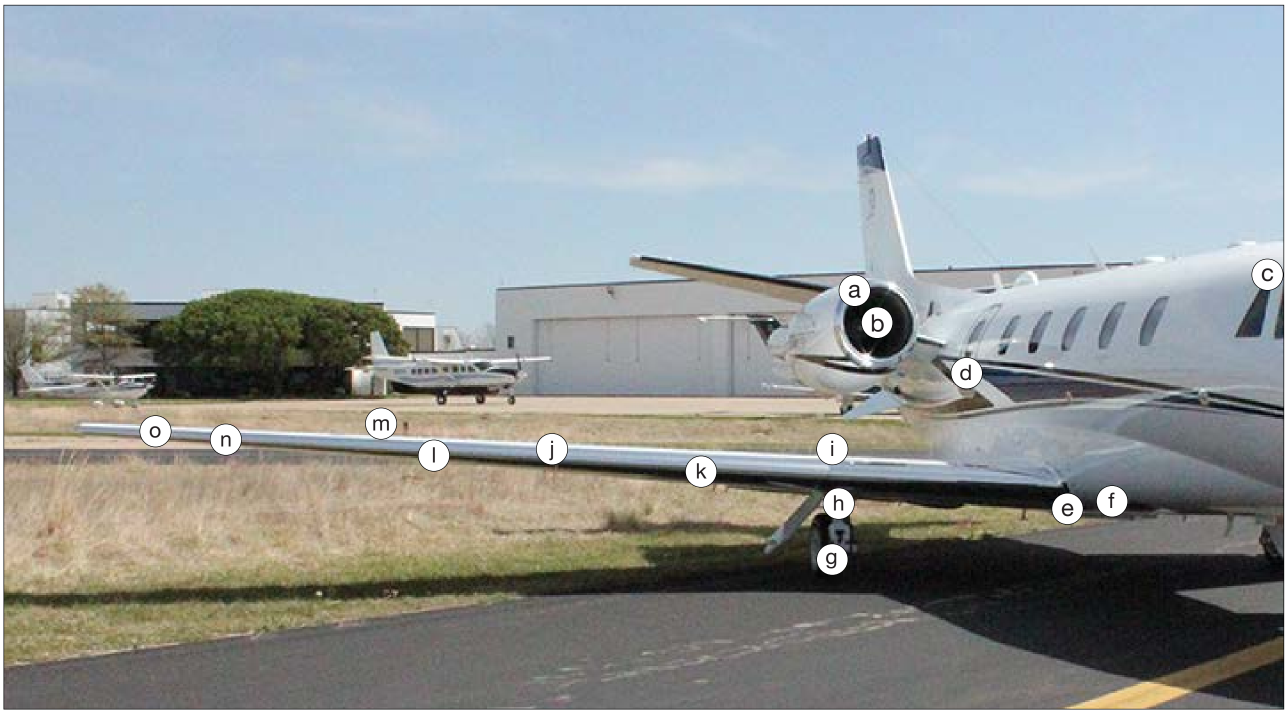
g. TOP AND BOTTOM ANTENNAS - CONDITION

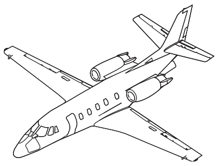


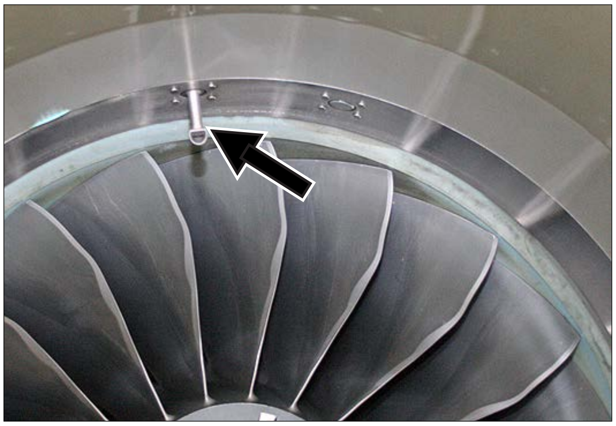
h. DORSAL FIN AIR INLET - CLEAR



RIGHT WING





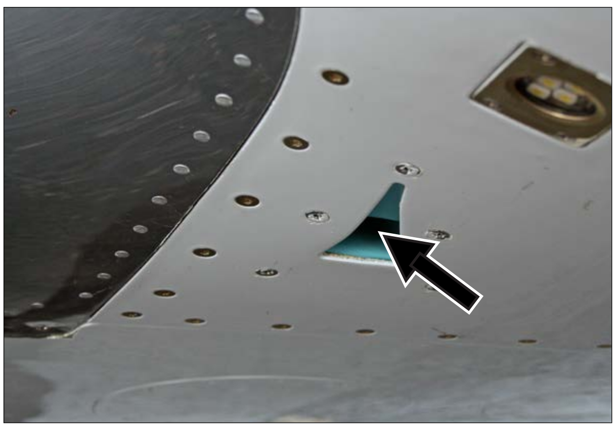


a. RAT PROBE (IN RIGHT ENGINE INLET) - CONDITION

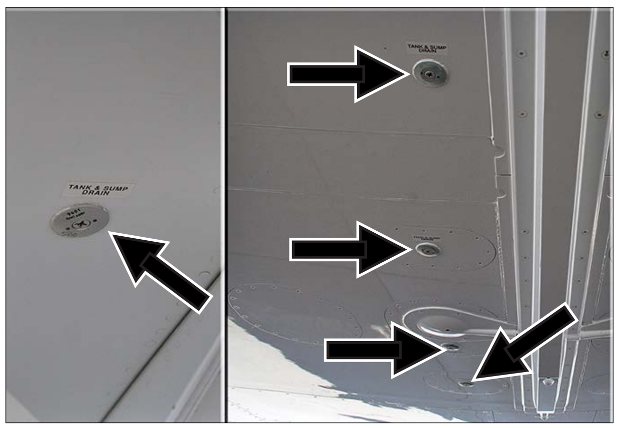
b. ENGINE FAN DUCT AND FAN - CONDITION

c. WING INSPECTION LIGHT - CONDITION

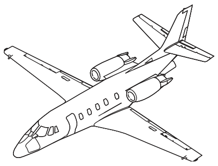
d. EMERGENCY EXIT - CONDITION



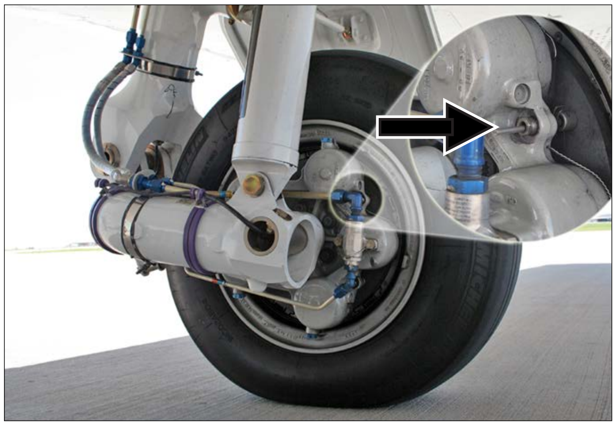
e. ANTI-ICE BLEED AIR COOLING AIR INLET - CLEAR



f. FUEL TANK AND SUMP DRAINS (5) - DRAIN/CHECK



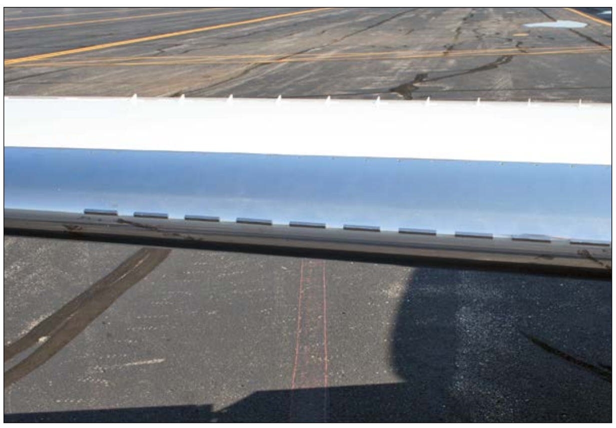
g. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE - CONDITION



g. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE - CONDITION

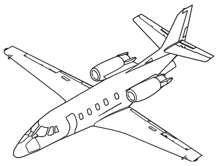
h. WHEEL WELL - CONDITION/NO LEAKS

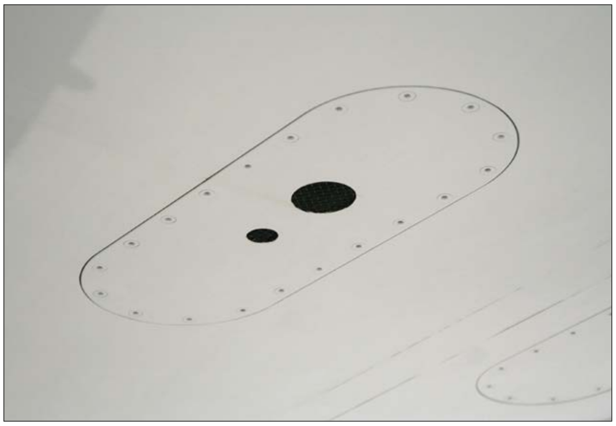
i. VORTEX GENERATORS (26) - CHECK



j. BOUNDARY LAYER ENERGIZERS (11)- CHECK

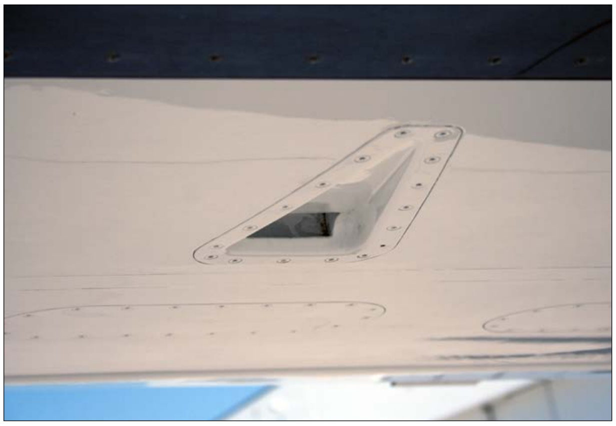
k. HEATED LEADING EDGE - CONDITION/VENT CLEAR



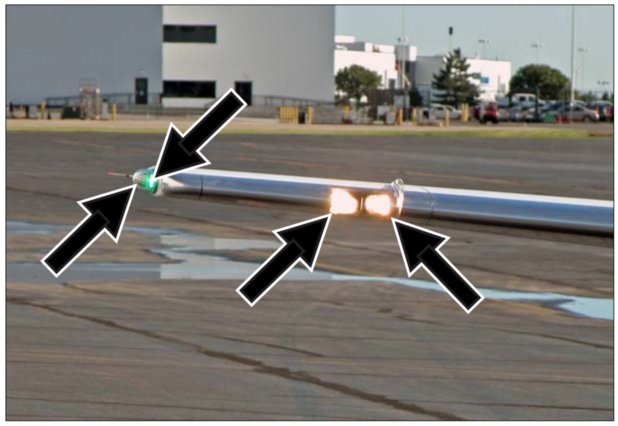


l. FUEL TANK RELIEF VALVES - CONDITION/NO LEAKS

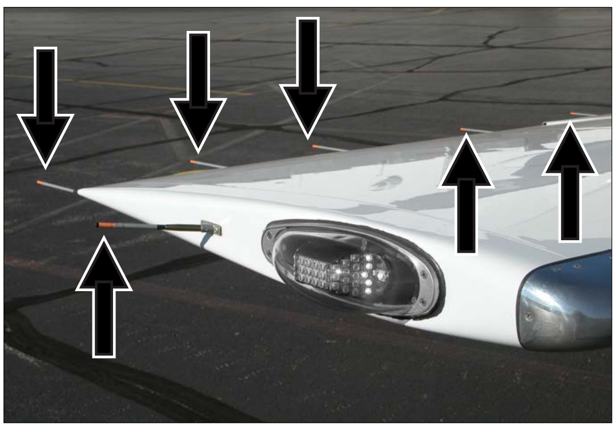
m. FUEL FILLER CAP - SECURE



n. FUEL TANK VENT - CLEAR

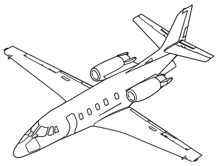


o. RECOGNITION, LANDING, NAVIGATION, AND STROBE LIGHTS - CONDITION



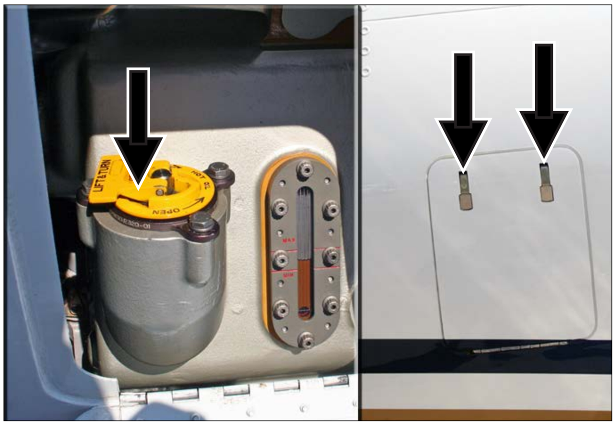
p. STATIC WICKS (6) - CHECK

q. AILERON, SPEEDBRAKES AND FLAPS - CONDITION

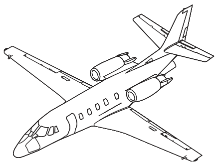


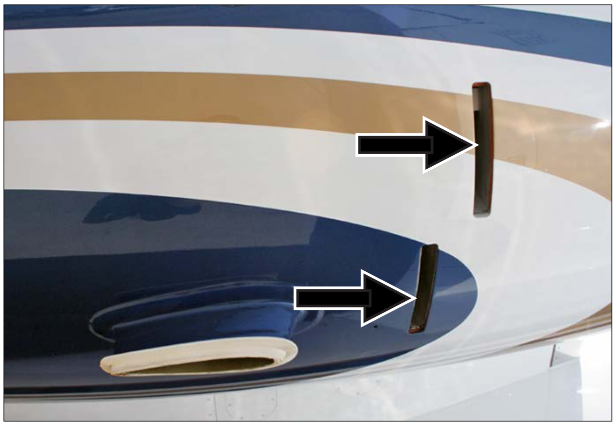
RIGHT NACELLE/PYLON

a. OIL LEVEL - CHECK

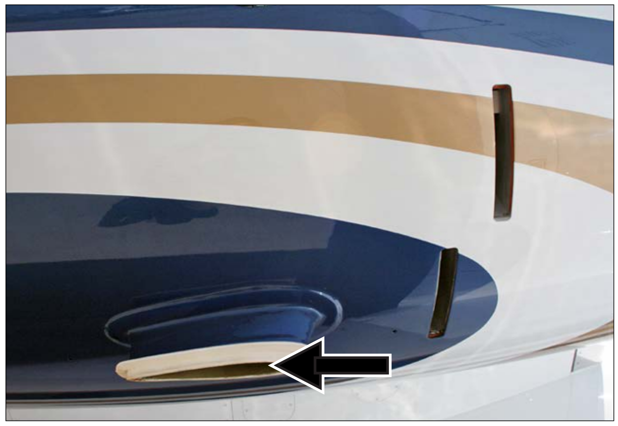


b. FILLER CAP AND ACCESS DOOR - SECURE

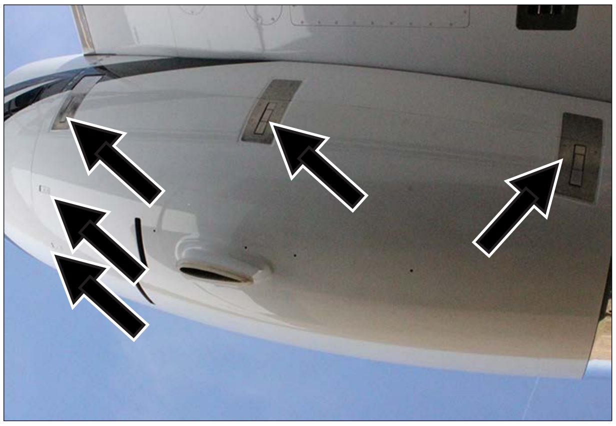




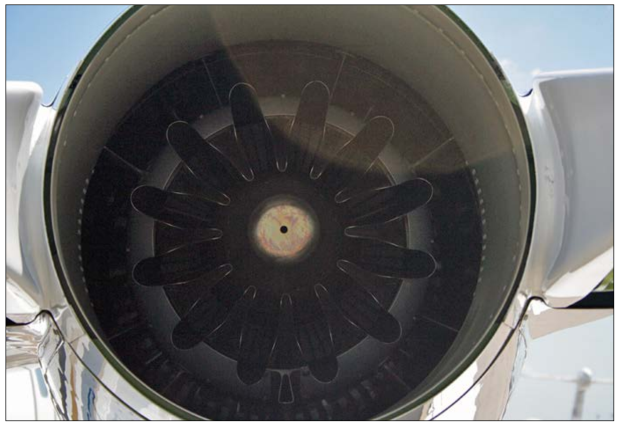
c. GENERATOR AND ALTERNATOR COOLING AIR EXHAUST - CLEAR



d. ENGINE FUEL DRAIN MAST - CLEAR



e. COWLING (5 LATCHES) - SECURE

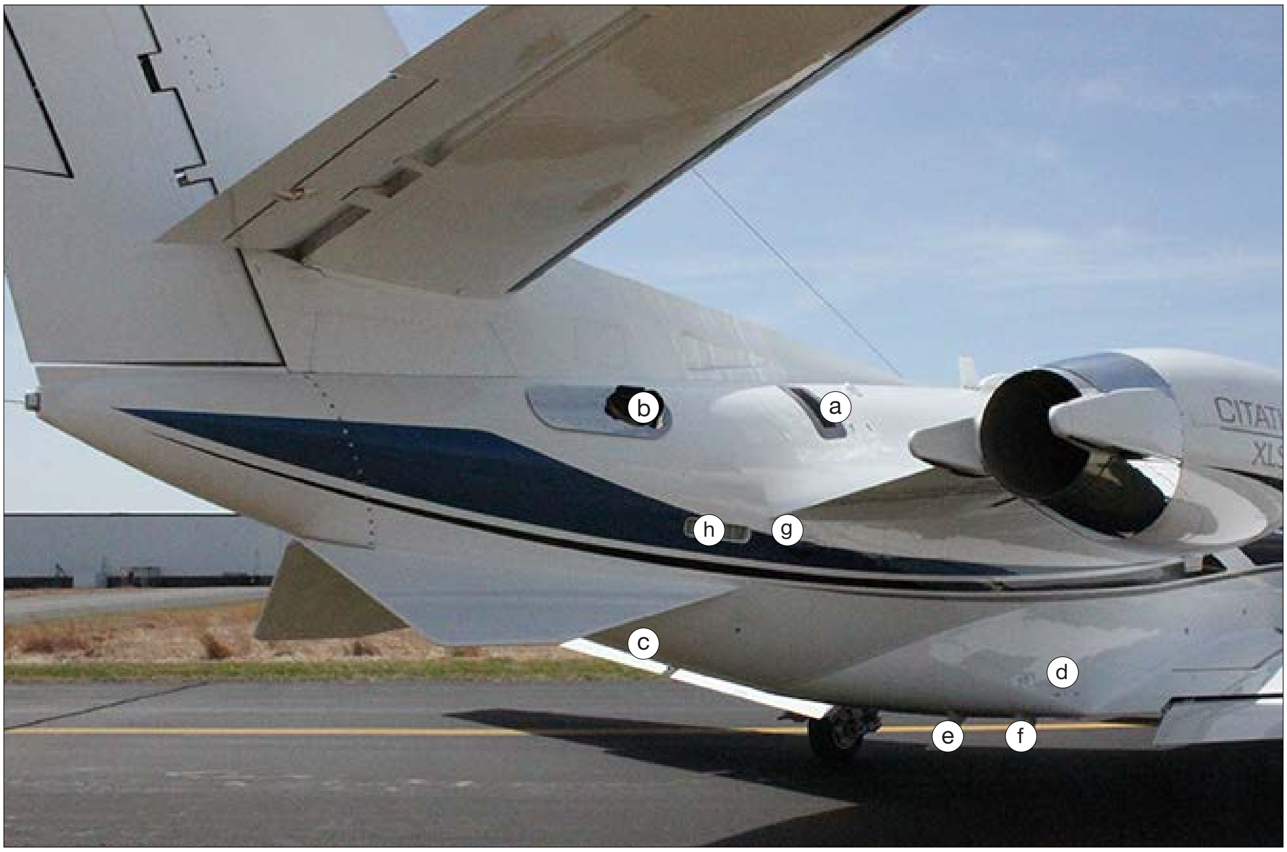


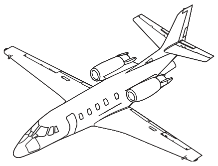
f. ENGINE EXHAUST AND BYPASS DUCT - CONDITION/CLEAR

g. THRUST REVERSER BUCKET - CONDITION/STOWED

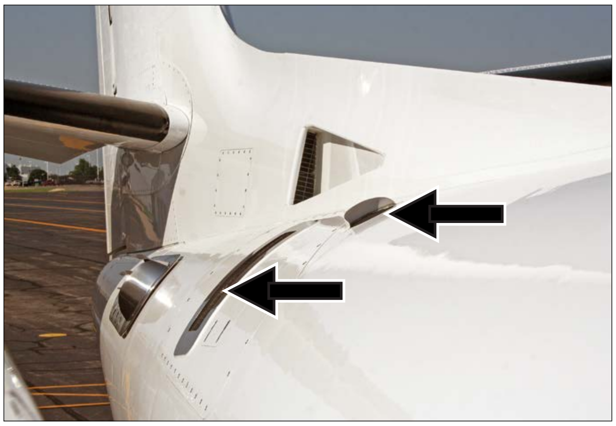
h. PYLON PRECOOLER EXHAUST DUCT - CLEAR

RIGHT AFT FUSELAGE





FlightSafety international



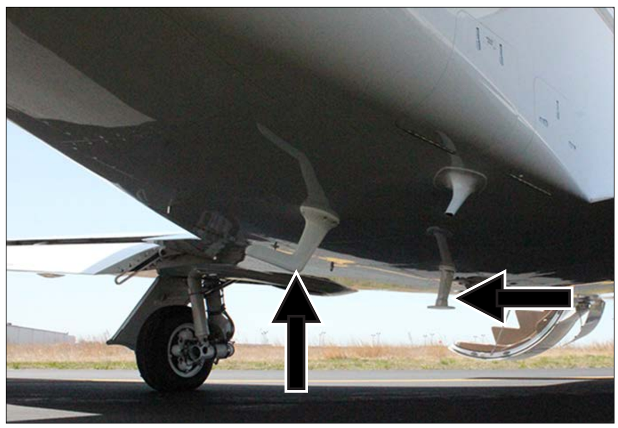
a. APU ENGINE AND GENERATOR COOLING INLET - CLEAR

b. APU EXHAUST - CLEAR

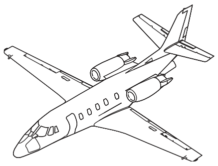
c. APU DRAIN - CLEAR

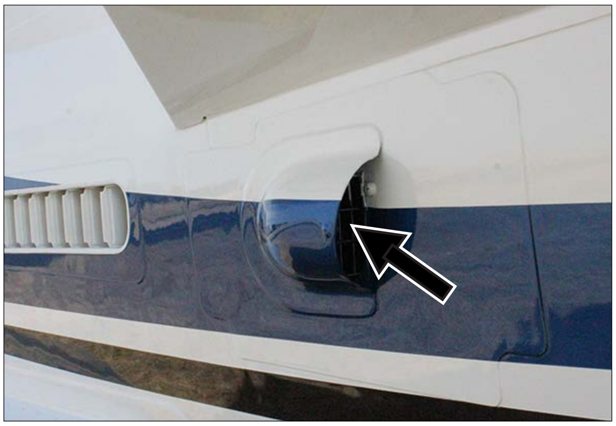
d. HYDRAULIC AND TOILET SERVICE DOORS - SECURE

e. HYDRAULIC DRAIN MAST - NO LEAKS

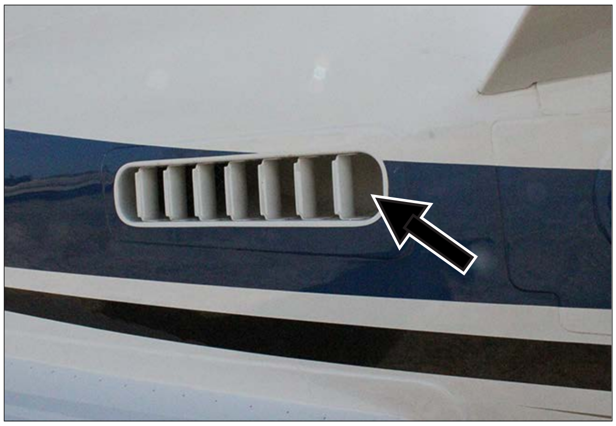


f. LOWER ANTENNAS - SECURE

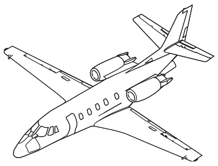




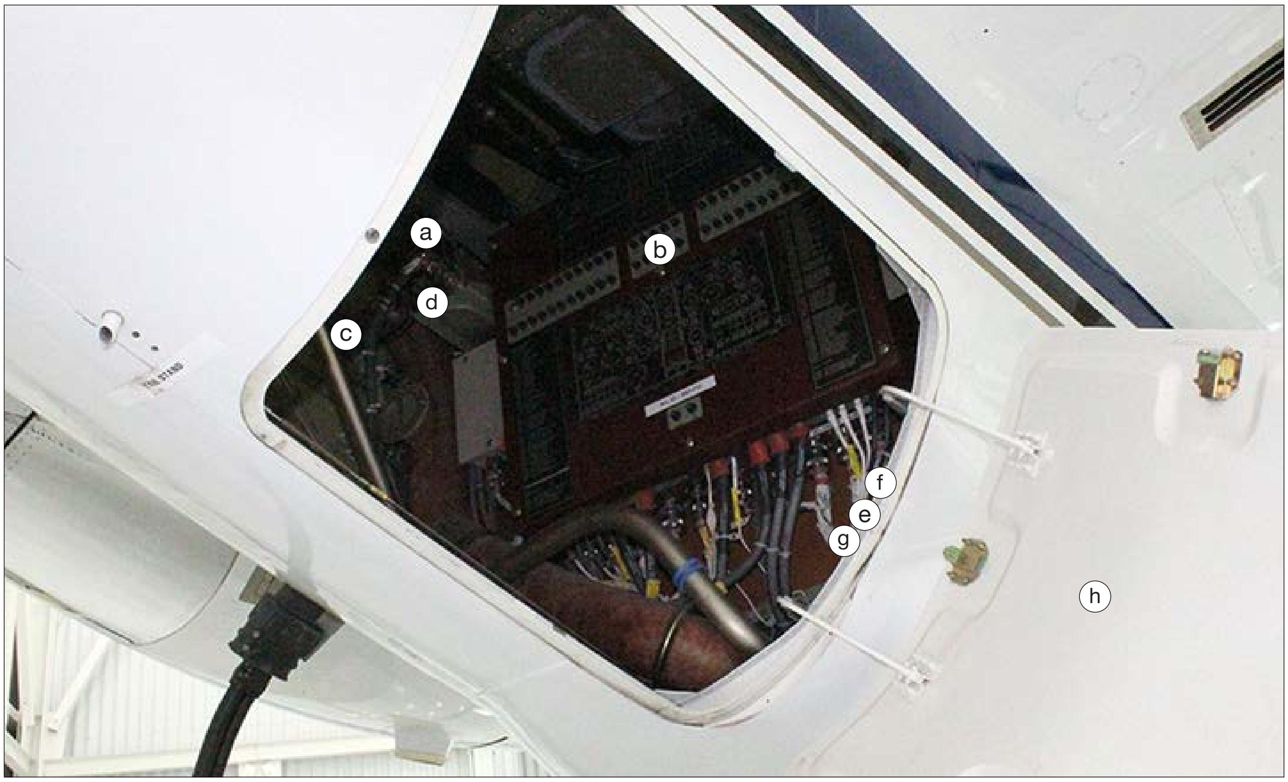
g. TAILCONE POSITIVE PRESSURE INLET - CLEAR

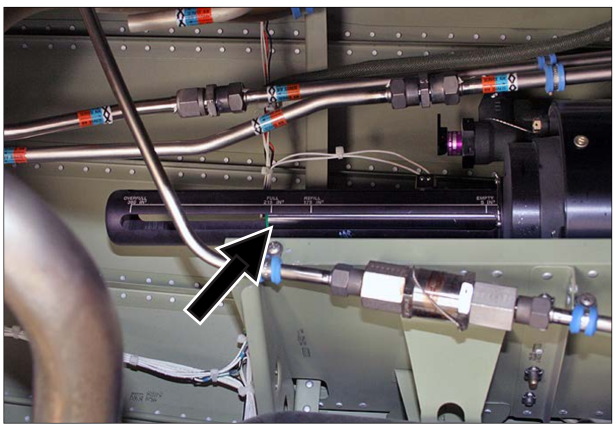


h. ACM EXHAUST - CLEAR

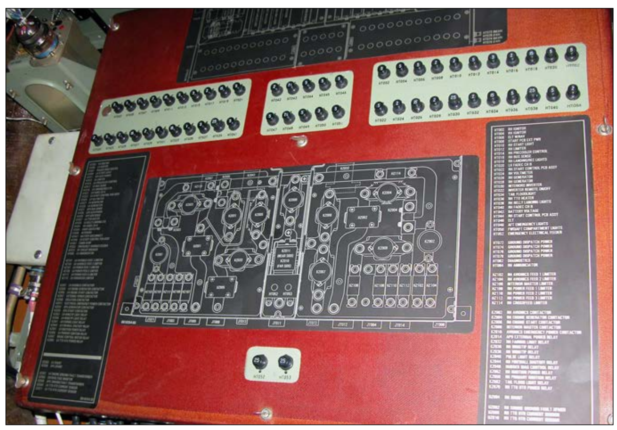


TAILCONE COMPARTMENT

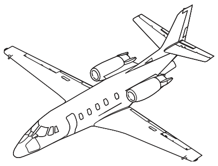




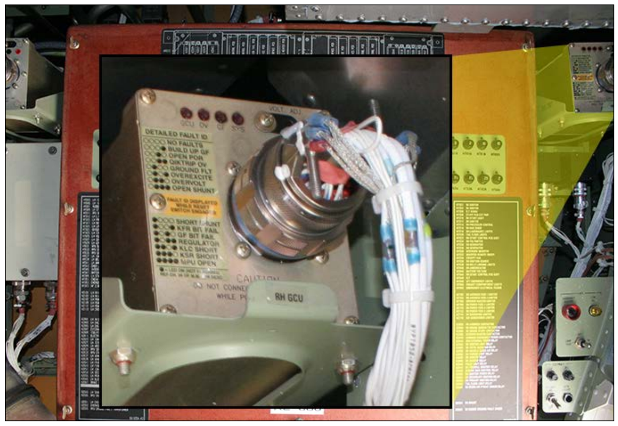
a. HYDRAULIC FUILD QUANTITY - CHECK



b. AFT JUNCTION BOX CIRCUIT BREAKERS - IN

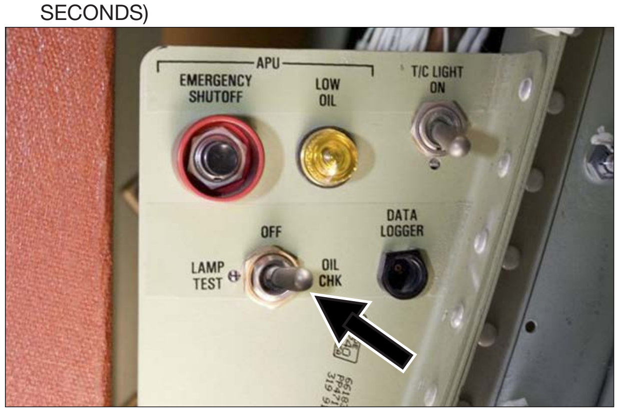


c. **EXTERNAL POWER RECEPTACLE CIRCUIT BREAKER** - IN

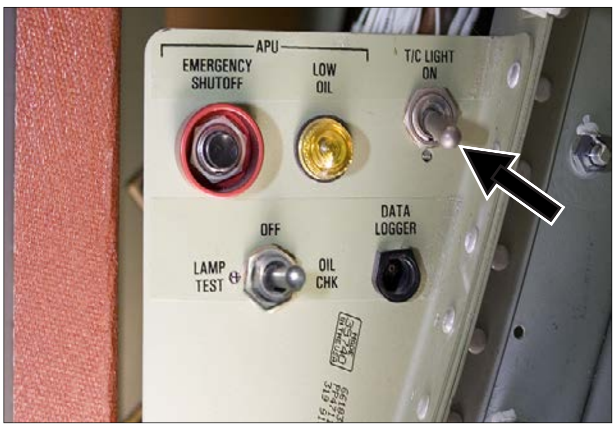


d. **LH AND RH GCU -** CHECK AND RESET IF REQUIRED (GEN SWITCH RESET THREE TIMES WITHIN THREE SECONDS)

e. **APU SERVICE PANEL SWITCH -** LAMP TEST

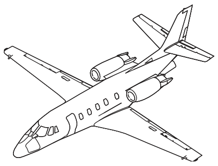


f. **APU SERVICE PANEL SWITCH** - OIL CHK
(1) IF THE AMBER LOW OIL LIGHT IS ILLUMINATED, THE APU MAY BE OPERATED FOR A MAXIMUM OF 20 HOURS PRIOR TO SERVICING THE AIRPLANE



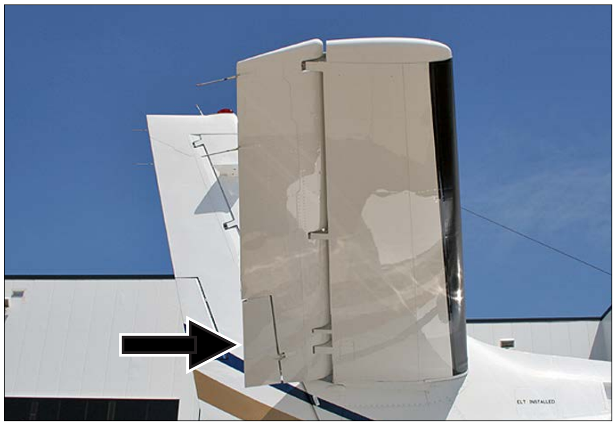
g. **TAILCONE LIGHT -** OFF

h. **TAILCONE ACCESS DOOR -** SECURE/LOCKED



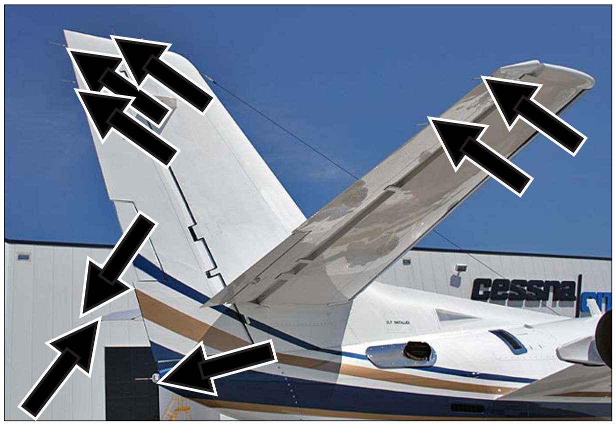
RIGHT EMPENNAGE

a. RIGHT HORIZONTAL STABILIZER DEICE BOOT - CONDITION

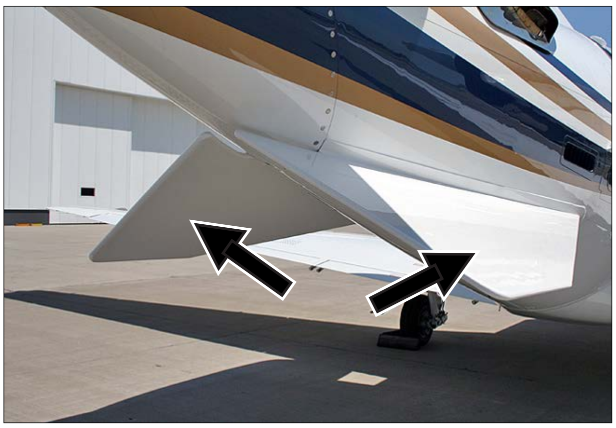


b. RIGHT ELEVATOR AND TRIM TAB - CONDITION

c. RUDDER AND TRIM TAB - CONDITION

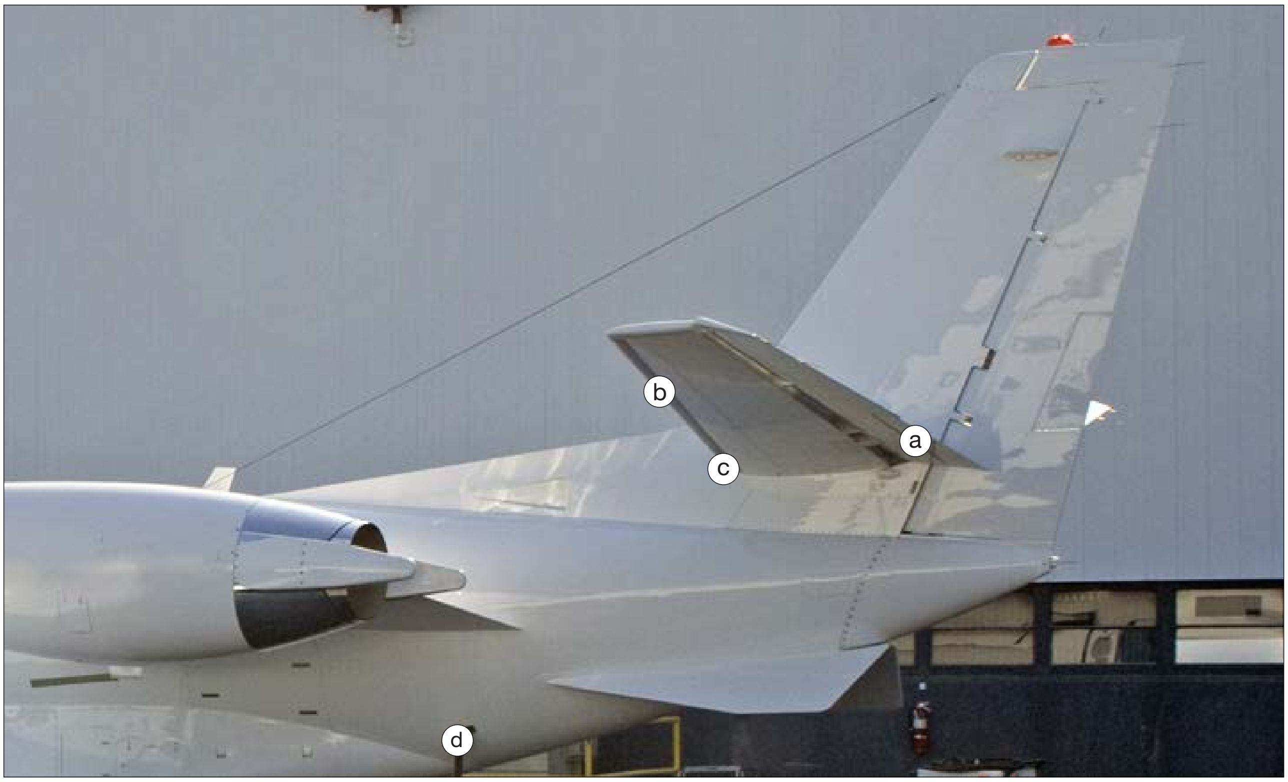


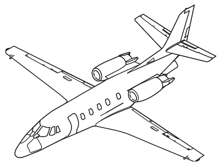
d. STATIC WICKS (RUDDER, VERTICAL STABILIZER AND BOTH ELEVATORS) (8) - CHECK

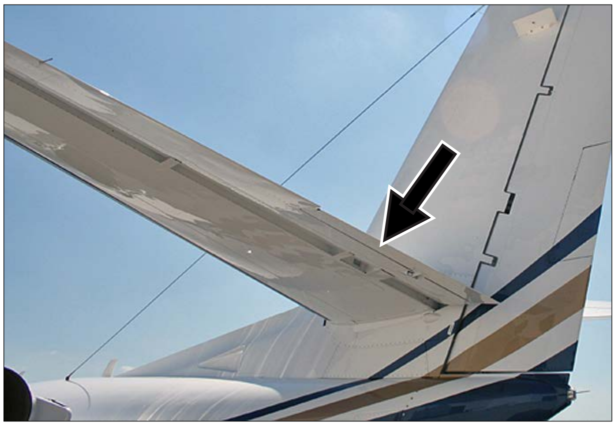


e. TAIL STRAKES - CONDITION

LEFT EMPENNAGE

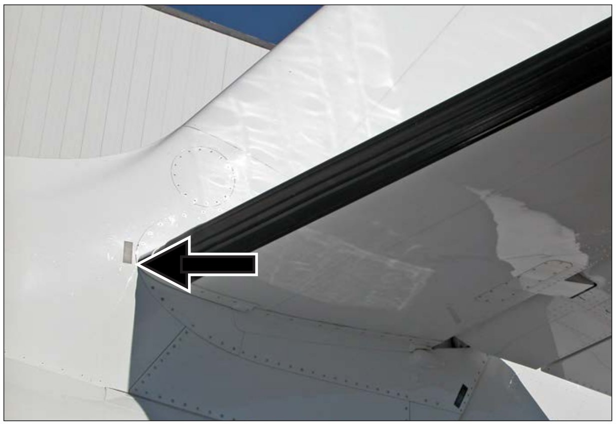






a. **LEFT ELEVATOR AND TRIM TAB -** CONDITION

b. **LEFT HORIZONTAL STABILIZER DEICE BOOT - **CONDITION



c. **HORIZONTAL STABILIZER POSITION INDEX - **CHECK; AGREES WITH FLAP POSITION

d. **EXTERNAL POWER SERVICE DOOR -** SECURE

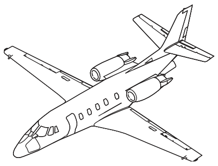
XLS+ WALKAROUND

BAGGAGE COMPARTMENT

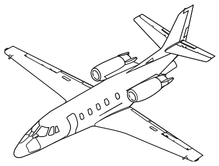
a. **BAGGAGE -** SECURE

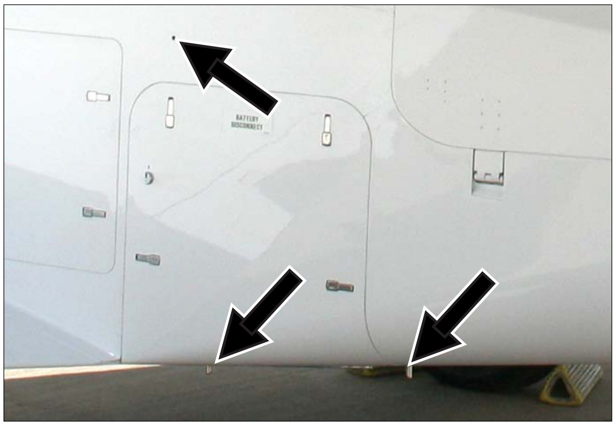
b. **BAGGAGE COMPARTMENT LIGHT -** OFF

c. **BAGGAGE COMPARTMENT ACCESS DOOR -** SECURE/LOCKED



LEFT AFT FUSELAGE



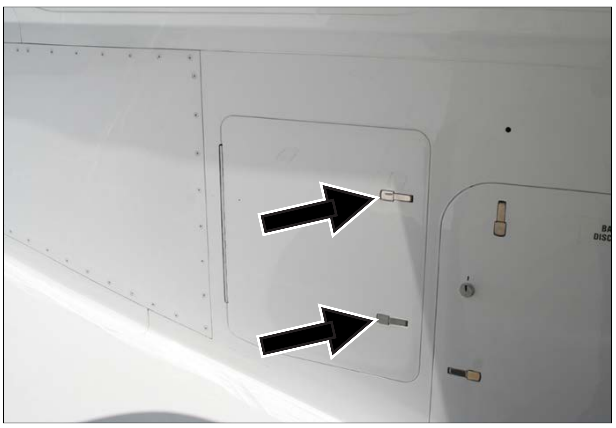


a. **BATTERY COOLING INTAKE AND VENT LINES -** CLEAR

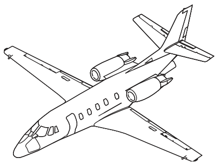
b. **BATTERY COMPARTMENT ACCESS DOOR -** SECURE/LOCKED

c. **BRAKE RESERVOIR OVERBOARD VENT LINE -** CLEAR

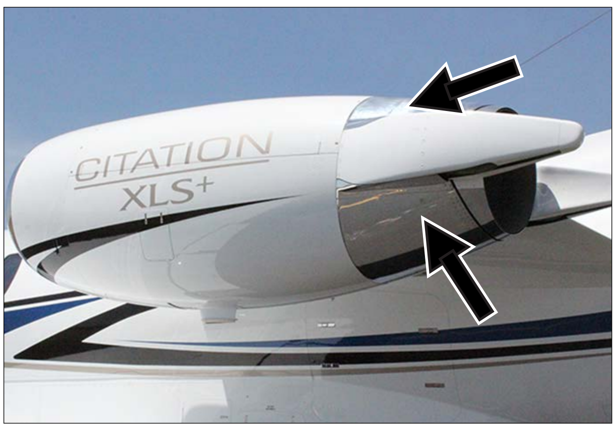
BRAKE COMPARTMENT ACCESS DOOR



13. BRAKE COMPARTMENT ACCESS DOOR - SECURE

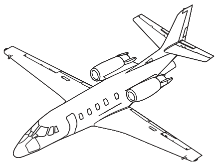


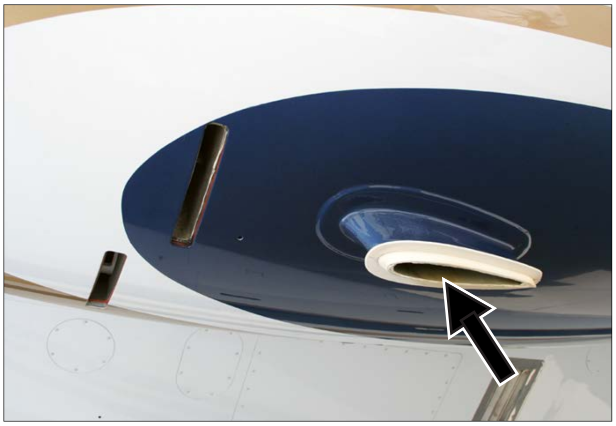
LEFT NACELLE/PYLON



a. THRUST REVERSER BUCKET - CONDITION/STOWED

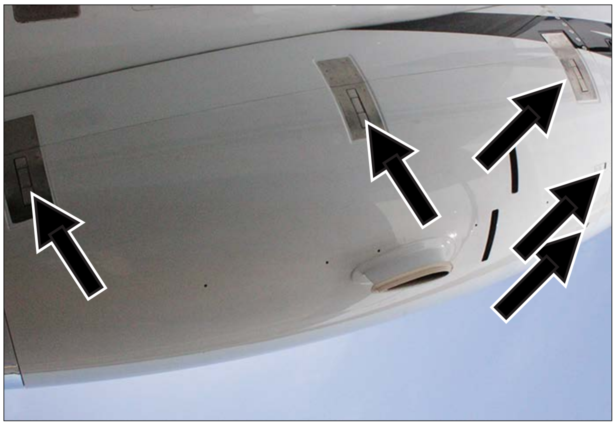
b. ENGINE EXHAUST AND BYPASS DUCT
- CONDITION/CLEAR





c. **ENGINE FLUID DRAIN MAST -** CLEAR

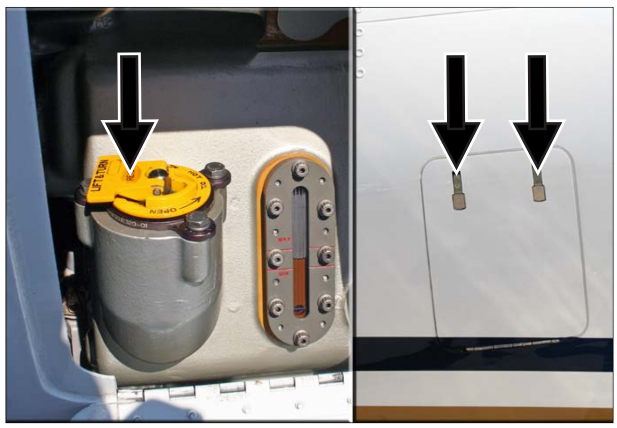
d. **GENERATOR AND ALTERNATOR COOLING AIR EXHAUST -** CLEAR



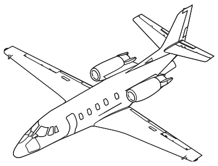
e. **COWLING (5 LATCHES) -** SECURE

f. **PYLON PRE-COOLER EXHAUST DUCT -** CLEAR

g. **OIL LEVEL -** CHECK

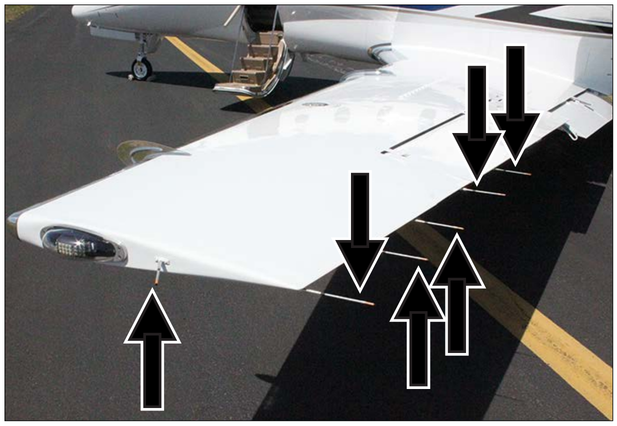


h. **FILLER CAP AND ACCESS DOOR -** SECURE

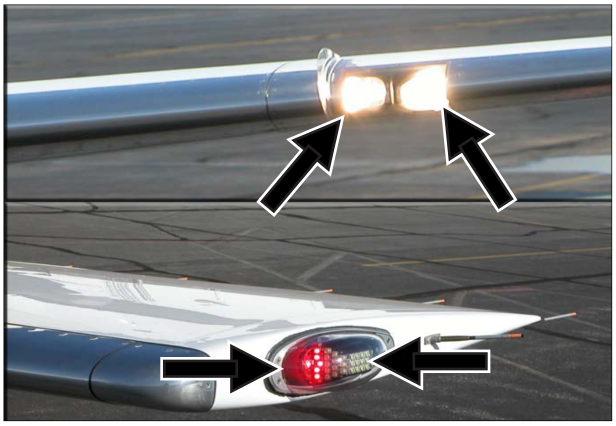


LEFT WING

a. FLAPS, SPEEDBRAKES, AILERON AND TRIM TAB - CONDITION



b. STATIC WICKS (6) - CHECK

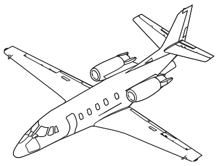


c. NAVIGATION. STROBE, LANDING AND RECOGNITION LIGHTS - CONDITION

d. FUEL TANK VENT - CLEAR

e. FUEL TANK RELIEF VALVES - CONDITION/NO LEAKS

f. FUEL FILLER CAP - SECURE



g. HEATED LEADING EDGE - CONDITION/VENT CLEAR

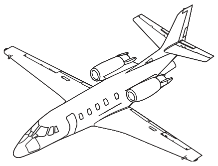
h. VORTEX GENERATORS (26) - CHECK

i. BOUNDARY LAYER ENERGIZERS (11) - CHECK

j. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE - CONDITION

j. MAIN GEAR DOOR, WHEEL, TIRE AND BRAKE - CONDITION

k. WHEEL WELL - CONDITION/NO LEAKS



l. FUEL TANK AND SUMP DRAINS (5) - DRAIN/CHECK

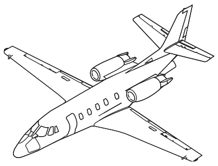
m. ANTI-ICE BLEED AIR COOLING AIR INLET - CLEAR

n. WING INSPECTION LIGHT - CONDITION

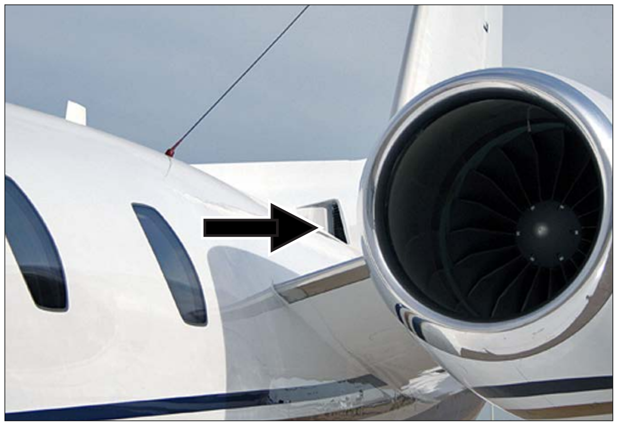
o. ENGINE FAN DUCT AND FAN - CONDITION

p. RAT PROBE (IN LEFT ENGINE INLET) - CONDITION

CABIN ENTRY



16. CABIN ENTRY - CHECK



a. DORSAL FIN AIR INLET - CLEAR

b. SECONDARY CABIN DOOR SEAL - CONDITION